NASA TECHNICAL MEMORANDUM

NASA TM X-53049

JUNE 12, 1964

-44p.

N64 26979 - N64 26993

Code 1 cat 29

Code-1-CAT-29

NASA TM X-53049

PROCEEDING OF THE SYMPOSIUM ON MANNED PLANETARY MISSIONS 1963/1964 STATUS

Future Projects Laboratory

OTS PRICE

XEROX $ 3.00ph

MICROFILM $ ______

NASA

George C. Marshall Space Flight Center, Huntsville, Alabama

FOREWORD

On January 28 through 30, 1964, the Future Projects Office of MSFC, under the chairmanship of Mr. J. N. Smith, sponsored a Symposium of Manned Planetary Mission Studies performed by industry for NASA in 1963 and 1964. The purpose of this symposium was to provide an overall view of the manned planetary mission study activities and an insight into the current state of the art.

The program was divided into four sessions and concluded with a panel discussion with questions from the floor. The session chairmen were Dr. H. O. Ruppe, MSFC; Mr. C. A. Syvertson, Ames; Mr. J. N. Smith, MSFC; Mr. C. R. Darwin, MSC; and Mr. V. Gradecak, MSFC. The panel discussion was moderated by Dr. H. H. Koelle, MSFC.

This document is a compilation of the individual papers presented during the symposium.

NASA-GEORGE C. MARSHALL SPACE FLIGHT CENTER

TECHNICAL MEMORANDUM X-53049

PROCEEDINGS OF THE SYMPOSIUM ON MANNED PLANETARY MISSIONS 1963/1964 STATUS

FUTURE PROJECTS OFFICE
RESEARCH AND DEVELOPMENT OPERATIONS

TABLE OF CONTENTS

TABLE OF CONTENTS (CONT'D)

PART 1

WELCOME TO GEORGE C. MARSHALL SPACE FLIGHT CENTER

by

Dr. Wernher von Braun, Director
Marshall Space Flight Center
Huntsville, Alabama

WELCOME TO GEORGE C. MARSHALL SPACE FLIGHT CENTER

By

Dr. Wernher von Braun, Director
Marshall Space Flight Center
Huntsville, Alabama

In the best Southern tradition I extend to each of you a very warm welcome to Alabama, to Huntsville, and particularly to the Marshall Space Flight Center, the home of Saturn. You know of course that the world's largest Moon rockets are the Saturn I, Saturn IB, and Saturn V.

I am just back from the Cape where we rather unsuccessfully tried to launch our first two-stage Saturn I, only to try again tomorrow morning. I think the reason for not getting it off the pad yesterday is important enough to be related to you here today, because it just shows how the most insignificant things can sometimes hold up great undertakings.

It turned out that somebody simply forgot to remove a blind flange from the liquid oxygen replenishing line in the ground support equipment, which is actually part of the installation of the permanent launch pedestal. We had almost filled the first stage with liquid oxygen. The last of the oxygen to go into the tank cannot be put in with the powerful feeding pumps that are used for fast filling, because there is great danger of over-filling the tank and bursting the bulkhead by bringing the entire fueling pump pressure to bear on the bulkhead. To avoid this possibility, we have a smaller replenishing, or pumping, line which is used to feed the last 10 per cent of LOX and keep it level in case of extended delay. Thus, we are always sure of launching with full tanks.

Well, you know that every small detail must be checked, in the rocket as well as in the ground support equipment, so this LOX pumping line had to be pressure tested. Now you cannot pressure test a line that opens into a tank, unless you want to put pressure on the tank, also.

Since the tank cannot take that much pressure, a blind flange is inserted in the end of the line and the line is tested. When the line was connected to the tank again, somebody forgot to remove the blind flange. We learned this only after the rocket was full of oxygen, and so we could not take the blind flange out again without emptying about 450,000 pounds of LOX on the launch pad.

Things like this are particularly embarrassing, of course, when you work in a gold fish bowl with "X" number of Congressmen and "X" number of corporation presidents present, plus all other NASA hierarchy with the exception of the boss himself. Well, there was nothing really serious about it except a few people were red faced. Things like this are relatively unimportant when you can work in the privacy of your own installation but not when you have ABC, NBC, and CBS present, and are on the air.

This incident, I think, is quite educational and helpful to this conference, because planetary launch windows are unforgiving. As you know, there is only a very limited period of time in which you can blast off for Venus or Mars. You must be on time, because those planets move on. We shall have a very good illustration of this fact tomorrow. Right after our launch at 10:00 in the morning there will be a four-day window for a Ranger shot at the Moon. And while we succeeded in pursuading the Air Force Missile people to postpone the Minuteman launching for a few days to give us another chance tomorrow morning, we will definitely not succeed in pursuading the Moon to wait a few more days so that Ranger could follow us. So, if we do not make it tomorrow, there will be a delay of several more days to give Ranger a crack.

I hope in due time we will learn how to launch on time. During the development of the smaller missiles we learned to reduce the launch preparations, including fueling and LOXing operations, to something like 15 minutes. Also there is no reason in the world why a Saturn cannot ultimately have a turn-around time comparable to that of a jet airliner, which is just about as complicated. And once we have demonstrated this, I hope there will be a market for these Saturn vehicles, and their successors which we shall build, in interplanetary exploration, also.

Now you propose to plan our next step for interplanetary exploration during these next three days at Huntsville, and I trust your stay here will be enjoyable as well as informative. It is gratifying to see such a good turnout for these study reviews. This indicates to me keen interest in the varied possibilities for manned exploration of the planets in the near future.

Interplanetary mission studies may seem somewhat premature to some people, particularly to those stubborn individuals who have been dragged, screaming and protesting violently, into the space age. Man has not yet traveled into space more than 200 miles from the Earth's surface and it will be some time before an astronaut or a cosmonaut steps out of a spacecraft and stands on the surface of the Moon, which is only 240,000 miles away. People have become accustomed today to four-hour trans-continental flights and they shudder to think of a four-month journey to Mars; but we should remember that the Dutch colonized the Indies, even though it took four years for a round trip by boat; and Marco Polo spent 24 years on his trip from Venice to China. We are not too early to begin serious study of the manned exploration of interplanetary space. When you consider that the time lapse, from the conceptual studies of the new launch vehicle to operational readiness, is considered to be approximately 10 years, we are definitely not early at all.

Feasibility studies, such as the ones to which you will be exposed during the next three days, have an important role in the planning of a program as large and complex as the nation's space effort. I shall mention only three of these benefits:

(1) Long Range Planning -- In any successful business, and believe me, NASA has grown too large to be a hobby, there must be competent program planning for the future. I can think of no better way to get a handle on these available vehicle systems and potential missions than well-supervised, well-coordinated feasibility studies.

(2) Time and Money Savings -- It is conceivable that a vehicle system without sufficient study could go all the way to hardware development before some part of it is found to be unworkable, or inadequate for the mission we have in mind. Or, in another extreme case, the system could be uprated from the beginning, and perform one or more additional missions, thereby saving the development of a new system.

(3) Stimulation -- Feasibility study reviews provide an opportunity for technical people to embrace the ideas that others have generated, to improve on them, or even to disagree with them; and the latter is sometimes the most important purpose of such a meeting as this. In any event, the problem areas are defined, and stimulation provided for deeper and broader study.

I would like to thank the Manned Spacecraft Center in Houston and the Ames Research Center for the cooperation with our Future Projects Office at Marshall in making all these studies available at one meeting. I am confident that many of them will be approved within our generation. You know that we, too, at Marshall have conducted a great number of studies, but let me point out that most of this effort was performed by industry.

Although our nation is firmly dedicated to achieving a manned lunar landing in this decade, a landing on the Moon is not an ultimate goal. Man will travel beyond the Moon to explore the solar systems. When, I do not know. But perhaps, after this symposium, we shall have a better idea of when man could conceivably venture out to Mars or Venus and return safely to Earth.

It should also become apparent that there is a tremendous amount of work to be done before such exploration becomes a reality. Let us define the problems, and work together toward their solutions. It will take all the disciplines of technology and management to push the frontiers of space into the backyards of the planets. Again, I hope that your visit here is both enjoyable and profitable.

Z6980

PART 2

A STUDY OF MANNED INTERPLANETARY MISSIONS

by

Dr. K. A. Ehricke
General Dynamics/Astronautics
Contract No. NAS8-5026

0 INTRODUCTION AND STUDY OBJECTIVES

01 Statement

This report summarizes the work performed under Contract NAS8-5026 and is submitted in partial fulfillment of technical documentation of the study. The work was performed by the Advanced Studies Office, General Dynamics/Astronautics under the cognizance of Dr. H. H. Koelle, Director, Future Projects Office, NASA/MSFC, and Dr. H. Ruppe, Deputy Director, Future Projects Office and Technical Manager of Contract NAS8-5026. The comments and recommendations by members of the Future Projects Office have been most helpful.

02 Study Objectives

The second phase of a Study of Early Manned Planetary Missions has been completed for the Future Projects Office of the NASA George C. Marshall Space Flight Center, Huntsville, Alabama. The primary study objectives were defined as follows:

A. A detailed definition of the mission profile of a fast trip to Mars in the 1975 time period. The auxiliary vehicles (i. e., manned landers, unmanned probes, etc.) to complete this mission profile should be considered as a secondary objective.

B. A preliminary design of a space vehicle system suitable for this mission profile, including Earth launch requirements, orbital operations requirements, nuclear engine requirements, scientific mission requirements and atmospheric re-entry requirements.

C. A compatibility study of this space vehicle system for other missions within the national space program.

D. The growth potential of the proposed space vehicle system

The expected results are to include the following:

A. Refinements of the analysis of the four basic mission modes investigated by GD/A in the first part of this study.

B. Refinement of the basic mission requirements in terms of weight, volume, power and other critical elements.

C. Refinement of launch window specifications for Earth and target planet.

D. Definition of abort and abort possibilities throughout the mission. Check list of the more probable emergency-type situations and how to cope with them.

E. Refinement and implementation of previous work done in convoy vehicle design and systems analysis.

F. Continued investigation of crew requirements.

G. Detailed study of the development plan for this mission. The preliminary development plan shall contain a cost estimate for the total mission.

03 Relationship to Other NASA Efforts

The relationship of manned planetary round-trip missions to other NASA efforts is surveyed in Fig. 0-1. The interrelation was divided into 6 basic areas.

(1) Destination payload, especially orbital reconnaissance equipment, data processing equipment, a variety of probes and the Mars excursion module (MEM)

(2) Propulsion system, design criteria and configuration of the interplanetary vehicle (I/V)

(3) Earth return conditions, particularly the state of the art in hyperbolic entry into the Earth atmosphere and in hyperbolic rendezvous with the returning I/V

(4) Earth launch vehicle (ELV) availability and characteristic constraints

(5) The supporting instrumented probe program with reference to Mariner, Voyager and roving interplanetary probes (RIP's)

(6) The manned space station program as the principal instrument for orbital development and testing of the ecological system and other life support equipment and for long-duration training of the mission crew. The manned space station (or the orbital laboratory) is the principal means of orbital development and testing of practically the entire operational payload of the I/V.

The individual areas are detailed further in Fig. 0-1. A distinction is made between contributory developments which presently add to the relevant state of the art and required research and development, both based on

conditions of FY-64. The contributory developments represent the principal foundation for an early "minimum-type" manned planetary mission.

It was established that development of a chemo-nuclear or all-nuclear I/V and the preparation of manned planetary flights would furnish the following contributions to other areas of astronautics:

1. At least one type of long-duration ecological system for a crew of about 8 persons operating over a period of 450 to 600 days.

2. Complete life support sections, modularized, which can be assembled in orbit to form a space station or on the Moon to form the nucleus of a base.

3. A lunar shuttle vehicle of a variable payload capability, depending primarily on the number of stages of the I/V configuration used.

4. Providing mission specifications and particular incentives for the development of nuclear engines.

5. Providing incentives and specifications for modifications of Saturn V and for the Post-Saturn ELV. Specifically, it was found that enlarging the diameter of Saturn V, in order to increase the length and volume of its payload section, is more important than increasing its payload by 10-20%, assuming hydrogen-based I/V's are being used.

1. MISSION ANALYSIS

1.1 Heliocentric Transfer Windows

Favorable transfer windows have been determined for

Earth	Venus, 1973-84
Earth	Mars, 1973-84
Earth	Mars, 1976-84

for transfer times ranging from 120 to 250 days (Fig. 1-1). The figure shows clearly that for round-trips between Earth and Mars and between Earth and Venus the favorable transfer windows are not in harmony. Upon arrival at the respective target planet (Venus, Earth or Mars), the opportunity for a favorable return flight has passed. For mono-elliptic transfers directly to the target planet one has, therefore a choice either to depart <u>ahead</u> of a given favorable transfer window, or to return <u>after</u> the respective favorable return transfer window.

It was found that the latter alternative is more attractive from an overall mission and vehicle systems point of view. The primary reasons for this preference are:

(1) The highest approach velocity is, in this case encountered upon return to Earth. Therefore, the largest amount of propellant can be saved here if the Earth return velocity is high and if Earth departure velocity and target planet approach and departure velocities are kept as low as possible.

(2) Most of the propellant is consumed by the time the vehicle starts on its return flight.

(3) Extreme conditions, such as a perihelion transit or an aphelion transit occur during the return transfers, because the outgoing transfer orbits are short if flown during a favorable window.

1.2 Venus Capture Missions

1.2.1 Heliocentric Mission Profiles

In May 1975 an Earth - Venus window begins to open up for short and medium transfer orbits (120 to 150 days). The window lasts through the middle of July. Beyond this time the Venus arrival velocity increases more rapidly than the Earth departure velocity. The next window to Venus begins to open up in November 1976 and phases out at the end of February 1977.

A return window, Venus - Earth opens up in June 1975. It phases out for short and medium transfer orbits in August 1975; which is too early to be useful for the vehicles which use the favorable outgoing window in June-July 1975. The window is extended to November, however for longer return orbits in the 200-250 day range. The next window opens up in February 1977 and begins to close in late March for short and medium transfer orbits, but stays open to early May for transfer orbits beyond 200 days.

The most suitable mission profile consists of a short transfer orbit to Venus, followed by a long return orbit whose aphelion lies beyond the Earth's orbit.

1.2.2 Venus Capture Orbits

In the case of Venus, an elliptic capture orbit has been selected. While this imposes some constraints on the planetary reconnaissance operations during the capture period, elliptic capture reduces the Venus capture mission energy requirement considerably. Some of this gain must be paid back when departing. Departing from any point in the elliptic orbit other than the periapsis increases the departure energy. If the departure point is considerably off the periapsis, it is more economical to rotate the major axis by temporarily entering a circular orbit at apoapsis distance which involves two additional maneuvers. Nevertheless, the sum of capture and departure maneuvers results in a net saving of energy, compared to capture in a circular orbit.

1.2.3 Reference Capture Mission

The reference mission profile to Venus, on which the sizing of the interplanetary vehicles is based, consists of a 150 to 120-day outbound orbit, a 20-day capture period and a 240-day return flight. The period of the $n = 8$ capture ellipse with $r_P^* = 1.3$ is 19 hours. Thus, the vehicle completes approximately 23 or 24 revolutions, depending on the length of the arc of the circular orbit (period = 49 hours) while rotating the major axis of the ellipse. The Venus departure process consists of two apoapsis maneuvers $2 \cdot \Delta v_A$ and a periapsis maneuver $\Delta v_{P,3}$ ($n = 8$), the latter increasing rapidly with

progressing time. Because of this, and in order to keep the Earth return velocity constant, the Venus arrival data was held constant to 11-5-75, limiting $\Delta v_{P,3}$ to 3.24 km/sec (10,200 ft/sec) and the Earth entry velocity to 14 km/sec. The mission profile is shown in Fig. 1-2. The associated characteristic mission data are listed in Tab. 1-1.

1.3 Mars Missions

1.3.1 Heliocentric Mission Profiles

Fig. 1-3 illustrates the change in mission profiles when leaving and returning early or late during the mission window. Mars missions change as follows upon gradual transition from early to late missions: Early missions (1, 2, 3, 4) are characterized by comparatively high Earth departure and Mars arrival velocities, by low Mars departure velocities and by medium Earth arrival velocities. The outgoing path passes through a perihelion inside the Earth orbit. The return path is short. The mission taking place late during the window (9, 10, 11, 12) is characterized by low Earth departure and Mars arrival velocities and by high Earth return velocities. The outgoing transfer path is short, the return path is long and leads through the perihelion of the transfer orbit. The perihelion lies closer to the Sun the later the return flight occurs. The later the return, the higher is the Earth return velocity due to an increasingly steep intersection angle between the return orbit of the space vehicles and the orbit of Earth. The lower the permissible re-entry velocity upon Earth return, the stronger must be the retro-maneuver for a given $v^*_{\infty 4}$. The overall mission velocity minimizes, therefore, for an earlier date during the mission window than if the retro-maneuver is small. The higher the permissible Earth re-entry velocity the later will be the date for minimum mission velocity.

From a comparison of the velocity requirements for the principal maneuvers (Fig. 1-4) and the perihelion distances encountered during Mars capture missions beginning in 1975, 1977 and 1979, the following results have been obtained:

(1) For an Earth departure window (EDW) of 30 to 50 days the minimum Earth departure impulse values are comparable in all three mission years. If outbound transfer periods from 160 to 190 days are acceptable, it is not necessary to provide for a higher Earth departure impulse than 0.15 EMOS (5 km/sec;15,000 ft/sec) in either mission year.

(2) The Mars arrival velocity requirement for capture in a circular orbit at $r^* = 1.3$, show the same trend in all three mission years. But the numerical values are higher in 1977 and 1979 than in 1975. They are highest in 1977, although the difference between 1977 and 1979 is small. If comparable Mars arrival velocity requirements are to be maintained in all cases, longer outbound flight times must be accepted for both, 1977 and 1979, than for 1975; i.e. 220 to 230 days in 1977, 200 to 210 days in 1979 compared with 180 to 190 days in 1975. The differences in outbound flight time do not appear to be crucial.

(3) The Mars departure velocities in the range of 220 to 240 days return transfer period are highest in 1975. While, in 1976, only the 230-day curve drops briefly below 0.2 EMOS, in 1978 the 220 and 230-day curves fall below 0.20 EMOS; and in 1980 transfer periods of 210 and longer lie temporarily below 0.20 EMOS.

(4) The Earth arrival velocities, represented by the entry velocities v_E^*, i.e. the velocity with which the vehicle arrives at an altitude of 100 km (distance 6470 km), are lowest in 1976 and highest in 1981. In all cases the velocities increase steeply with time, except for velocities in 1981; and in all cases the entry velocities are very high, i.e. between 0.55 and 0.65. Longer return flight times result in higher entry velocities than shorter transfer periods. The choice in transfer periods suggested by this fact tends to conflict with the suggestions for low Mars departure velocities which require selection of a long return flight time. This is particularly true in 1978, where the increase in entry velocity with time is strongest.

(5) The perihelion distances encountered in the outgoing leg of the mission are negligible.

(6) Considerably shorter perihelion distances are encountered during the return flight to Earth. As in (4), longer transfer times lead closer to the Sun. Perihelion distances between 0.4 and 0.5 must be tolerated by the vehicles following these mission profiles.

(7) The retro-maneuver velocity requirements associated with Earth entry velocity restrictions to 70,000, 50,000 ft/sec or Apollo conditions reflect the entry velocity characteristics. Unless entry velocities of 0.6 EMOS or higher are feasible, propulsion systems for considerable Earth retro-maneuvers must be taken along.

(8) Capability of close perihelion passage and high Earth entry velocities are, therefore, the two most important qualifications associated with these Mars mission profiles.

(9) The fact that, in 1977/78 and 1979/81 Mars is located in the vicinity of its aphelion tends to have a detrimental effect on the mission energy requirements, caused primarily by high Mars arrival velocities and secondarily by high Earth return velocities. However, by relaxing the mission period, particularly the outbound transfer period, and by attaining a high entry velocity capability, the effect of the unfavorable position of Mars during these missions can practically be eliminated. If, on the other hand a reduction in entry velocity to 0.5 EMOS or to Apollo conditions is required, then, and only then, are the 1977 and 1979-missions distincly more expensive than the 1975 and the 1982-missions.

1.3.2 Perihelion Braking (PB)

The path intersection angle at return crossing of the Earth orbit can be reduced significantly, and the return velocity lowered correspondingly, by slowing the vehicle down at the perihelion passage. At the small perihelion distances encountered (.45 to .55 AU), perihelion braking by 5000 to 1000 ft/sec causes a reduction in Earth arrival velocity by 10,000 to 20,000 ft/sec. Weight-wise, the effectiveness of PB is reduced by the fact that a heavier payload must be slowed down than near Earth where everything except the EEM is jettisoned. This effect can be overcome by making use of the proximity to the Sun through the application of a solar-thermal propulsion system.

1.3.3 Mars Capture Orbits

In the case of Mars, the energy saving due to capture in an elliptic orbit is far less pronounced than near Venus, because of the weaker gravitational field of Mars. Therefore, a circular capture orbit has been selected. Its standard distance is 1.3 planet radii. The departure maneuver from Mars

is likely to be a 3-burn maneuver. During the latter, necessary plane change, re-orientation of the major axis of the ellipse and adjustment of the periapsis distance take place, assuring a wide orbital launch window at comparatively smaller energy requirements than if these changes were carried out in the circular orbit proper. For best orbital reconnaissance and mapping, the capture orbit preferably is polar or near-polar. The orbital departure plane is close to the plane of the Mars orbit. Therefore, a considerable plane change will have to be negotiated prior to Mars departure.

1.3.4 Powered Fly-By (PFB) Missions

Investigation of fly-by missions established that powered fly-by, in contrast to gravitational fly-by only, increases the number of mission opportunities, for each mission opportunity broadens th departure window, results in many cases in shorter overall mission period and lower mission velocities, primarily, because a greater amount of orbit change can be affected than is possible with the comparatively weak g-fields of these planets alone. Thereby return orbits become available which have a lower hyperbolic excess velocity at Earth return. Examples are shown in Fig. 1-5.

Another advantage of powered fly-by is that it makes possible to utilize planetary fields for the modification of heliocentric orbit connecting two planets. Thus, when flying from Earth to Mars or from Mars to Earth, encounter with the gravitational field of Venus can be used for heliocentric midcourse changes for the purpose of reducing either the Earth departure velocity, the Mars arrival velocity, or the Earth arrival velocity. Under these conditions, however, the positions of Earth and Mars, the target planet are the determining factors. They do not necessarily correspond to conditions carefully selected to assure non-powered fly-by; e.g. they may lead too close to Venus. In those cases, powered maneuvers during the fly-by process provide the necessary flexibility.

The flexibility of the PFB missions is illustrated further by the fact that a trade-off is possible between Earth departure velocity and Earth entry velocity upon return. By varying the launch date between spring 1975 or 1977 and fall 1975 or 1977 the conditions can be varied from relatively high Earth departure velocities and low (0.5 EMOS) Earth entry velocities to low Earth departure velocities and close perihelion return orbits with high Earth entry velocities.

1.3.5 Synodic Missions

Synodic missions depart during a favorable transfer window and stay with the target planet until a favorable transfer window back to Earth occurs. On account of this, they always involve long capture periods. The transfer periods, however, may either be long or short. They are similar outgoing and

returning. The stay period is, therefore, equal to the synodic period between two similar transfer paths to and from the target planet. When very long transfer orbits (transfer angles larger than 180°) are used, the flights take place around the superior conjunction (Venus) or around the conjunction (Mars). Total mission periods are 800 - 1000 days. Hohmann-type missions belong in this group. If the transfer paths are short the waiting time for the return flight increases, so that the overall mission period stays long.

The hyperbolic excess velocities are very low in all cases, because favorable windows can be used both ways. Return velocities are so low that direct hyperbolic entry can be postulated. The transfer conditions improve in the 1975 through 1979 period and become somewhat less favorable in 1981/83. This trend is opposite to that observable for the fast missions where 1977 and 1979 are unfavorable years. For the synodic missions, these years are definitely favorable.

Because of their favorable characteristics (low mission velocities; comparative invariance of mission velocity from window to window) synodic capture missions were investigated briefly with the following principal preliminary results:

(1) The mission velocity, without any retro-maneuver at Earth return, is in the range of 0.3 to 0.4 EMOS (9-12 km/sec) and, therefore, in principle, accessible to the use of chemical propulsion systems.

(2) Mars arrival and Earth return velocities are little higher than parabolic. Therefore, use of atmospheric braking at Mars and, above all, application of the hyperbolic entry mode at Earth return appears engineering-wise less of a problem than for any other mission (typical Earth entry velocity without prior retro-maneuver: 0.389 EMOS or 11.6 km/sec). If only atmospheric use at Earth return, but not at Mars arrival is considered, the above quoted mission velocities are obtained.

(3) Typical <u>long-transfer synodic missions</u> (also called <u>conjunction missions</u>) have outbound and return transfer orbits of about one year each with a capture period at Mars of seven to eight months. <u>Short-transfer synodic missions</u> typically have transfer periods of 170 to 250 days outbound as well as return and capture periods between 400 and 500 days. The overall mission velocities are comparable, but tend to be slightly higher and less

invariant from mission window to mission window than the slow synodic missions, because their transfer angles are close to 180 degrees. Unless the conditions are given for a nodal transfer[1], these transfer orbits tend to have high angles of inclination relative to the elliptic.

(4) On long-transfer synodic missions most of the time is spent in transfer. This was found to increase the required intransit payload (crew size, spares, life support) to such extent that it tends to eliminate the advantage of lesser mission velocity as far as orbital departure weight (ODW) is concerned.

(5) Short-transfer synodic missions are well suited for extended Mars surface explorations (synodic base); but as such, are beyond the scope of an initial reconnaissance mission.

Windows for fast missions and synodic missions are shown in Fig. 1-6.

1.3.6 Reference Missions

A total of six reference missions was defined. These are presented in Tab. 1-2. Mission I represents the reference mission to Venus, described in Sect. 1.2.3. Missions II and III follow the same overall profile. Mission II involves capture in a circular orbit (circular capture, CC) only. Mission III includes, in addition, a surface excursion (SE). The Mars-bound transfer orbit of Mission IV is similar to that of Missions II and III; the circular capture operation is the same. But the Earth-bound transfer orbit passes through the gravitational field of Venus and, by means of a powered fly-by maneuver (PFB) alters the remainder of the return orbit in such a manner that considerably more favorable Earth arrival conditions are obtained at an extension of the mission duration by 160 days compared to Mission II or III. Mission V is a powered fly-by round-trip mission to Mars. Mission VI leads to a Venus PFB on the way to Mars rather than on return. The mission year is 1977 in this case.

There are modifications which were considered for almost all of the reference missions. Mission IIB (or IIIB), for instance, specifies a limiting hyperbolic entry velocity at Earth return of 15.25 km/sec (50,000 ft/sec or 0.512 EMOS). Mission IIF (or IIIF) specifies a perihelion brake (PB) maneuver

1) Mars located at one of the nodes of its orbit with the ecliptic at vehicle departure.

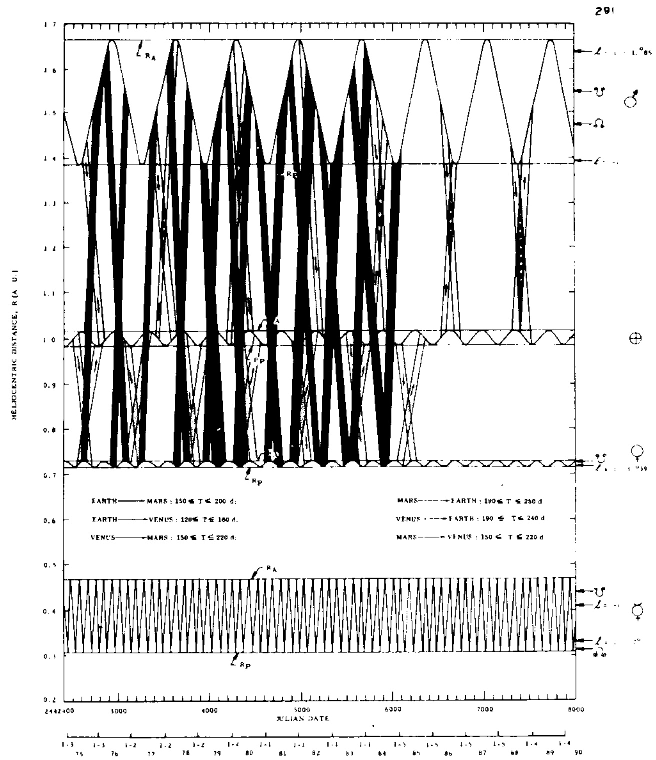


Fig. 1-1 TRANSFER CORRIDORS EARTH→VENUS, EARTH→MARS, VENUS→MARS

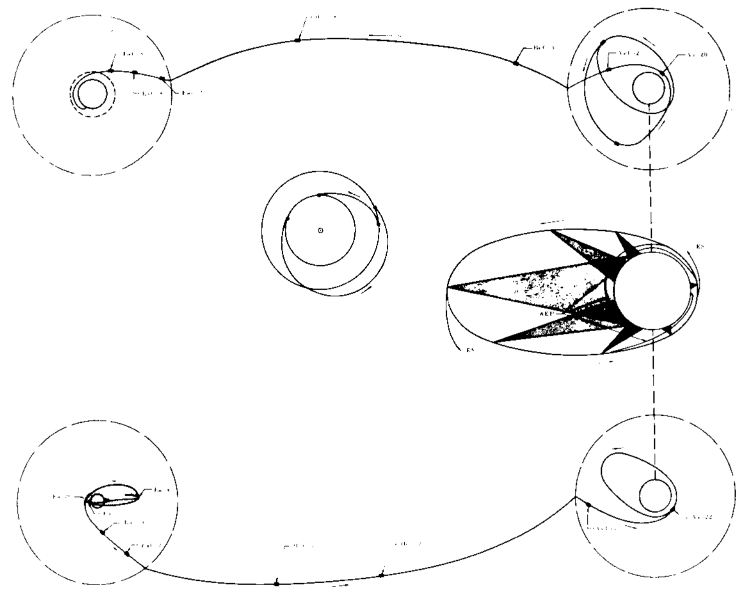


Fig. 1-2

Fig. 1-3

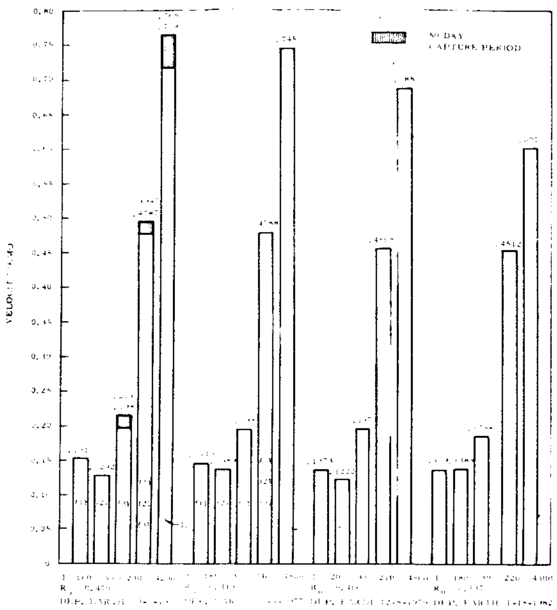


Fig. 1-4 VELOCITY DISTRIBUTION IN MARS MISSIONS 1975 THROUGH 1982 FOR MINIMUM ODW LAUNCH DATES

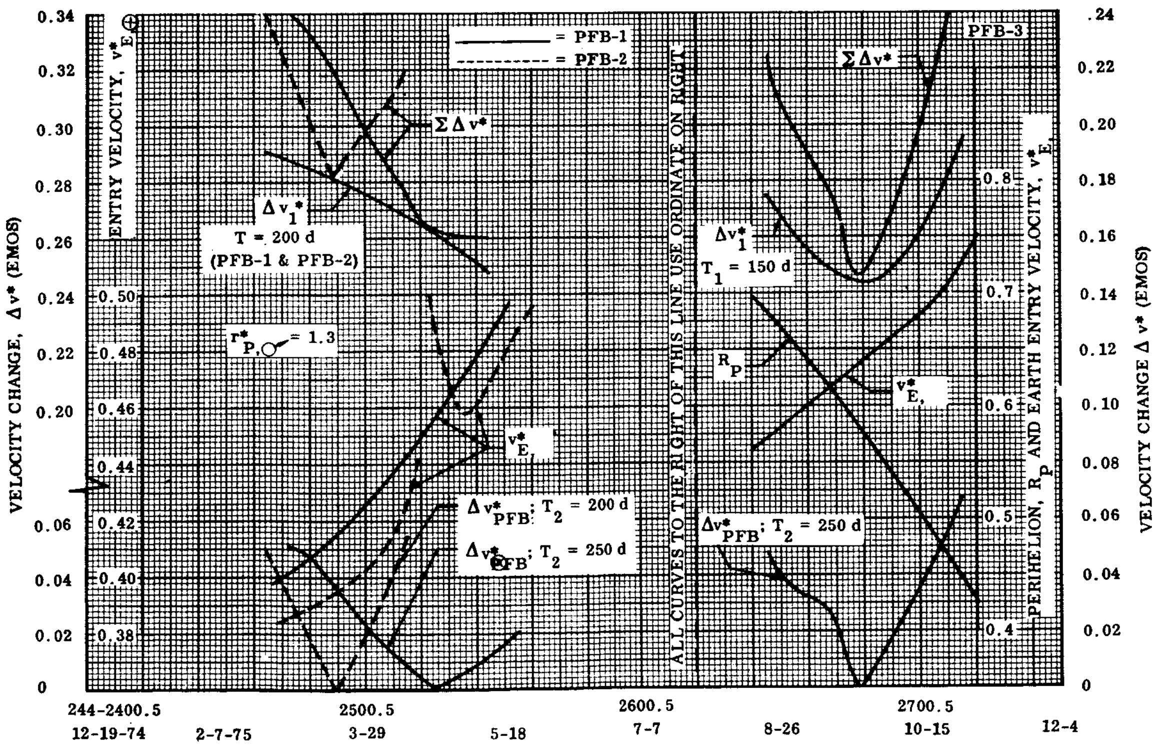


Fig. 1-5 THREE POWERED FLY-BY MISSIONS TO MARS IN 1975: PFB-1 PFB-2 AND PFB-2

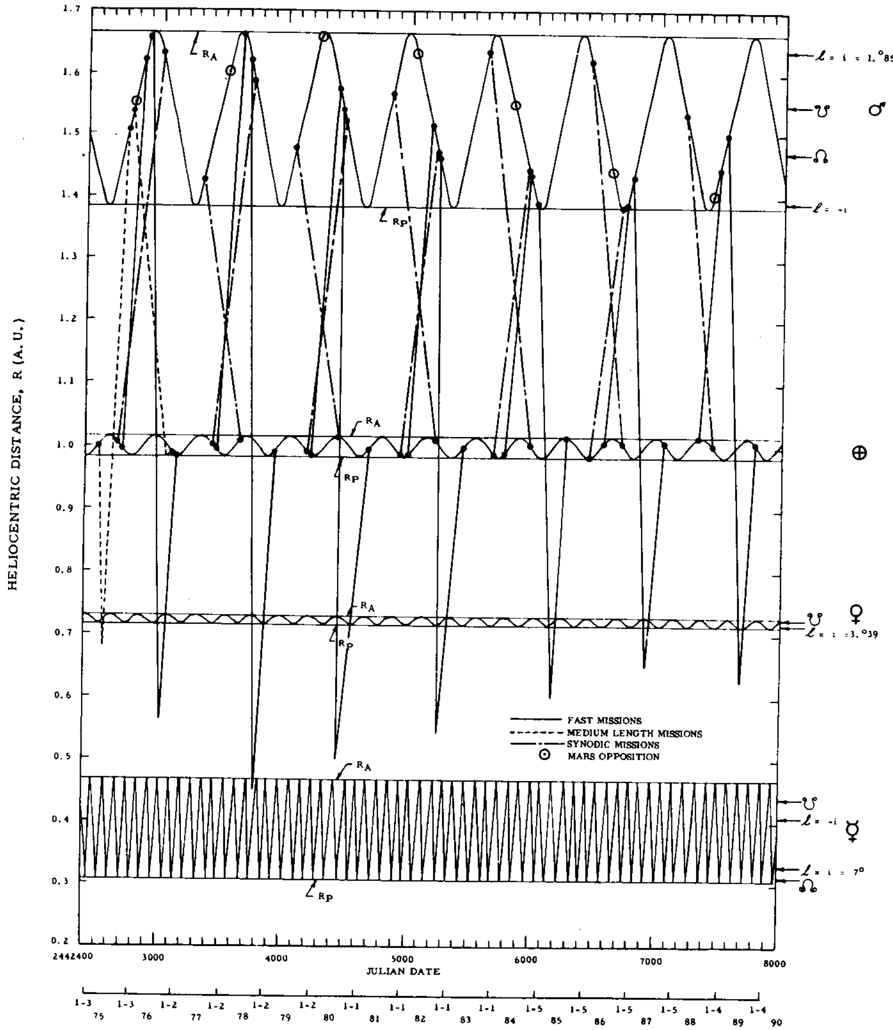


Fig. 1-6 WINDOWS FOR FAST AND SYNODIC ROUND-TRIP MISSIONS TO MARS

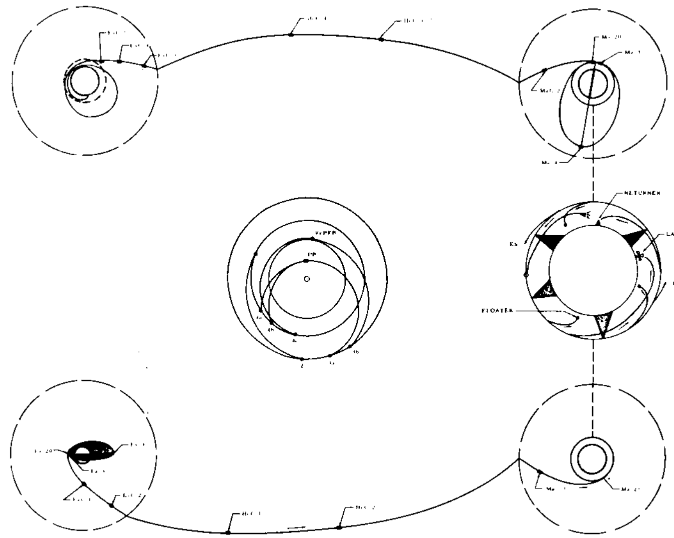


Fig. 1-7 MARS REFERENCE MISSION PROFILES M-II AND M-IV

during the Earth-bound coast and an unretarded hyperbolic entry into the Earth's atmosphere. Mission IVB, too, specifies Earth atmospheric entry without retro-maneuver. The mission profiles are illustrated in Fig. 1-7. Characteristic mission data are listed in Tab. 1-3.

1.4 Mars - Venus Bi-Planet Missions

It was found that, because of the relatively large difference in angular velocity between Venus and Mars, it is practically always possible to arrange a flight to Mars in such a manner that the planet Venus is encountered during the return flight to Earth. It is also possible to travel the other way around, i.e. first to Venus and from there to Mars. Opportunities either way around are indicated in Fig. The bi-planet missions are divided into bi-planet capture missions with capture occuring at both planets; and PFB/capture or capture/PFB missions with capture at one planet, powered fly-by (or powerless fly-by as boundary case) at the other; finally, there is the bi-planet PFB/PFB mission.

1.4.1 Bi-Planet Capture Missions

Bi-planet capture missions require less stringent timing than the other classes of bi-planet missions, since capture periods are inserted between transfers, which permit adaptation of the overall mission profile to favorable transfer windows between any two planets. The philosophy underlying the bi-planet capture missions is simply that, if a favorable transfer window does not exist between planets A and B, it may exist between A and C, and subsequently between C and B.

In 1975 it is possible to combine a flight to Mars with a return trip via Venus without unduly long capture periods until a favorable return window to Earth opens up. In 1977 the same mission cannot be flown under equally favorable conditions.

It is felt that bi-planet capture or capture/PFB missions of the type described below, rather than single-planet missions, will represent the actual future mode of interplanetary transportation. Unfortunately, for initial missions with energy-limited propulsion systems and severe limitations in Earth-to-orbit weight transportation these missions are too ambitious. They are very attractive in connection with later explorations using more advanced nuclear propulsion systems, such as the nuclear pulse or the gaseous core reactor engine.

1.4.2 Bi-Planet Capture/PFB Missions

The initially separate consideration of bi-planet (BP) missions on the one hand and PFB missions on the other, opened up a new field of inter-planetary mission profile research and lead to the development of BP missions characterized by capture at Mars and PFB at Venus for the dual purpose of raising the mission yield by visiting two planets in one mission and for reducing the Earth arrival velocity to values below 50,000 ft/sec without expending nearly as much propellant as in a retro-maneuver near Earth, designed to accomplish the same velocity reduction. Tab. 1-4 shows some of the results of the Venus PFB computation during return flight from Mars. The table compares the mission velocity and mission period resulting from the previously discussed single-planet Mars mission with the mission velocity obtained by Venus PFB, both reduced to the same Earth entry velocity, equal to that attained in the Venus PFB missions. It is seen that the combined effect of the Venusian gravitational field and thrust maneuver change the subsequent heliocentric orbit such that the vehicle should be able to enter the Earth atmosphere without further powered maneuvers other than required for path correction. This advantage is accompanied by a mission period penalty in excess of 25 percent. The 2-planet combination also is not always equally well applicable, because of constellational constraints. In any case, whenever it can be used, and this is the case for the majority of the Mars mission windows, it represents a very effective method of reducing the overall velocity requirement of Mars missions.

Mission	II	III	IV	V	VI
Target Planet	Mars	Mars	Ma/Ve	Mars	Ve/Ma
Dep. Earth	9-5-75	→	10-15-75	8-31-75	1-27-77
Transfer Period, T_p (d)	160	→	160	150	150
Planet Mode	CC	CC/SE	CC	PFB	PFB
Capture Period, T_{cpt} (d)	30	→	20	0	0
Departure Window, (d)	20	→	0	-	-
Transfer Period, T_2 (d)	-	→	200	-	200
Planet Mode	-	-	PFB	-	CC
Capture Period, T_{cpt} (d)	-	-	0	0	20
Departure Window, (d)	-	-	-	-	0
Transfer Period to Earth,(d)	220		200	250	220
Mission Period, T (d)	440	→	580	400	590
Earth Dep. Δv_1^* (EMOS)	.153	→	.164	.166	.1605
Target Pl. Arr. Δv_2^* (EMOS)	.1735	→	.1163	-	-
PFB, Δv^*_{PFB} (EMOS)	-	-	-	.0359	.0171
Target Pl. Dep., Δv_3^* (EMOS)	.200	→	.1725	-	-
Target Pl. Ar., Δv_4^* (EMOS)	-	-	-	-	.106
PFB, Δv^*_{PFB} (EMOS)	-	-	.00505	-	-
Target Pl. Dep., Δv_5^* (EMOS)	-	-	-	-	.169
Unbroken Earth Entry Vel. $\Delta v_{E,}$ (EMOS)	.74	→	.415	.592	.547
Earth Ar. Maneuver at Entry Vel. limited to $v_{E,}$ = .512, Δv_6^* (EMOS)	.236	→	0	.0825	.036
Mission Velocity $\Sigma \Delta v^*$ (EMOS)	.7625	→	.4578	.2844	.3886
(ft/sec)	74,500	→	44,600	27,800	38,000
(km/sec)	22.7	→	13.6	8.47	11.6

Tab. 1-2 MARS REFERENCE MISSIONS

	Dep. Ea	Transf.	Ar. Ve	Cpt. Period	Dep. Ve	Transf.	Arr. Ea
Cal. Date	5-20-75		10-7-75		10-27-75		6-24-76
Time (days)		140		20		240	
v_∞^* (EMOS)	0.1327		0.1400		0.2455		0.2824
Maneuvers, Δv:							
Principal (km/sec)	4.24		1.5		$2 \times 1.31^{1)}$ + $3 \times 24^{2)}$		$14^{3)}$
(ft/sec)	12,900		4,900		$2 \times 4300^{1)}$ + $10,200^{2)}$		$46,000^{3)}$
Corrections (km/sec) (ft/sec)		0.0915 300		0.122 400		0.0915 300	
Plane Change (km/sec)					0.615 (I_{sp} = 450) 1.050 (I_{sp} = 765		
(ft/sec)					2020 3440		

1) Twice apoapsis maneuver; 2) Planet departure from periapsis (capture ellipse: n = 8)
3) Hyperbolic entry velocity; no retro-maneuver required

Tab. 1-1 REFERENCE MISSION I (VENUS 1975, ELL. CAPT. n = 8)

	Dep. Ea.	Transfer	Ar. Ma	Cpt. Period	Dep. Ma	Transfer	PB	PFB Ve	Transfer	Ar. Ea.	Totals
Mission II B: Cal. Date Time (days) Δv^* (EMOS) Corrections (km/sec) (ft/sec) Plane Change(km/sec) (ft/sec)	9-5-75 0.153	160 0.091 300	2-22-76 0.1735	30 0.122 400 1.0 3200	3-13-76 0.200	230 0.091 300				10-29-76 0.236 ($v_{E,\oplus}^*$ = 0.512)	420 d 0.7625 EMOS = 74,500 ft/sec = 22.7 km/sec
Mission II F: Cal. Date Time (days) Δv^* (EMOS) Corrections (km/sec) (ft/sec) Plane Change(km/sec) (ft/sec)	9-5-75 0.153	160 0.091 300	2-22-76 0.1735	30 0.122 400 1.0 3200	3-13-76 0.200	180 0.091 300	9-9-76 0.08		80	11-28-76 0 ($v_{E,\oplus}^*$ =0.512)	450 d 0.6065 EMOS = 59,400 ft/sec = 18.1 km/sec
Mission IV B: Cal. Date Time (days) Δv^* (EMOS) Corrections (km/sec) (ft/sec) Plane Change(km/sec) (ft/sec)	10-15-75 0.164	160 0.091 300 1.0 3200	3-23-76 0.1163	20 0.122 400	4-12-76 0.1725	200 0.091 300	0.00505	10-29-76	200 0.091 300	6-17-77 0 ($v_{E,\oplus}^*$ = 0.415)	580 d 0.45785 EMOS = 44,700 ft/sec = 13.65 km/sec

Tab. 1-3 MARS MISSIONS II B, II F AND IV B (MISSION II B = MISSION II WITH HYPERBOLIC ENTRY VELOCITY OF 50,000 FT/SEC = 15.3 KM/SEC)

Dep $\oplus$	T_1 (d)	Δv_1 (ft/sec)	Δv_2 (ft/sec)	T_{cpt} (d)	Dep ♂	T_2 (d)	Δv_3 (ft/sec)	Δv_{PFB} (ft/sec)	$\Delta v_{E\oplus}$ (ft/sec)	Δv_4 (ft/sec)	$\Sigma \Delta v$ (ft/sec)	T (d)
9-5-75	160 "	14,933 "	16,978 "	30 "	3-13-76 "	230 200/180	19,976 13,800	- 285	64,361 43,700	20,661 0	72,584 45,711	420 570
"	160 "	" "	" "	30 & 20 "	4-2-76 "	230 190/160	19,416 13,400	- 2448	68,189 41,700	27,489 0	78,816 47,759	450 560
10-5-75 "	160 "	14,798 "	12,617 "	30 "	4-12-76 "	230 1[illegible]0/160	19,350 12,95[illegible]	- 21[illegible]	74,786 [illegible]1,600	33,186 0	75,951 38,563	420 530
"	" "	" "	" "	30 & 20 "	5-2-76 "	230 220/230	20,081 13,200	- 2345	74,786 45,000	29,786 0	77,282 42,960	450 660

Tab. 1-4 COMPARISON OF SINGLE PLANET MARS MISSION WITH BI-PLANET MARS CAPTURE, VENUS PFB WITH BI-PLANET MARS CAPTURE, VENUS PFB MISSIONS

2. INTERPLANETARY VEHICLE SURVEY

2.1 Vehicle Classes, Based on Solid Core Reactor Engines

A variety of vehicle configurations was developed. All configurations to Mars are propelled by nuclear engines, except for Earth return retro-maneuvers in some cases. Vehicles to Venus involve nuclear, combinations of chemical and nuclear propulsion modules and all-chemical vehicles. The design principles which are similar in all cases are shown in Fig. 2-1. The vehicle consists of propulsion section and life support section or service section. In the early manned planetary vehicles, the life support section contains essentially the operational payload and the intransit payload; the service section contains the destination payload.

The vehicle is propelled by nuclear solid core reactor engines, using liquid hydrogen. Engines and hydrogen tanks are arranged in a number of stages which are associated with the major powered maneuvers of the mission. Chemical propellants consist of oxygen and hydrogen.

Nuclear engines using graphite reactors and metal-clad reactors were considered. According to NASA's main development effort on nuclear engines, solid core reactor graphite engines (SCR/G) were given primary consideration, namely, a Nerva-type engine of 765 sec and 63k thrust (SCR/G-1 and a follow-on engine of 825 sec specific impulse and 250k thrust (SCR/G-2)). A fast neutron metal-clad engine (SCR/M) of 50k thrust and 900 sec specific impulse was selected as example of an advanced solid core reactor, which is not only characterized by higher specific impulse, but, in addition, by potentially superior operational lifetime and multi-start capability. Such characteristics are at least as important as the I_{sp}-increase, for planetary operations. Application of the fast neutron engine does not require separate engines for each main maneuver, as required for graphite reactor engines of limited operating life; two fast neutron engines are used for all maneuvers except Earth departure.

Hydrogen tanks, or combinations of tanks and engines are jettisoned as the tanks are emptied. Each propulsion module is surrounded by a combination heat and meteorite protection shield which is jettisoned just prior to ignition of the particular module. By this means, a high mass fraction is obtained for the operating propulsion module. Engines are located underneath hydrogen tanks so as to keep at all times as much hydrogen as available between nuclear engine and front section of the vehicle.

The vehicle design in this study phase took into consideration four major factors:

(1) Diameter constraints
(2) Engine operating life limitations
(3) Transport mode
(4) Vehicle assembly mode

Principal diameter restrictions were imposed where the interplanetary vehicle (I/V) modes had to be Saturn V compatible. In this case the diameter limit was 33 ft. Because this represented a considerable constraint, a hypothetical 50 ft diameter Saturn VM was introduced, to study the effect of relaxed diameter restrictions. Tandem arrangement of the individual propulsion modules, as shown in Fig. 2-1 was applied only where dictated by diameter restrictions. In absence of diameter restrictions, central tanks surrounded by satellite tanks were selected, both, because of the flexibility offered by this design approach, as well as because of reduced sensitivity to meteoritic damage.

If the operating life of the engine is limited, due to heavy fuel element erosion by the hydrogen, to the order of one hour, it becomes necessary, for reasons of vehicle weight, velocity requirements and thrust limitations (up to 250k considered) to provide separate engines for each major propulsion phase. If longer operating times are feasible, engines of lower thrust, lighter weight and less demanding after-cooling requirements can be used for several or all principal maneuvers during the mission. The fast neutron engine (SCR/M) was used in this capacity for all maneuvers following Earth escape.

The term "transportation mode" refers to the alternatives of transporting the payload in a convoy of two or more vehicles; or of transporting it in a composite vehicle, briefly referred to as multiplex vehicles. The multiplex vehicle concept was developed as an alternative to the single vehicle traveling in a convoy in which crew vehicles and cargo carrying service vehicles are separate. If crew facilities and cargo are to be combined, e.g. for reasons of better cargo maintenance, it is possible to envision just a larger version of any one of the above described vehicle classes. This single vehicle mode is unsatisfactory inasmuch as this mode denies the crew the opportunity to move themselves in a back-up vehicle in case the original vehicle is incapacitated. This is a serious disadvantage at least of the vehicle category presently under consideration, which is of limited payload and performance capability. In the multiplex mode, the individual convoy vehicles are clustered to form one vehicle which can be taken apart---if portions are damaged and must be abandoned---without necessarily impeding the capability of the remaining system to function as a crew vehicle.

This work has led to standardization of solid core reactor engine driven interplanetary vehicles (I/V's) into six classes. Four of these belong to single vehicles (Fig. 2-2) to be gathered in orbit to form a convoy.

Class	Description of Interplanetary Propulsion Modules (Excl. Earth Departure Module (PM-1)
-22	Tank cluster arrangement, consisting of control tanks (surrounded by satellite tanks). Thrust is provided by separate graphite core engines for each major maneuver, M-2, M-3 and by chemical engines for M-4 (if any).
-23	Tank cluster arrangement, similar in principal to those of -22, except that the clustered tanks are more nearly of equal diameter. Thrust is provided by one pair of metal core engines, common to those maneuvers M-2, M-3 and M-4 (i.e. all major maneuvers, except Earth departure).

The propulsion module PM-1 for Earth departure (maneuver M-1) consists of a single tank powered by one or more graphite core reactor engines. The overall diameter of the PM-1 tank and the clustered tanks is equal to, or in excess of 50 ft.

-28	Single tank arrangement in tandem for all propulsion modules, including PM-1. Separate graphite engines are used for each principal maneuver. Tank diameter: 50 ft.
-28V	Same as -28, but tank diameter is restricted to 33 ft to make it compatible with Saturn V.

The residual two classes belong to multiplex vehicles. The multiplex vehicle can be conceived as duplex or as triplex.

-26	Duplex: Two vehicles of the -28 or -28V class are coupled. The life support section (LSS) originally centrally mounted forward of the two tank rows can be swung or rotated over either tank row.
-27	Triplex: Three smaller vehicles are coupled together.

The triplex is less advantageous from the standpoint of LSS integration with the propulsion modules (PM); and from the standpoint of crew protection against radiation fromthe nuclear reactors. But it shows superior flexibility over the duplex at mission energy levels where only small engines (such as NERVA) are available. In the other cases, the duplex is weightwise superior.

In the single-vehicle classes -22, -23, -28 and -28V single tanks were used for the Earth departure propulsion module PM-1. In the case of 28V this is caused by the restriction to 33 ft tank diameters. Tank clustering at this diameter involves a considerable weight penalty.

For larger diameter configurations, such as the -28, -22 and -23 classes, it pays in terms of overall weight and reduced meteoroid sensitivity to cluster the tanks. This has been done consistently in PM-2 and PM-3 of the -22 and -23 classes and partly in PM-2 and PM-3 of the -28 class. The Earth departure propulsion modules have been provided with single tanks, because excessive length of PM-1 can interfere more disturbingly with the orbit delivery of PM-1 of the -28 class as a unit atop the 50-ft diameter (hypothetical) Saturn VM; or with orbit delivery of the entire -22 or -23 vehicle as a unit atop a post-Saturn vehicle. Therefore the investigation of the PM-1 tank configuration was postponed until a better vehicle-mission integration had been achieved. This was the case by the time the present follow-on study was almost completed.

The mode of assemblying the interplanetary vehicle determines its orbital departure weight (ODW). The designation of the four principal vehicle-assembly modes is defined in Tab. 2-1.

In the OVAM (Fig. 2-3), the individual interplanetary vehicle (I/V) is assembled in Earth orbit to a condition of complete self-sufficiency as far as its capability of completing the mission is concerned. If a retro-maneuver is required at Earth return prior to atmospheric entry, only the Earth entry module is slowed down.

In the IVAM and COVAM, the vehicle is sent on the mission in sections.

In the IVAM one vehicle carries the propulsion module PM-2 for the target planet capture maneuver (PM-2 Carrier); the second vehicle (Crew Carrier) carries the crew section, PM-3 and PM-4, if the latter is needed. Following injection into the Earth escape orbit, the burned-out PM-1 modules are jettisoned and the two payloads are combined to form a crew vehicle which is now capable of completing the mission without further orbital manipulations.

In the COVAM, the crew vehicle also consists of two vehicles at orbit departure, namely the PM-3 Carrier, carrying the propulsion module PM-3 for departure from the target planet and also PM-4, if one is needed; and the Crew Carrier, transporting the crew section. Therefore, each of the departing vehicles is equipped with a PM-1 and a PM-2,in contrast to IVAM, which requires duplication only in PM-1. The two vehicles stay separate until target planet capture conditions are readied. Following capture maneuver, the crew section is separated from the burned-out PM-2 and is attached to PM-3 (which jettisoned its burned-out PM-2), to form a vehicle which is capable of returning to Earth.

If the service vehicle is to accompany the crew vehicle to the target planet only, the above described modes, designated by the number (2), are modified and simplified in some respects.

The modification of OVAM, referred to as OVAM (2), involves the assembly of a service vehicle which is considerably smaller than the crew vehicle, or which could use chemical instead of nuclear propulsion, assumed in most cases for the crew vehicle. This alone would eliminate many operational hazards and problems associated with the use of a nuclear powered companion ship.

In the modified IVAM, referred to as IVAM (2) (Fig. 2-4), the initial number of vehicles for a convoy of two OVAM vehicles is reduced to three, namely, the Crew Carrier, the M-3 Carrier and the Service Carrier. The Crew Carrier consists only of the crew section and a PM-1. The two other vehicles carry their payload (the PM-3 and the service section, respectively), "on top" of a propulsion section consisting of PM-1 and PM-2 each. Following injection into the departure hyperbola, the crew section is transferred to the M-3 Carrier which now becomes the Crew Carrier for the remainder of the mission.

In COVAM (2) (Fig. 2-5), crew section and service section are combined. By doing so, two additional advantages are obtained; the number of departing vehicles is reduced to two; and the combination of crew and service payload simplifies the reconnaissance operations during the capture period. The two departing vehicles are the M-3 Carrier and the C/S-Carrier (Crew/Service-Carrier). The vehicles travel to the target planet and carry out the capture maneuver. The crew section remains connected with the service section throughout the nominal capture period. At the beginning of readiness operations for target planet departure the crew section separates from the service section and attaches itself to M-3.

Subsequently, the Venus and Mars Vehicles are surveyed. The propulsion modes of both vehicle groups are summarized in Tab. 2-1 and Tab. 2-2. The vehicle weights are shown in Fig. 2-6. The resulting trend

in the variation of orbit departure weight (ODW) with mission velocity is shown in Fig. 2-7. The width of the band reflects the opportunities offered by variations in mission profile and by the use of a limited variety of engines, as well as the effect of the vehicle assembly mode. Inclusion of hyperbolic Earth entry velocities in excess of 15.25 km/sec (50,000 ft/sec) or application of aerodynamic braking at Venus or Mars would extend the band width further in the direction of lower weights (cf. Fig. 5-11 of Vol. I Condensed Summary).

2.2 Venus Mission Vehicles

A series of Venus vehicles was investigated for reference Mission I.

The nuclear engine considered is of the "first generation" type, SCR/G-1. Three hypothetical versions were considered:

Model 1 (SCR/G-1.1):	Thrust = 56,000 lb I_{sp} = 763 sec No. of restarts: 0, 1, 2
Model 2 (SCR/G-1.2):	Thrust = 63,000 lb I_{sp} = 790 sec No. of restarts: 0, 1, 2
Model 3 (SCR/G-1.3):	Thrust = 102,000 lb I_{sp} = 779 sec No.of restarts: 0, 1, 2

These data cover a performance range postulated to be representative of a hypothetical early first-generation solid core reactor nuclear engine. Two ranges of operating life were considered for all models: 1200 to 1500 sec and 2100 to 3600 sec. Purpose of the variation of burning time, thrust, specific impulse and restart characteristics was to assess their effect on the ODW of the Venus vehicles and thereby, indirectly, also on Mars vehicles, to provide recommendations from the mission point of view regarding areas of emphasis in the development program of first-generation nuclear engines.

Seven propulsion mode configurations were studied, ranging from all-chemical to all-nuclear configurations and covering a variety of combinations of chemical and nuclear propulsion modes. For the Venus elliptic capture mission, 8-man (8) Venus (V) vehicles with a variety of propulsion modes were considered as shown in Tab. 2-1 for all principal maneuvers except Earth return for which the approach velocity has been limited to 46,000 ft/sec to be able to postulate aerodynamic capture and descent.

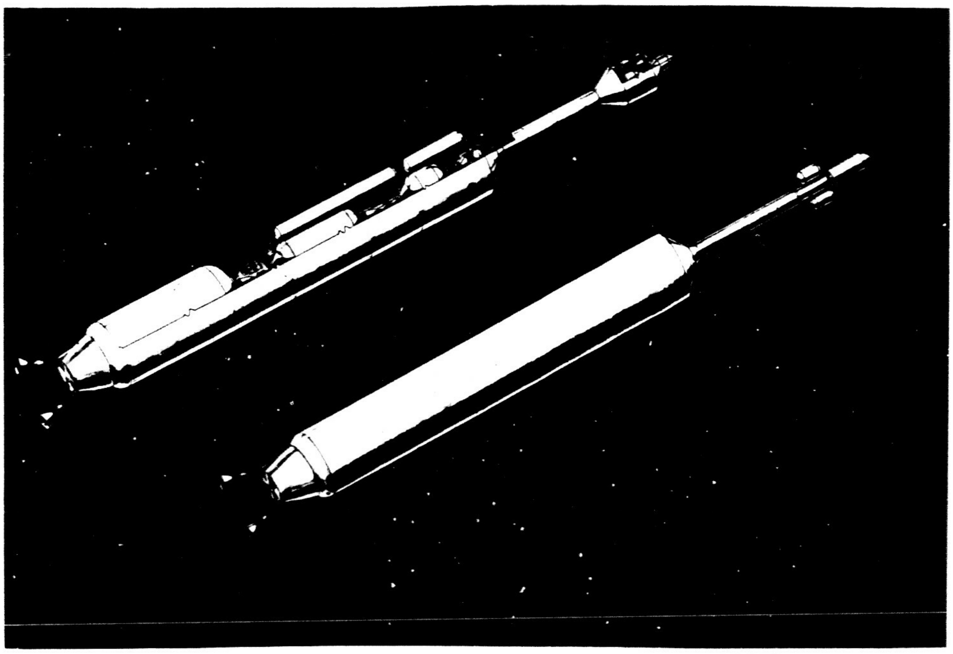


Fig. 2-1 CONVOY CONSISTING OF CREW VEHICLE AND SERVICE VEHICLE (CUT-AWAY)

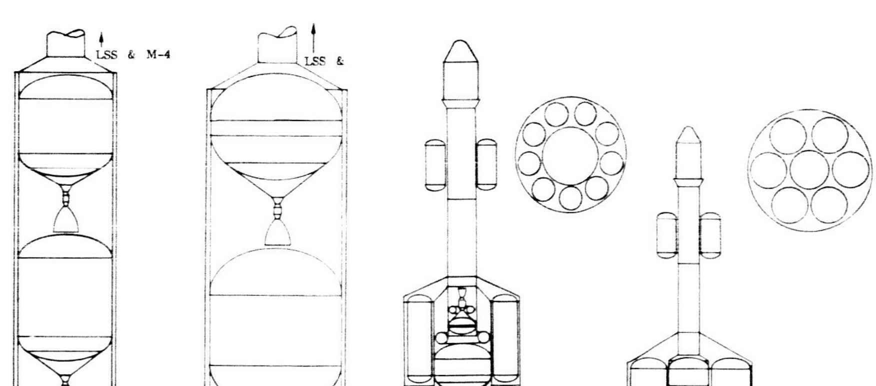
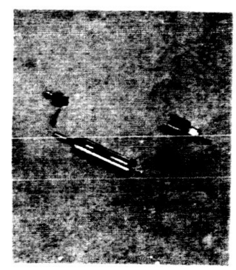


Fig. 2-2 FOUR BASIC CONFIGURATIONS FOR INTERPLANETARY VEHICLES

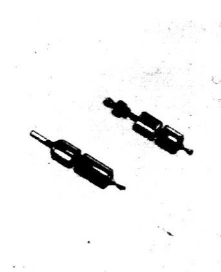
Nuclear Powered Vehicle, -23 Config., to Mars
Chemical Vehicle to Venus

Fig. 2-3



Transfer of LSS During Outbound Coast

Fig. 2-4



Earth Departure Condition

Fig. 2-5

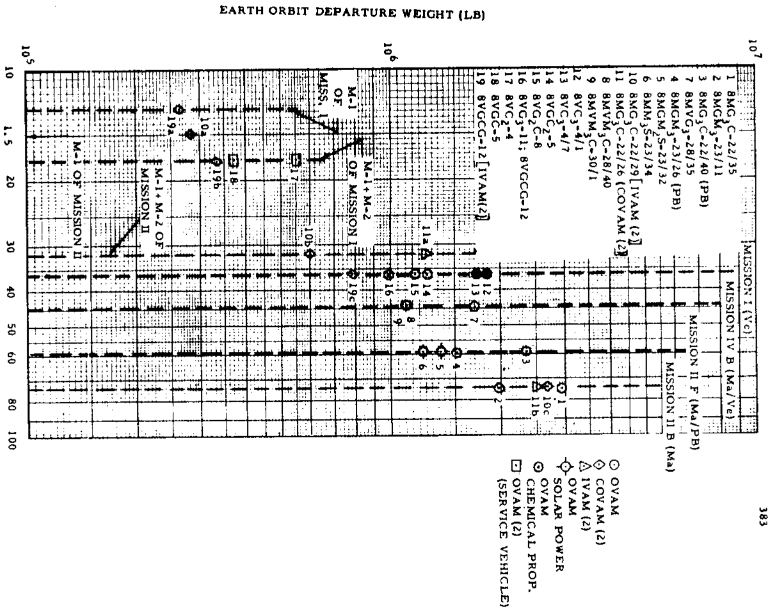


Fig. 2-6 ORBITAL DEPARTURE WEIGHT AND MISSION VELOCITY CORRELATION OF VEHICLES DESCRIBED IN THIS VOLUME

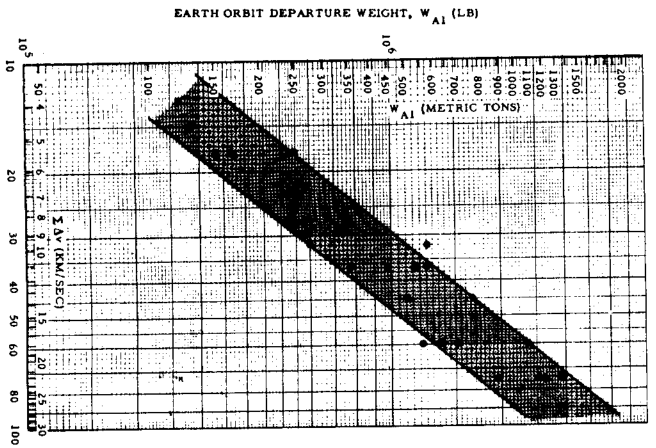


Fig. 2-7 VARIATION OF ORBITAL DEPARTURE WEIGHTS WITH MISSION VELOCITY

Configuration	Earth Dep.	Mars Ar.	Mars Dep.	Perih. Brake	Venus PFB	Earth Ar.	Mission
$8MG_3C$-22/35	2 SCR/G-2[1)]	1 SCR/G-2	1 SCR/G-2			O_2/H_2	II B[2)]
$8MGM_3$-23/11	2 SCR/G-2	2 SCR/M-1[3)]	Mars Ar. Eng's.			Mars. Ar. Eng's.	II B
$8MG_3C$-22/40	2 SCR/G-2	1 SCR/G-2	1 SCR/G-2	O_2/H_2		HE w/o Retro	II F
$8MM_3S$-23/34	4 SCR/M-1	2 SCR/M-1	Mars Ar. Eng's.	Solar (H_2)		HE w/o Retro	II F
$8MVG_3C$-28/35	4 SCR/G-1[4)]	2 SCR/G-1	1 SCR/G-1		O_2/H_2	HE w/o Retro	IV B
$8MVM_3C$-28/40	4 SCR/M-1	2 SCR/M-1	Mars Ar. Eng's.		O_2/H_2	HE w/o Retro	IV B
$8MVM_3C$-20/1	4 SCR/M-1	2 Ea. Dep. Eng's.	Mars Ar. Eng's.		O_2/H_2	HE w/o Retro	IV B
$8MG_3C$-22/26 (COVAM C/S-Carrier)	1 SCR/G-2	1 SCR/G-2					II B
$8MG_3C$-22/26 (COVAM PM-3 Carrier)	2 SCR/G-2	1 SCR/g-2	1 SCR/G-2			O_2/H_2	II B
$8MG_3C$-22/29 (IVAM Crew Carrier)	1 SCR/G-2						II B
$8MG_3C$-22/29 (IVAM Service Carrier)	1 SCR/G-2	1 SCR/G-2					II B
$8MG_3C$-22/29 (IVAM PM-3 Carrier)	2 SCR/G-2	1 SCR/G-2	1 SCR/G-2			O_2/H_2	II B

8M = 8-man mission to Mars; G = graphite reactor engine (SCR/G); M = metal-base engine (SCR/M)

1) SCR/G-2 = "Second generation" graphite-base engine; F = 250k
2) Mission II with 50,000 ft/sec hyperbolic entry speed
3) SCR/M-1 = "First generation" metal-base engine; F = 50k
4) SCR/G-1 = "First generation" graphite-base engine; F = 63k

Tab. 2-2 SURVEY OF MARS VEHICLES BY PROPULSION MODELS

Configuration	Earth Dep.	Venus Ar.	2 Apo-Maneuvers	1 Peri-Maneuver	Remarks
$8VC_3$-4	Chemical	Chemical	Chemical	Chemical	O_2/H_2; I_{sp} = 450 sec.
$8VGC_2$-5	Nuclear	Chemical	Chemical	Chemical	
$8VG_2C$-8	Nuclear	Nuclear	Chemical	Chemical	
$8VG_2$(CG)-9	Nuclear	Nuclear	Chemical	Nuclear	
8VGC(CG)-10	Nuclear	Chemical	Chemical	Nuclear	
$8VG_3$-11	Nuclear	Nuclear	Nuclear	Nuclear	
8VGCG-12	Nuclear	Chemical	Nuclear	Nuclear	
Δv (km/sec) (ft/sec)	4.24 12,900	1.5 4900	2.36* + 1.31 7740* + 4300	3.24 10,200	

8 = Crew size; V = Venus; G = Graphite engine (SCR/G); C = Chemical engine (O_2/H_2)

* incl. plane change provision (cf. Tab. 16-2)

Tab. 2-1 SURVEY OF VENUS MISSION VEHICLES BY PROPULSION MODES (MISSION I, VENUS, 1975, ELL. CAPT., n = 8)

The fact that nuclear propulsion is not needed for the Venus capture maneuver into an n = 8 elliptic orbit, (for the specific engines under consideration) is operationally advantageous for two reasons: The convoy vehicles (if it is a convoy) can be close together during the maneuver, since there is no problem of side radiation from nuclear reactors which either would force the vehicles at considerable distance from each other or into a tightly controlled specific formation during the powered maneuver and for a while thereafter. Secondly, no "hot" nuclear stages are moving around Venus in closed orbits which are similar to the capture orbits of the interplanetary vehicles.

For most powered fly-by mission to Mars chemical propulsion for the fly-by maneuver yields lower ODW's for the same reasons as it does for the elliptic Venus capture maneuvers.

Thus, the SCR/G-1 engines are applied most advantageously to Earth departure and (for the Venus capture mission) to Venus departure.

2.3 Mars Mission Vehicles

Mars vehicles were studied for reference missions II and IV. Tab. 2-2 surveys the Mars vehicles by their propulsion modes and mission. Their number indicates the structural class.

Vehicle $8MG_3C$-22/35 uses SCR/G-2 engines with 250k thrust. The operating life of these engines was limited to one hour in accordance with expected probable constraint. The burning time limitations require provisions for a new engine for each maneuver.

The previously shown -22 vehicle has two torus-shaped reserve tanks, one in propulsion module PM-2, the other in PM-3.

Configuration $8MGM_3$-23/11, uses the more advanced SCR/M-1 engines with 50k thrust, 900 sec specific impulse and unrestricted operating life as well as unrestricted number of restarts.[1] The vehicle is laid out for mission II B, like $8MG_3C$-22/35. The large weight difference of some 900,000 lbs or 400 t (cf. Fig. 2-6) is due to higher specific impulse and reduced overall as well as specific engine weight.

1) It is realized that these characteristics remain to be demonstrated. There is reason to expect that they can be attained, especially restart capability and engine life. Inasmuch as these capabilities would make development of this engine worthwhile, they are used here.

Configuration $8MG_3C$-22/40 is designed for mission II F (perihelion braking maneuver, PB). While velocity braking close to the perihelion is more effective than near Earth, (cf. Vol. III), the heavier payload weight which must be maintained in view of the remaining 60 to 80 day mission period tends to eliminate this gain. However, by jettisoning the remaining two extension modules (which contain laboratories, data storage equipment, etc., assumed to be no longer needed, because all information has been transmitted to Earth) and eliminating other weight no longer needed, the payload weight can be reduced to 35, 300 lb. With this reduction in payload weight, the ODW falls off by some 540,000 lb (246 t) below the ODW of $8MG_3C$-22/35.

If the SCR/G-2 (250k; 825 sec) engines for M-2 (Mars capture) and M-3 (Mars departure) are replaced by two SCR/M-1 (50k; 900 sec) engines, everything else remaining constant, the ODW of this configuration ($8MGM_2C$-23/21) falls to $1.52 \cdot 10^6$ lb (691 t), compared to $1.988 \cdot 10^6$ lb (903 t) for $8MGM_3$-23/11, a reduction by 486,000 lb (220 t).

The effectiveness of the perihelion brake is improved decisively, if use is made of the abundant solar energy available at heliocentric distances of 0.4 to 0.55 AU by using a solar radiation collector to heat the hydrogen. In using this approach a high specific impulse can be attained without the weight penalties associated with the use of a heavy nuclear reactor and shield system; and a low-weight propulsion system can be assured without the penalty of heavy oxidizer weight which must be paid when a chemical engine is used. A small amount of thrust (10 lb) is generated at a specific impulse of approximately 700 sec for 16 to 17 days to produce a velocity change of 10,000 ft/sec (3.05 km/sec), requiring a net weight[2] of 22,000 lb (10 t), compared to 44,000 (20 t) for the same velocity change with an O_2/H_2 propulsion system.

With solar propulsion (designated by the letter S) for the perihelion brake (PB) the $8MG_3S$-22 configuration has an ODW of $1.888 \cdot 10^6$ lb (857 t), compared to $2.371 \cdot 10^6$ lb for the same configuration with chemical propulsion ($8MG_3C$-22/40).

The ODW of the $8MGM_2S$-23/32 is reduced to $1.37 \cdot 10^6$ lb (623 t), compared to $1.52 \cdot 10^6$ lb (691 t) for $8MGM_2C$-23/21. This weight is low enough to investigate the use of fast neutron engines also for Earth departure. This vehicle, $8MM_3S$-23/34 has an ODW $1.243 \cdot 10^6$ lb (565 t).

Configuration $8MVG_3C$-28/35 is a mission IV B Mars vehicle. Comparison of the ODW's shows that the use of the Venus gravity field and a small powered maneuver during fly-by reduces the vehicle weight by about $1.2 \cdot 10^6$ lb (547 t), albeit at a penalty of extending the mission period by 160 days.

---

2) Net weight = ignition weight minus payload weight

The low ODW of this configuration encouraged consideration of the use of the 50k thrust SCR/M-1 engines in all propulsion modules. The resulting vehicle 8MVM3C-28/40 can be reduced to a Saturn V compatible 33-ft diameter and still is shorter than the 50-ft diameter 8MVG3C-28/25.

Consideration of the fact that the SCR/M-1 engine models used in this study are considered reusable, an additional configuration, referred to as -30 class, was adopted using a multi-cell torus tank in PM-1, thereby permitting the attachment of two of the four SCR/M-1 engines needed for Earth departure on PM-2. By this parallel staging of PM-1 and PM-2 the weight of two engines is saved. Fig. 3-2 depicts the resulting configuration 8MVM3C-28/40. The ODW of this configuration is reduced slightly below that of 8MVM3C-28/40, but not by a significant amount. Both weigh about $1.1 \cdot 10^6$ lb (500 t).

All preceding configurations represent examples for assembly of the complete vehicle in orbit (OVAM). If the service vehicle accompanies the crew vehicle to Mars and back, serving as back-up crew vehicle in case of an emergency, it looks identical to vehicles shown, except for the front section which would contain a service section rather than the life support section shown. If the mode of the convoy is OVAM (2), the service vehicle accompanies the crew vehicle to the target planet only. The ODW of the service vehicle is 500,000 lb (273 t) in this case. The vehicle is equipped with two stages (PM-1 and PM-2). For the ODW of 600,000 lb, the payload is 131,000 lb (60 t) and the thrust unit for each propulsion module consists of an SCR/G-2 engine (F = 250k; I_{sp} = 825 sec). The thrust of these engines is assumed to be variable, so that the thrust/weight ratio of the service vehicle can be made equal to that of the crew vehicle; a necessary condition, if the two vehicles are to be kept together during a powered maneuver.

Similar conditions exist if the vehicle-assembly mode is IVAM (2) or COVAM (2).

Mars vehicles 8MG3C-22/29 are laid out for IVAM (2). Mars vehicle convoy 8MG3C-22/26 is laid out for COVAM (2). OVAM (2) is different from IVAM (2) only inasmuch as crew vehicle and cargo vehicle are merged into one vehicle in COVAM (2), while the PM-3 carrier is the same. This merging has three advantages: it reduces the spread in ODW and, therefore the thrust control problems associated with the spread in weight within the convoy; and it combines LSS and service module, thereby placing the crew in a preferred position of monitoring and using its destination payload; finally, only two, instead of three, vehicles must be launched successfully out of the Earth's orbit.

in M-3 (contained in both halves) is enough to escape the target planet with one LSS, two R-V's[6], two M-4's and two M-3 engines. However, the tankage volume of each half of the duplex is enlarged to be capable of carrying propellant requirements for target planet escape with one LSS, one R-V, one M-4 and one M-3 engine. Thus damage to one half of the duplex, will allow transfer of required propellant to the remaining half, and jettisoning of damaged tankage with its associated hardware.

The convoy and the multiplex vehicle modes each have a number of characteristics in their favor. Those particularly favoring the convoy are listed in Tab. 2-3. They are not necessarily presented in the order of increasing or decreasing significance.

Arguments in favor of the multiplex vehicle are listed in Tab. 2-3 They are not necessarily listed in the order of increasing or decreasing significance.

The duplex vehicle is composed of a crew and service ship of the convoy type tied together to form one vehicle (cf. Fig. 2-7). The convoy requires at least one crew and one service ship to leave Earth with each carrying a complete PM-3[1] (W_{p3}[2] capable of escaping target planet with one LSS[3], one R-V[4] and one PM-4[5]. In the duplex vehicle the total amount of propellant

---

1) Propulsion Module 3 (for Mars Departure)
2) W_{p3} = useful propellant weight of PM-3
3) LSS = life support section
4) Re-entry vehicle
5) PM-4 = Propulsion Module 4 (for Earth arrival)
6) R-V = Return Vehicle (This designation has been replaced by EEM = Earth Entry Module)

CONVOY VS. MULTIPLEX VEHICLE
Arguments in Favor of Convoy

1. Low over-all vulnerability.
2. High Insurance that backup ship will be available to crew in emergency.
3. In case of loss of one vehicle, problems of vehicle separation are avoided.
4. Flexibility in orbital departure weight distribution.
5. No inherent limitations in total orbital departure weight.
6. Freedom in selecting different types of main propulsion systems.
7. No weight penalty for connecting structure between vehicles.
8. Simplification of orb. operations by avoiding coupling two large systems
9. Higher probability of successful departure.
10. Procurement need not consider potential failure in final coupling process of a duplex vehicle.

MULTIPLEX VEHICLE VS. CONVOY
Arguments in Favor of Multiplex Vehicle

1. Simplified engine start control and flight control.
2. Crew module transfer from one ship to another in case of emergency avoided.
3. Good accessibility of auxiliary vehicles.
4. Because of 3., need for taxi capsules may be eliminated.
5. Lighter weight of meteoroid and heat shield for tanks, since multiplex tanks partially shield each other.
6. Because of 5., easier access to tanks and easier propellant transfer if needed.
7. Transfer of individual tanks and associated demating and mating process are avoided.
8. Problems of intra-convoy vehicle maneuvering avoided.
9. Because of 8., increased safety of crew against accidental irradiation by reactors.
10. Because of 8., lower propulsion weight required for attitude and path control.

Tab. 2-3 COMPARISON OF CONVOY AND MULTIPLEX TRANSPORTATION MODE

3. VEHICLE DESIGN

The conceptual design of interplanetary vehicles must include a very large number of parameters ranging from scientific estimates of the interplanetary environment, mission plan and velocity, to space technology and human factors.

After the basic configurations, described in Sect. 2, were established (Fig. 3-1), some pertinent structural loadings and structure arrangements were varied while new mission data was introduced to determine their effects upon vehicle weight. Thus a Mars mission employing a perihelion brake or a Venus powered fly-by maneuver as a braking scheme to reduce Earth entry velocities yielded $8MM_3S$-23 and $8MVM_3C$-30 vehicles (refer to Fig. 3-2). Along with vehicle conceptual designs, a section describing methods to determine preliminary estimates of advanced vehicle structures is included.

3.1 Principal Assumptions

Interplanetary vehicle structural investigations, including Earth and/or interplanetary space requirements, were based upon:

(a) LH_2 and LO_2 insulation
(b) Free standing fail safe structure
(c) Dynamic and acoustic vibrations
(d) Launch accelerations
(e) Aerodynamic heating and loading
(f) Weight
(g) Manned safety (1.4 factor based on ultimate strength)
(h) Meteoroid and thermal protection
(i) Orbital and interplanetary rendezvous and assembly
(j) Engine burning time
(k) No advancement in material beyond 5AL-2.5Sn titanium alloy

3.2 Generated Data

New data generated during the study is briefly described and discussed in the following paragraphs.

3.2.1 Orbital Tanker

Studies performed determining conceptual interplanetary vehicle designs seem to suggest the desirability of orbital fueling. A fueling method and a preliminary orbital tanker design are presented. Fig. 3-3 illustrates an orbital tanker, containing 2.2×10^5 pounds of liquid hydrogen, is 33 ft in diameter and is assumed to be launched by Saturn V. The tanker wall can be

a sandwich or semi-monocoque, covered with nonjettisonable insulation, structure. One end of this vehicle contains an extendable fuel line and shock absorbing struts, the latter to permit gentle contact with the interplanetary vehicle.

At the beginning of the fuel transfer process, the propellant is caused to rotate inside the tanker, creating a mild gravity field which keeps the propellant against the tank wall. A velocity of 3 ft/sec in the outer periphery of the liquid and the gravity field created combine to keep a positive head at the pump inlet. The propellant is caused to rotate by means of liquid jet pumping. This jet flow comes from a tap off the high pressure side of the pump. A small attitude control must be active on either the tanker or the tanked vehicle to prevent roll of the tanker. After the propellant velocity has reached an equilibrium, the friction on the tanker wall tends to balance the jet pump reaction, and very little roll tendency exists.

3.2.2 Reserve Tank

A method to increase vehicle mission completion assurance is to take along an empty reserve tank or tanks. Fig. 3-4 illustrates a scheme to carry the reserve tank in the form of a torus that is centrally located and well protected by the satellite propellant tanks. Propellant lines from the tank would be manifolded into one line which enters the torus tank through a single pump. Artificial gravity (which is provided by spinning the vehicle) would provide NPSH for the pump. The damaged tank could be repaired and the propellant could then be pumped back in, again leaving the central tank as a reserve.

3.2.3 Preliminary Estimates of Advanced Vehicle Structures

Advanced vehicle designs will probably include semi-monocoque waffle, corrugated, and sandwich structures which will require testing and detailed analysis. However, for conceptual designs an area estimate of structural weight is adequate. The design aids in this reports should be used as first cut estimates to determine initial vehicle size. Anyone using these aids should be familiar with the criteria and application in order to obtain reasonable area estimates.

This section describes general theory of columns, frames and rings, pressurized structure, shear criteria, and sandwich structure as required for preliminary weight estimates. Curves and aids presented are:

(a) Buckling allowables vs. r/t and b/t for 5AL-2.5Sn titanium alloy and 2014-T6 aluminum alloy.

(b) *Loading vs. column and frame areas

*For 5AL-2.5Sn titanium alloy only

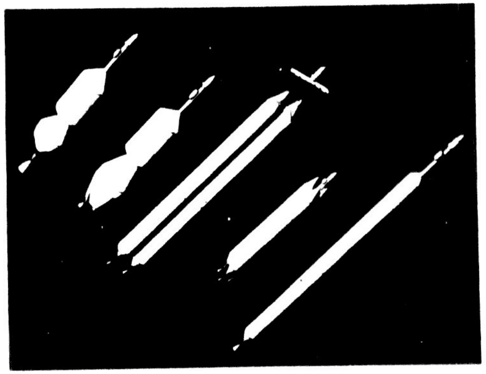


Fig. 3-1

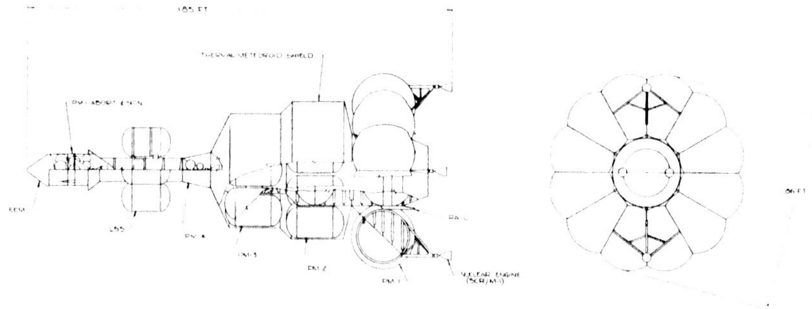


Fig. 3-2 MARS I V WITH VENUS PFB 8 MVM3C-30

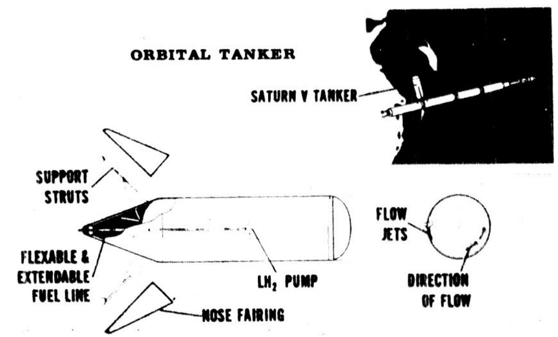


Fig. 3-3

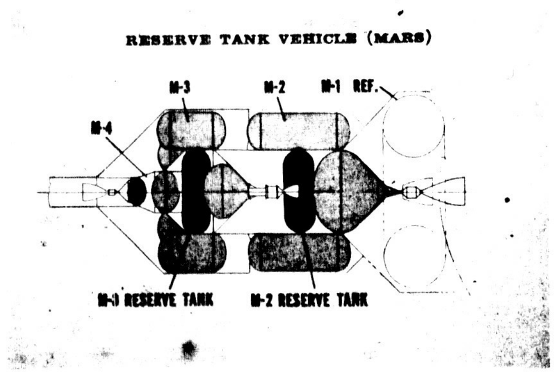


Fig. 3-4

(c) Moment of inertia vs. frame area

(d) *Shear vs. thickness for τ / τ_{cr} = 1.0, 2.0, and 3.0.

(e) * τ/τ_{cr} vs. deflection and Δt.

(f) *Tank pressure vs. thickness for various values of tank radius.

(g) Buckling allowables for sandwich structure

*For 5AL-2.5Sn titanium alloy only

3.2.4 Propellant Tanks

The structure of propellant tanks requiring a free standing capability usually consists of a semi-monocoque, waffle or sandwich design. The skin or wall thickness of the tank is usually determined by tank pressure and the stiffness required based on non-pressure loads. Because of the relatively low free-standing ground loads, a semi-monocoque design was selected. However, the 10 psi tank pressure requirements were lower than the non-pressure loads, and the latter determined both the skin thickness and the stiffness required. The design criteria (shear buckling) used caused the skin between stiffeners to resist shear loads, that resulted from the ground requirements, without buckling or wrinkling. A more liberal design criteria, tension field, absorbs loads above buckling in the form of diagonal tension, causing gentle wrinkles in the skin. The amount of weight saved with the latter criteria indicates that more testing in this area is desirable. An example of gentle wrinkles occuring in a structure exposed to a liquid hydrogen atmosphere is the steel, buttwelded, insulation bulkhead of the Centaur vehicle, designed and built at General Dynamics/Astronautics. During the propellant loading cycle, gentle wrinkles form in the skin and across chem-milled, buttwelded joints, then disappear during pressurization and load removal.

Pressure increase vs. tank weight is also presented for interplanetary vehicles launched on Saturn V or Post-Saturn. A pressure increase from 10 to 30 psi increases tank weight approximately 21 to 50% depending upon configuration.

3.3 Principal Conclusions

(a) After choosing an interplanetary mission the two main factors affecting the spacecraft configuration are engines and ELV.

(b) Launching the entire propulsion structure of an interplanetary vehicle with one Post-Saturn and supplying the propellant needed with an orbital tanker is more desirable than launching the vehicle in two fully fueled modules and mating in Earth orbit.

(c) A chemical LH_2/LO_2 module for PM-4 is recommended for vehicles employing graphite core nuclear engines.

(d) A clustered propellant tank arrangement with an empty reserve tank or tanks centrally located is recommended for vehicles with diameters greater than 50 ft.

(e) The properties of a metal core reactor nuclear engine (suggested by rocket engine contractors) permit reduction in vehicle weights when compared to vehicles using graphite core reactor engines.

(f) Permitting aerodynamic launch loads to be absorbed by an external fairing rather than by the interplanetary vehicle results in lighter structural weights. (Especially in cases where the Post-Saturn ELV launches the inter-planetary propulsion and life support system structure in one launch.) The launch fairing also serves as the thermal meteoroid shield for PM-1 (excluding that insulation required for ground conditions) and reduces the thermal meteoroid shield of PM-3 and PM-2 while the vehicle is in Earth orbit.

(g) Location of PM-4 within PM-3 thermal meteoroid shield reduces puncture probability, amount of shield for the Earth approach module, and aids in placing the return coast CG at a greater distance aft of the LSS.

(h) Jettison thermal-meteoroid structure prior to the associated maneuver. The clustered tanks plus the empty reserve tank arrangement discussed in Sect. 3.2.2 affords acceptance of one or two punctures and thus reduces the shield weight by a large amount (refer to Sect. 4).

4. FUEL CONSERVATION AND METEOROID PROTECTION

The term" fuel conservation system" (FCS) which will be used throughout this section includes all equipment and material (and associated structure) used for thermal and meteoroid protection of the cryogenic fuel and oxidizer. The missions considered are Mission I Venus Elliptic Capture 1975 and Mission III Mars Circular Capture 1975. The two primary objectives were to determine the most effective form of thermal and meteoroid protection and to obtain meaningful weight data for input to the overall vehicle weight analysis.

4.1 Thermal Control Devices

Three primary means of preserving the cryogenic propellants were considered: superinsulation, refrigeration equipment, and shadow shield. Increase in tank skin gage to accommodate propellant vapor pressure rise was also considered in some of the "trade-off" analyses.

For the purpose of analysis, a superinsulation density of 5 lb/cu. ft. and k of 2.5×10^{-5} BTU/HR-FT^2-^{o}R/FT was used for all liquid hydrogen application with outer wall temperature less than 560^{o}R. For applications involving high wall temperatures, density of 6 lb/cu. ft. was used together with appropriate variation in k with temperature.

The data on a liquefaction system described in the proposal by Malaker and Daunt of Cross-Malaker Laboratories, Mountainside, New Jersey was used in the analysis. Preliminary weight data and electrical power requirements are given in Fig. 4-1.

A preliminary analysis of the shadow shield was conducted to determine its performance when used in conjunction with superinsulation. Although a number of shadow shield configurations were evaluated, the design shown in Fig. 4-2 gave the best results. The V shaped foil was used to reduce the view factor between the V foil and the second foil. Performance of this shadow shield when used with superinsulation is given in Fig. 4-3 together with those of superinsulation alone.

4.2 Meteoroid Protection

In evaluating the meteoroid protection requirements for the Mars and Venus vehicles, Fred Whipple's 1963A near-Earth and deep space meteoroid flux data was used. For the hypervelocity penetration process, Bjork's equation was used. The probability of puncture was assumed to follow Poisson's distribution.

The combination of fibrous material and thin foils characteristic of superinsulations lends itself to effective absorption of meteoroid impact energy. It is of advantage, then, to incorporate the superinsulation into the meteoroid shield design and utilize its dual capabilities. Although a high percentage of superinsulation is desirable from the thermal standpoint, a combination of 70% aluminum skins and 30% superinsulation (by weight) was selected to insure an effective meteoroid barrier. This design was assumed to be five times more effective than a single sheet of the same weight per unit area. Fiberglass honeycomb core was added to the skins and superinsulation to form a structurally stable panel for handling and jettisoning. The honeycomb core was assumed to make negligible contribution to meteoroid protection.

The following criteria were used to determine the required meteoroid protection:

(1) 99% probability of less than one penetration for the single-tank and chemical stages.

(2) 99% probability of less than four penetrations for the clustered-tanks stages. (The clustered-tanks stages are equipped with a reserve tank adequate for two satellite tanks. A loss of a third satellite tank results in only a 10% propellant loss.)

4.3 Target Planet Capture and Departure Stages PM-2 and PM-3 Fuel Conservation System Analysis

The solar heat flux history for the PM-2 and PM-3 stages are shown in curves (1) and (2) of Fig. 4-4. They are both characterized by long duration of moderate heating. For such an environment, a combination of liquefaction system and superinsulation appears most attractive. Therefore, this combination was selected for analysis as the primary form of thermal protection system. A superinsulation system and a combination of superinsulation, liquefaction system, and shadow shield were also evaluated for comparison purposes.

Two main criteria govern the "tank side" fuel conservation system: the meteoroid protection requirement and the optimum combination of superinsulation and liquefaction equipment. The evaluation of these criteria was accomplished by determining the weight of superinsulation plus meteoroid "bumper" skins for the following conditions:

(a) Meteoroid protection requirements using a 70/30 combination of skins and superinsulation.

(b) Optimum combination of superinsulation and liquefaction system, and using the minimum skins required for structural purposes.

(c) Optimum combination of superinsulation and liquefaction system for a 70/30 combination (but disregarding the meteoroid protection requirements).

Result of the analysis for the single-tank and chemical stages show that the meteoroid shielding requirements far surpass the other designs and dictates the "tank side" thermal and meteoroid protection system. On the other hand, the results for the clustered-tanks stages are quite different in that the design is dictated by the meteoroid criteria for only the higher propellant ranges. The remainder is designed by optimum thermal protection requirements. This large reduction in meteoroid shielding requirements is attributed to the relatively high number of allowable meteoroid penetrations incorporated into the stage design.

The fuel conservation system (FCS) weight, which includes the "tank side" thermo-meteoroid shield as well as other insulations and liquefaction systems, are given in Fig. 4-5.

4.3.1 Comparison of Liquefaction-Plus-Superinsulation System with Other Fuel Conservation Systems

Past studies have shown the attractiveness of the Liquefaction-Plus-Superinsulation system over other systems for long term missions with moderate solar thermal flux. But it would be of interest to determine the magnitude of the weight advantage offered by this system.

The two systems selected for comparison with the above Liquefaction-Plus-Superinsulation system are: (1) Superinsulation system, and (2) Superinsulation-plus-liquefaction-plus-shadow shield system (S-L-S system).

For the purpose of analysis, a Mars mission PM-3 stage with a 60 ft diameter clustered-tanks configuration was used. The result is shown in Fig. 4-6 where the systems equipped with liquefaction equipment shows considerable advantage over the Superinsulation system.

4.3.2 Effect of Meteoroid Hazard on the Fuel Conservation System Weight

The discussions in the previous sections indicate the major contribution made by the meteoroid protection requirements in determining the

fuel conservation system weight. Yet the meteoroid hazard is still a topic of constant controversy and data regarding meteoroids, especially their flux density, vary by more than an order of magnitude. An attempt was made to evaluate the effect of variation in the meteoroid flux density on the fuel conservation systems weight. Both a single-tank stage and a clustered-tank stage were analyzed and the results are shown in Fig. 4-7. Since the meteoroid protection requirements were much greater than other criteria for the single-tank stage (using Whipple's flux), a marked decrease in FCS weight results with reduced meteoroid flux. A substantial increase in FCS weight is noted for increased meteoroid flux for both the single-tank stage and the clustered-tanks stage. The straight line portion of the curves is determined by the optimum thermal design and structural requirements.

An additional study was made to determine the degree of FCS weight reduction that can be realized by increasing the allowable number of meteoroid penetrations without stage failure. This was accomplished by designing the meteoroid shield for a 99% probability of less than n penetrations during a single mission. The results are shown in Fig. 4-8 for a Mars mission PM-3 with a 50 ft diameter single-tank configuration. A major reduction in FCS weight is obtained by increasing the allowable number of penetrations from less than one to less than two. Further weight reductions (up to 40%) can be realized with additional increase in allowable penetrations but the lower limit is constrained by the thermal and structural requirements.

4.4 Earth Retro-thrust Stage - Chemical PM-4 - Mission III - Mars 1975

The Mars to Earth transfer path for Mission III is characterized by a close perihelion passage to minimize the overall mission velocity. However, this subjects the vehicle to a gantlet of intensive solar irradiation during a portion of the flight. The maximum thermal flux history for this mission is shown in Fig. 4-4.

Particular consideration was given to the preservation of liquid hydrogen aboard the chemical Earth retro-thrust stage, the PM-4. Because of the large amplitude of the solar heat influx, no one combination of fuel conservation systems exhibits obvious advantage over others. Therefore, analyses of four different fuel conservation systems were performed. These FCS are as follows:

System I	Superinsulation system
System II	Superinsulation-Plus-Shadow Shield system
System III	Superinsulation-Plus-Liquefaction system
System IV	Superinsulation-Plus-Liquefaction-Plus Shadow Shield system

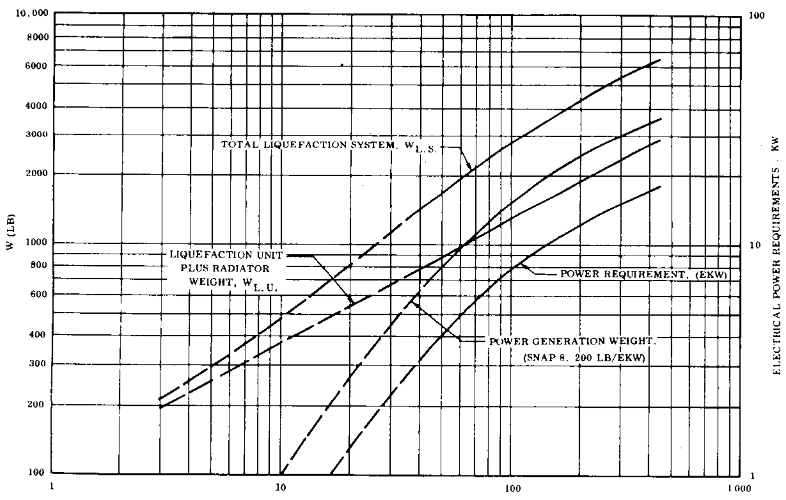


Fig. 4-1 LIQUEFACTION SYSTEM WEIGHT

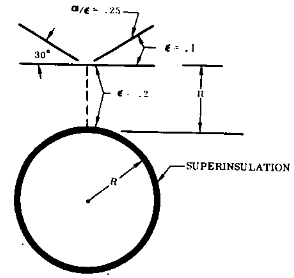


Fig. 4-2
SHADOW SHIELD CONFIGURATION

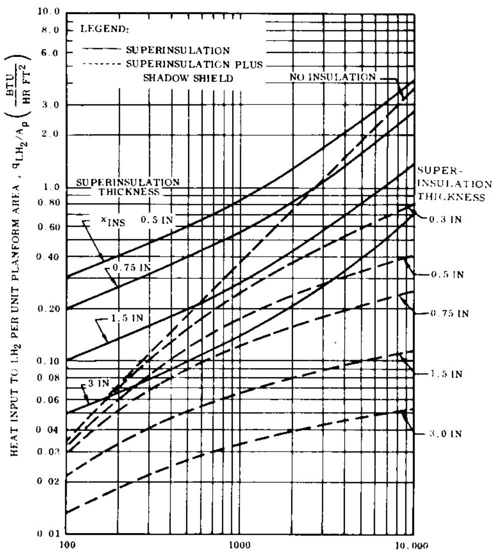


Fig. 4-3 COMPARISON OF SHADOW SHIELD PLUS SUPERINSULATION AND SUPERINSULATION ALONE

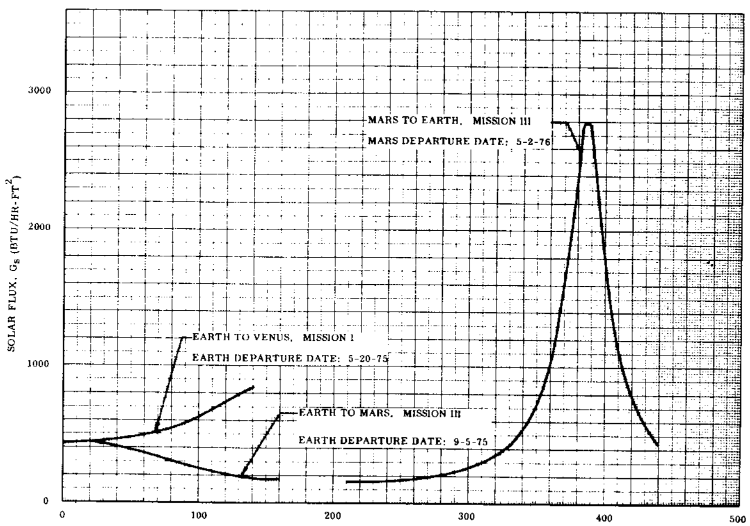


Fig. 4-4 SOLAR THERMAL FLUX HISTORY FOR MARS AND VENUS MISSIONS

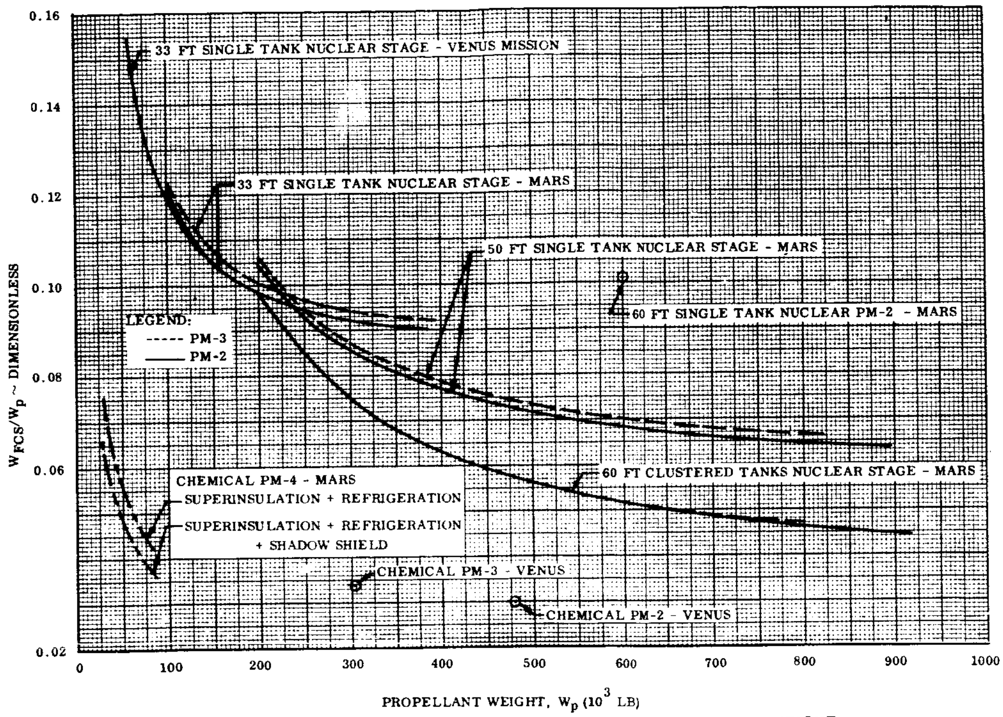


Fig. 4-5 FUEL CONSERVATION SYSTEM WEIGHT PER UNIT PROPELLANT WEIGHT

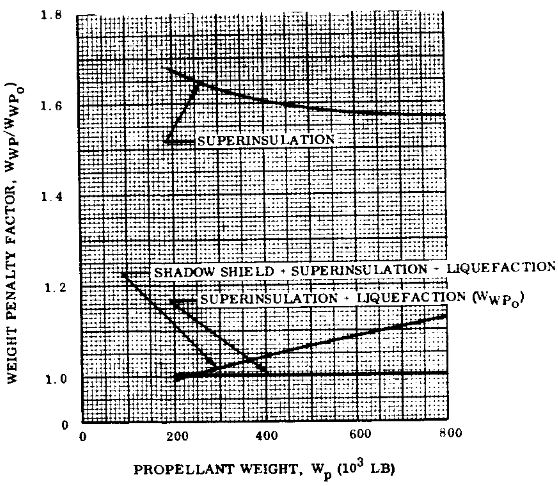


Fig. 4-6 COMPARISON OF LIQUEFACTION PLUS SUPERINSULATION SYSTEM WITH OTHER SYSTEMS

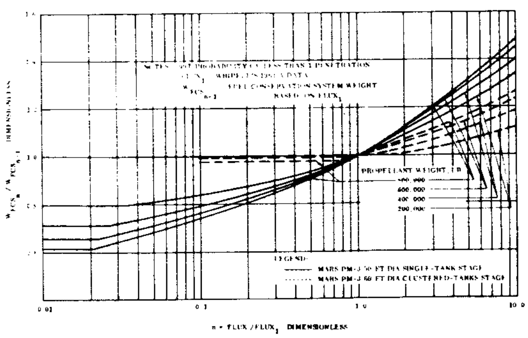


Fig. 4-7 EFFECTS OF METEOROID FLUX VARIATIONS ON FUEL CONSERVATION SYSTEM WEIGHT MARS PM-3 - 50 FT SINGLE TANK STAGE

Systems III and IV showed appreciable advantage over I and II, and therefore, were considered for the overall analysis of the PM-4 stage. For the forward area insulation, a "trade-off" analysis was made between insulation thickness and liquefaction requirements for a constant heat source of 530°R from the entry capsule. Analysis of the LO_2 tank showed that one inch thickness of superinsulation is adequate to preserve the liquid oxygen during the mission. For this case, a tank design pressure of 24 psia is sufficient for a non-vented design.

The total fuel conservation system weight for the PM-4 chemical stage is presented in Fig. 4-5. The weight includes that of the superinsulation, insulation jettison structure, liquefaction system, and (for System IV) shadow shield, but does not include the tank weight. The values compare favorably with other chemical stages in spite of the high heating environment, the advantage being the low meteoroid shielding requirements.

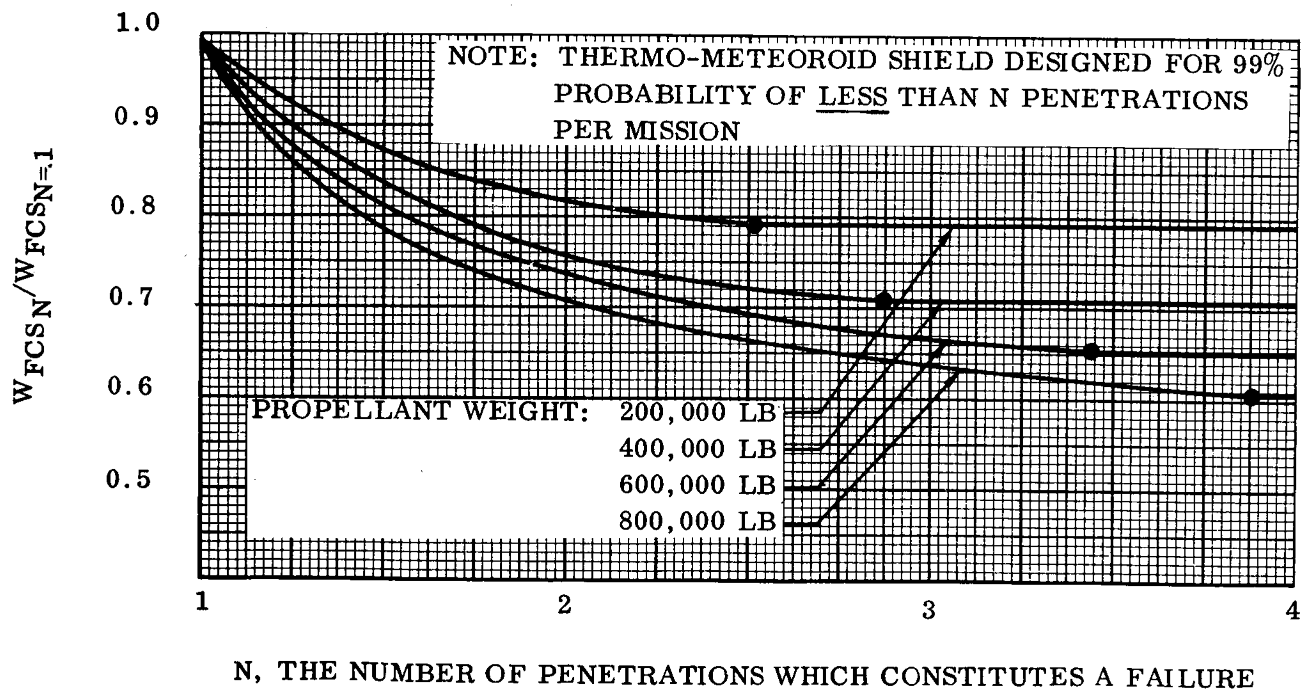


Fig. 4-8 EFFECTS OF ALLOWABLE NUMBER OF METEOROID PENETRATIONS

5. LIFE SUPPORT SECTION (LSS)

5.1 LSS Configuration and Design Concepts

There exists a large number of alternatives in LSS design concepts, because the LSS represents an interface between technological and biological requirements. The number of influential parameters, therefore, exceeds that usually controlling the design of systems which are subject to technological considerations only. A survey of the configurational concepts studied is shown in Fig. 5-1.

LSS designs can be distinguished on the following premises:

- Operation at zero gravity
- Operation at artificial gravity
- Operation under both conditions (g/zero-g conditions)

The third of the above alternatives has been adopted as frame of reference in this study, since it requires somewhat more weight and is slightly more involved than the assumption of zero-g flight.

Two alternatives exist for generating artificial g-conditions and the LSS has to be designed accordingly:

- Tumbling vehicle LSS
- Spinning vehicle LSS

For vehicles of great length, such as most nuclear-powered interplanetary vehicles, it was found preferable to adopt the tumbling vehicle LSS design. In this manner an inherent characteristic of the vehicle is utilized and the LSS can be compact, rather than consisting of several more or less isolated sections. However, in cases where the I/V configuration becomes small or stubby in the course of the mission, design for a spinning vehicle LSS becomes comparatively more advantageous.

The tumbling vehicle LSS configurations can be divided into two groups:

- Radial LSS
- Horizontal (or Lateral) LSS

The radial LSS conforms better to shadow shielding requirements. The horizontal LSS is potentially superior from the standpoint of several human engineering aspects. Therefore, both configurations were given considerable attention.

The spinning vehicle LSS configurations fall into three categories:

- Torus shaped LSS
- Parallel-bar shaped LSS
- Dumbbell LSS configuration

The torus shape is inherently suitable only for larger crew sizes than those considered here (3 to 12 persons). The parallel-bar configuration is a modification of the axially expanded dumbbell configuration proposed earlier by Kramer and Byers[1]. It does not contain dumbbells, but cylinders of the type shown in L-44 (horizontal LSS) with the same advantages from the human engineering point of view (cf. below). Radiation shelter, command module and Earth entry module (EEM) are on the foreward end of the spine in a zero-g region, separated from the spinning vehicle by a counter-spin mechanism which eliminates the spinning sensation for the crew working at the controls. The dumbbell design lends itself potentially best for providing a large spin radius. Two, three or four spokes are used, at the ends of which are located one-, two- or three-story capsules for living and working. The spokes can be folded, as indicated in Fig. 5-1, to facilitate the transport into orbit and to move the capsules into the shadow cone of the nuclear shield during powered maneuvers. Radiation shelter, command module and EEM are located at the forward end, atop a counter-spin mechanism, as in the parallel-bar configuration. In the dumbbell configuration, crew members in the individual capsules are subject to a comparatively greater degree of isolation than in the parallel-bar configuration.

An important part of the LSS is the radiation shelter. Among the incidentals available in the LSS, water is the best shielding material, from the standpoint of quality as well as quantity. However, even though LSS designs up to 1000 persons were investigated (as part of another study contract),

---

1) Kramer, S. B. and Byers, R. A., A Modular Concept for a Multi-Manned Space Station, Proceedings of the Manned Space Stations Symposium, Inst. of the Aeronautical Sciences, New York, 1960.

no case was found in which a sufficient quantity of water could be reasonably assumed to be available to provide the necessary attentuation of solar flare radiation (to about 1 rad average dose per day). Thus, while water was utilized for shielding, it had to be complemented by another substance which, in turn, could be:

- solid shield (carbon, borated polyethylene)
- liquid shield (LH_2, CH_4, Monomethyl Hydrazine (MMH)

The use of LH_2 as shield material was studied first and compared with carbon shielded LSS configurations. LH_2 is a very efficient absorber on the weight basis, but not on the volume basis. Its use, therefore, pays off only if it has to be carried along anyway as propellant for the Earth return retro-maneuver. Since this constraint makes this design mission sensitive, i.e., not applicable if no, or only a small Earth retro-maneuver is involved, it was abandoned in favor of LSS designs using solid or heavy liquid shielding material. Carbon was replaced by a superior combination of boron-containing polyethylene. A comparison of denser liquids (cf. Vol. V) showed MMH and RP-1 to be particularly effective. Since MMH is the better of the two and requires little more weight than polyethylene; and since for the chemical auxiliary propulsion on board the propellant combination OF_2/MMH was adopted for independent reasons, MMH became the preferred liquid shield material. In the overall design, hardly a difference is noticeable between a radiation shelter design based on polyethylene or on MMH. The use of a liquid has several advantages. If it is a fuel, then, together with an oxidizer it can be burned during the Earth retro-maneuver, thereby serving two purposes, a principle which almost always yields weight savings; or the liquid can be heated and ejected in a solar-thermal propulsion system; or, if the vehicle weight must be lightened, even at the penalty of an increase in average radiation dose for the rest of the mission, a liquid shield can be jettisoned conveniently in any proportion to the original quantity. As to the radiation shield requirements, cf. Sect. 5.3.

As far as the overall design philosophy is concerned, two alternatives are available:

- Integrated design
- Modular design

The modular design, in spite of a comparatively small weight penalty, is preferable to the integrated design in vehicles of limited-to-marginal mission performance, such as the vehicles dealt with in this study. In the modular design, parts not vital to the survival of the crew can be jettisoned in case of an emergency. For these reasons, the modular design philosophy was adopted for the LSS configurations considered in this study.

5.2 Evolution of Life Support Section Design During the Study

A brief review of the principal configurations for tumbling vehicles, considered primarily in this study, follows. All LSS shown are laid out for a crew of eight persons.

Initial investigations were concerned with optimization of the shielding system for a command-module plus radiation shelter combination (L-27). But since the nuclear engines likely to be available in the 1973/77 period probably have specific impulses below 850 sec, chemical (O_2/H_2) systems for M-4 yield lower vehicle weights for most of the Earth capture maneuvers. In this case, not enough H_2 is available to justify the basically heavier design for a shelter region in the fuel tank.

For the life support system (LSS) with the "dry" shielding provisions, polyethylene with boron is among the lowest weighing shielding materials. Two basic concepts were investigated. One is the integrated design; the other the modular design. L-36 depicts the best design among the quasi-toroidal configurations investigated. L-46 is a system serving 25 persons (L-46-25). It is an example for the toroidal design which is preferable for larger passenger numbers and which has been investigated for up to 1000 passengers.

In spite of favorable features, the integrated configurations were not selected, since they sacrifice flexibility inherent in the modular versions, which permit jettisoning modules in emergencies or addition of modules for larger crew sizes without significant change of the modules already in existance. These are considered to be operational and economic advantages of overriding importance for initial missions with vehicles of comparatively limited performance and in view of the many developmental uncertainties facing development toward an initial mission of such magnitude as a planetary flight.

The module cluster is best suited for longitudinal (or radial) arrangement of the life support system, i.e., in extension of the spine. The best of these configurations is L-42, shown in Fig. 5-1. In the modular design, a distinction is made between the (vital) "primary modules" and the (not vital)" extension modules which can be jettisoned if necessary. In L-42 the primary modules are located in the forward end of the spine.

The cylindrical arrangement is best suited for horizontal configurations (laterally oriented to the spine). The best of these configurations is the L-44 shown in Fig. 5-1. Here the outer sections on both sides of the cylinder are extension modules.

Comparison shows that L-42 is better shielded from the engines by the propellant tanks. The extremities of L-44 project beyond this shielding and though fiberglass structure is used, some metallic equipment is present. However, in most other respects L-44 appears the superior configuration. It

is lighter, simpler, and provides a hedge against the somewhat ambiguous area of man's tolerance to Coriolis acceleration. To gain these advantages, some added shielding may be a reasonable trade-off.

The investigation of spinning-vehicle life support systems is less complete than the one for tumbling vehicles. Among the configurations shown in Fig. 5-1, the modular design is again more advantageous for the vehicles and crew sizes under consideration in this study. Of the two modular configuration concepts, the parallel-bar concept appears preferable compared to the dumbbell configuration.

5.3 Radiation Shield Requirements

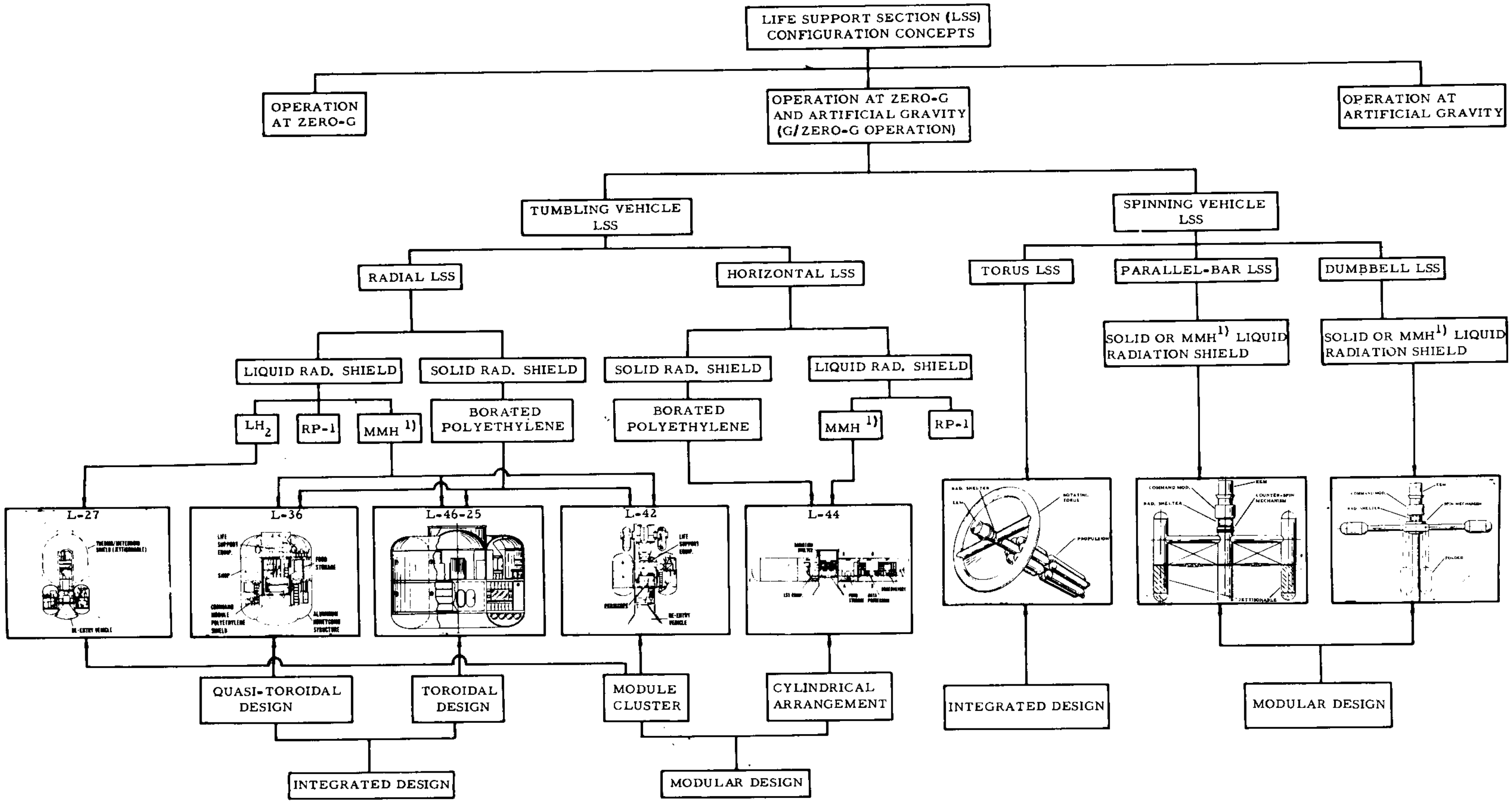
Another area requiring considerable attention for long missions is that of the solar radiation dose and associated radiation shelter requirements.

In the recently completed EMPIRE (Early Manned Planetary-Interplanetary Round-trip Expeditions) study, conducted by GD/Astronautics for NASA, an analog computer program was developed for calculating mean solar flare fluxes encountered over interplanetary trajectories. The purpose was to establish working methods for predicting mean annual solar flare fluxes, particle encounter probability, relative particle intensities from flares of different classes, and probable fluxes filtering through crew shielding.

The changing solar distance was approximately accounted for by assuming an inverse square law spatial dependence. Despite the paucity and uncertainties in solar data, the program yields probable mean radiation dose rates for interplanetary missions and allows comparisons to be made between missions.

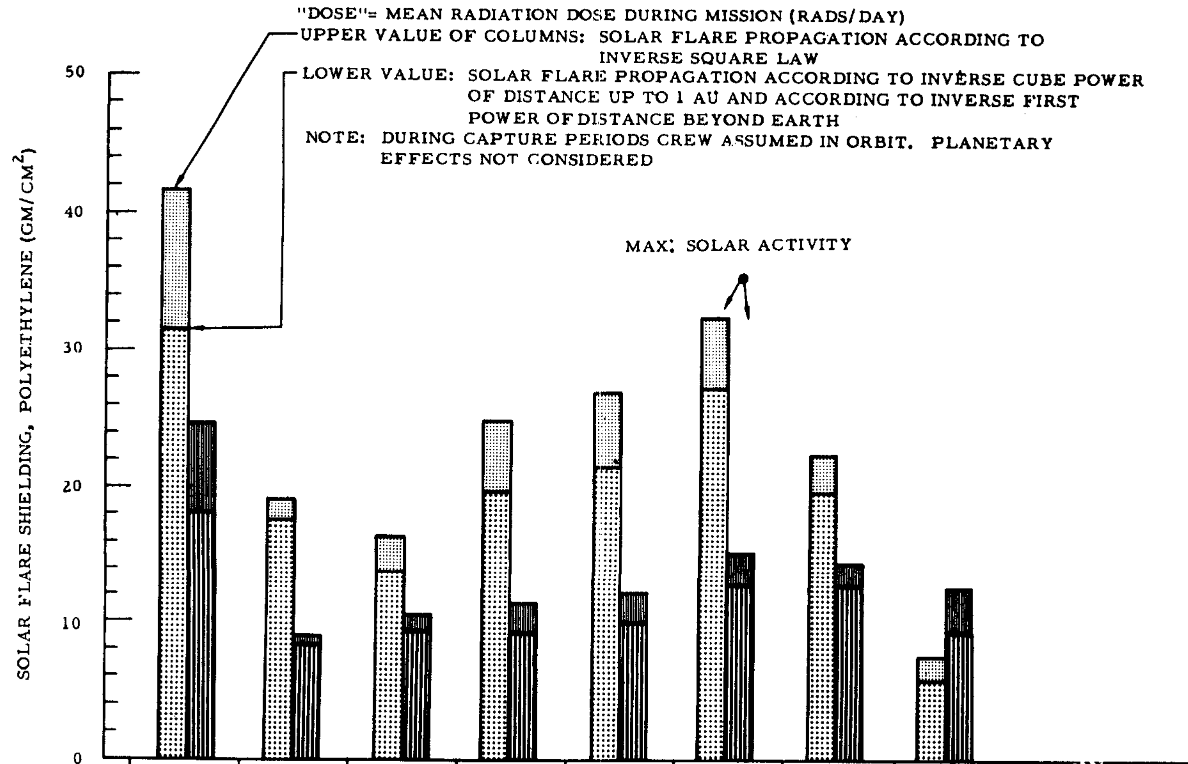
Because of its utility, and its potential for increased scope on interplanetary missions, the work was expanded and modified for use on the 7090 computer.

The program has been applied to a wide variety of missions. Fig. 5-2 shows a number of examples, directed primarily at missions to Venus and Mars for typical radiation doses as specified in the figure. The effects of solar activity level, heliocentric distances encountered during the mission and mission period are clearly reflected in the shielding requirement. A Mercury mission and a Jupiter mission are shown for purposes of comparison. The shielding requirements obtained from computations such as these form the basis for the radiation shelter weight determination.



1) MMH = MONOMETHYL HYDRAZINE

Fig. 5-1 SURVEY OF LIFE SUPPORT SECTION CONFIGURATION AND DESIGN CONCEPTS



DOSE (rads/d)	.25 1.0	.25 1.0	.25 1.0	.25 1.0	.25 1.0	.25 1.0	.25 .50	.25 .50
DESTINATION	Me	Ve	Ma	Ma	Ma	Ma	Ve,Ma	Ju
MISSION PERIOD (d)	80+110+190	140+20+220	360+220 +370	160+50+230	180+40+220	200+40+210	120+246 +234+141 +269	600+70+500
EPOCH: DEP. EA.	12-13-79	10-7-75	9-15-75	10-15-75	11-10-77	12-5-79	6-19-75	5-20-86
EPOCH: AR. EA.	12-27-80	6-23-76	4-22-78	12-18-76	1-24-79	2-27-81	3-19-78	8-2-89
PERIHELION (AU)	0.3447	0.7209	1.0	0.3677	0.409	0.4477	0.719	0.3795

Fig. 5-2 VARIATION OF SOLAR FLARE POLYETHYLENE SHIELDING WITH MISSIONS TO VARIOUS PLANETS AT VARIOUS EPOCHS

6. INTERPLANETARY VEHICLE WEIGHT ANALYSIS

6.1 Approach to Individual Weight Determination

Fig. 6-1 presents the standard approach applied to the weight determination throughout the study. In addition, the principal velocity terms are shown.

The vehicle weight is designated either as ignition weight, W_A and burnout weight W_B; or as heliocentric coast weight W_C or satellite orbit weight W_S.
$W_A - W_B$ = expended propellant W_p
$W_B = W_b + W_\lambda$ where W_b is the wet inert weight, i.e., the weight of the emptied propulsion system from tanks to engines plus residual fluids and W_λ is generally the gross payload. The gross payload of the last stage consists of the sum of operational payload, intransit payload and destination payload.

The vehicle always eliminates weight immediately prior to a principal powered maneuver (i.e., planetary escape or capture or a major heliocentric path change). This is the jettisoned weight W_j. The numbers attached to W_j and to the other weights refer to the number of the powered maneuver. For a single-planet mission the principal maneuvers are numbered consecutively, beginning with 1 = Earth departure through 4 = Earth arrival and 5 the terminal weight, essentially the weight of entry vehicle and crew. Thus, W_{C4} is the weight of the interplanetary vehicle (I/V) at termination of heliocentric coast and W_{j4} is the weight eliminated just prior to the Earth approach retro-maneuver or atmospheric brake maneuver.

The basic structure (i.e., tanks, connections and also thrust frame) of the I/V is very light, resulting in a low ratio of propulsion module wet inert weight to propellant weight, $W_b/W_p = (1 - x)/x$, where x is the mass fraction $(W_p/(W_b + W_p)$. On the other hand, the tanks and all other sensitive parts of the vehicle are surrounded with a strong combination of heat shield and meteoroid protection shield. Immediately preceding a major maneuver, the shield of the tanks to be emptied during this maneuver, will be jettisoned, thereby restoring the favorable mass fraction when it counts.

The abort and spin requirements are satisfied by a common system. If an abort condition does not arise during maneuver one, part of the system will be jettisoned and the balance used to satisfy the spin requirements during outbound and return coast. This is a chemical system utilizing oxygen difloride (OF_2) and mono-methyl hydrazine (MMH) with a specific impulse of 405 sec. The abort system uses a 75,000 lb thrust, pressure fed engine with a 40:1 expansion ratio. The abort system is capable of 10,000 ft/sec velocity. During

the outbound coast there does not appear to be any apparent problems with respect to minimum spin radius of 40 ft. However, during return coast the smaller vehicles will require spine lengths in the order of 100 to 150 ft if a minimum spin radius of 40 ft is to be maintained. This area will require further study in the future (cf. also spinning vehicle life support systems, Sect. 5).

The crew vehicle payload is defined as the operational payload and the service vehicle payload is defined as the destination payload. The operational payload is divided into two groups: (1) the mission payload and, (2) the flight equipment.

The mission payload consists of those items that are necessitated by the presence of a crew (Earth entry support, reentry capsule, crew, communication system, life support section, and scientific payload) and the weight will vary from 105,000 lbs to 116,000 lbs. These weights correspond to Mars Mission II or III with an eight man crew and the variation is due to the type Earth entry considered (hyperbolic rendezvous or 70,000 ft/sec, respectively). If the destination payload and operational payloads are to be of equivalent weight at PM-1 ignition, the weight of the landers, excursion modules, satellites, probes, etc. should equal the weight of the mission payload. The flight equipment weight is equivalent (18,600 lbs) for both the operational and destination payloads.

The radial (L-42) life support section is used with the weight analysis rather than the horizontal (L-44) section. This is acceptable since they are of equivalent weight (87,600 lbs for L-42 and 87,300 lbs for L-44). The dry LSS/inside volume varies from 3.5 lbs/ft^3 to 5.0 lbs/ft^3 and the dry LSS/floor area is approximately 37 lbs/ft^2. The ecological weight/man is 4,740 lbs and the payload density will vary between 2.0 and 2.5 lbs/ft^3.

The eight man Mars OVAM vehicles based on Mission III will vary from 1.1 million to 3.0 million lbs depending upon the vehicle configuration and type entry condition considered. These weights are not the lighest that could be shown but it is the range that is acceptable if the study is based on realism rather than phantasy. They include 3% for Δv variation, 10% for future growth, and redundancy for some of the major ecological systems. There are many extremist ways of reducing these Earth orbital departure weights but the more conservative approach was assumed for the detail weight analysis associated with this study.

The forward motion of the vehicle will probably be oscillating from a minimum drag configuration of 0° (nose forward) to a maximum drag configuration of 90° (broadside) during the time it is in the target planet capture orbit. As the vehicle oscillates, the W/A value will also vary and this

variation will have an effect on the vehicle lifetime in orbit. The W/A values are shown in Tab. 6-1. The W/A values for the vehicles associated with this study vary between wide limits depending upon the configuration, mission, and Earth entry condition. Therefore, Tab. 6-1 merely shows a range of W/A values to give an indication of the magnitude of W/A values that can be expected.

Tab. 6-1 NOMINAL RANGES OF W/A VALUES

Target Planet	Mars	Venus
W/A_{0° lbs/ft^2 (nose forward)	250 - 350*	350 - 500
W/A_{90° lbs/ft^2 (broadside)	70 - 100	100 - 150

*This value will approach the 700 - 800 lb/ft^2 range with some of the more expensive missions and small diameter vehicles.

6.2 Scaling Coefficient, Mass Fractions and Nomographic Vehicle Weight Determination

Following a number of detailed designs, a body of firm point design data on the standardized versions -22, -23, -28 and 28V became available from which extrapolation could be made toward larger and smaller sizes with completely high accuracy. In addition extensive and systematic analyses were undertaken of thrust structures and engine-tank integration aspects involving the tank configurations belonging to the four classes mentioned above and the three nuclear engines specified in the second section of this report. Finally, extensive investigations were made of fuel conservation systems and in connection therewith of the combined solar heat and meteoroid shield which protects the fuel tanks until just prior to their respective burning period.

From this body of data, a comprehensive set of scaling coefficients was established. The scaling coefficients refer to all types of engines considered

and to combinations of one to four engines, to thrust structure, aft adapters, residuals, refrigeration, forward adapter, the tanks, the heat and meteoroid shielding, boil-off insulation and mass fractions. From the mission point of view, scaling coefficients were established also for weights, jettisoned on route, such as W_{j34}, W_{j23} and W_{j12}.

From these scaling coefficients the mass fraction can readily be computed. Knowing the mass fraction, mass ratio and specific impulse, the major weight groups of a given vehicle stage are quickly computed.

An example of the nomographic arrangement of the final data for a given combination of structural configuration and engines is shown in Fig. 6-2.

6.3 Vehicle Weight Determination Computer Program

The weight analysis of the type of interplanetary vehicle (I/V) under consideration here can be divided into two main phases:

I: Determination of the weight, W_{C3} at the beginning of (heliocentric) return coast to Earth.

II: Determination of the Earth orbital departure weight (ODW), W_{A1}.

The reason for this distinction is a practical one. W_{C4} is determined partly by the crew size and the weight of the vehicle's operational payload; in this respect it is not different from weight groups in the other phase. But to another part, W_{C3} is a function of Earth return conditions and hyperbolic excess velocity at Earth arrival. This latter dependency can, in principle, be a cause of considerable variation as well as uncertainty. The second part relies on celestial mechanics and is well predictable.

By programming these two computation schemes separately, they can either be used independently, or jointly in cases where a complete computation for a mission is required. The flow chart (Fig. 6-3) shows the rough schematics of determining the weight W_{C3} as function of the Earth arrival mode and weight of operational payload.

Once W_{C3} is known, or given or varied parametrically, the planet departure performance and the engine and thrust characteristics of PM-3 the departure propulsion module are needed. From these, the ignition weight at the Mars departure maneuver is determined and so forth back to Earth departure (Fig. 6-4).

W_{C3} is the principal independent variable. Secondary independent variables are the weight W_{j23} jettisoned during the capture period and the weight W_{j12} eliminated during the outbound transfer. The weight W_{j23} is specified in fractions of the vehicle weight at the end of the capture period. The ratio W_{j23}/W_{S3} is a useful parameter to characterize scientific payload carried aboard the vehicle during a reconnaissance mission; or to define the cargo load of a vehicle supplying an orbital base or a surface base at a later time. The weight W_{j12} is specified in fractions of the vehicle's arrival weight W_{C2}, i.e. the weight at the end of the outbound heliocentric coast. The ratio W_{j12}/W_{C2} is a useful parameter to characterize the size of the crew and the amount of spare parts to be used during the outbound leg of the mission; or to define the number of passengers to be carried to the planetary base, but which do not return with the same flight. Thus, these two parameters are equally useful and applicable to initial reconnaissance.

In this arrangement, the program permits the user to insert his own assumption regarding the Earth arrival mode and weight eliminations during capture and during outbound as well as return transfer.

The program can easily be modified to apply to powered or non-powered fly-by; or be extended to bi-planet missions.

The program does not use simply impulsive velocity changes during the mission. The initial orbital departure weight is determined on the basis of finite propulsion during departure and arrival in the respective planetary field.

The method is described in greater detail in Vol. IV of this report.

Fig. 6-5 shows the orbital departure weight (ODW) W_{A1} versus calendar date computed on the 9074 according to the method outlined before. W_{C3} = 120,000 lb; the configuration selected corresponds to the tank and engine arrangement in the -22 class. The terms p_{12} and p_{23} designate weight eliminated during outbound heliocentric coast and during the capture period, both expressed in terms of the respective terminal weight of their phase. The required performance inputs were derived from the Earth-Mars-Earth transfer calculations for the periods indicated.

The weight minimums are practically the same for 1975 and 1977 because no dependency on Earth return conditions is involved. They fall off in the subsequent years. The weight minimum occured on different dates than the velocity minimum, but this was to be expected on the basis of earlier results. The discrepancy is based on the fact that the mass fractions of the individual stages are different.

Fig. 6-6 shows the ratio of the heliocentric return initial cost weight to the Earth orbital departure weight for the same conditions as described before.

Fig. 6-7 shows the weight eliminated during the capture period at Mars in terms of the orbital departure weight for the above described conditions.

The variation of the orbital departure weight with the return coast initial weight is shown in Fig. 6-8. The variation is shown for the previously established date at which W_{A1} is a minimum and for an interval of eight days before and aft of this date and finally for a day which precedes and which follows by 16 days with respect to the minimum weight date.

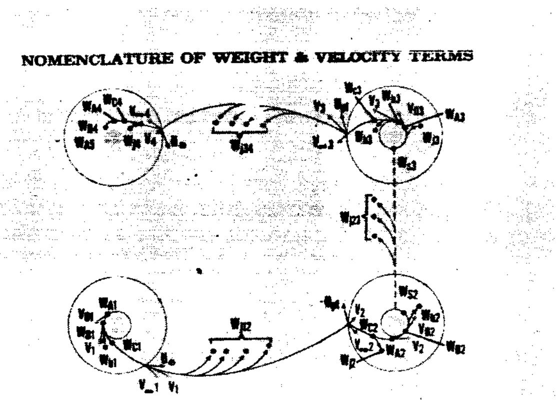


Fig. 6-1

INTERPLANETARY MISSION INFORMATION PROGRAM

DETERMINATION OF RETURN COAST INITIAL WT.-1 (EARTH RETURN)

TERMIN. WT.: W_{A5} APOLLO CAPSULE & CREW

APOLLO ENTRY: $W_{A4} = W_{A5} + W_{b4} + W_{p4}$

DRAG BRAKE CAPTURE: $W_{A4} = W_{A5} + W_{D.B.}$

RETRO - D.B.: $W_{A4} = W_{A5} + W_{b4} + W_{p4} + W_{D.B.}$

TERMINAL COAST WEIGHT: $W_{C4} = W_{A4} + W_{j4}$

RETURN COAST INITIAL WEIGHT : $W_{C3} = W_{C4} + W_{j34}$

Fig. 6-3

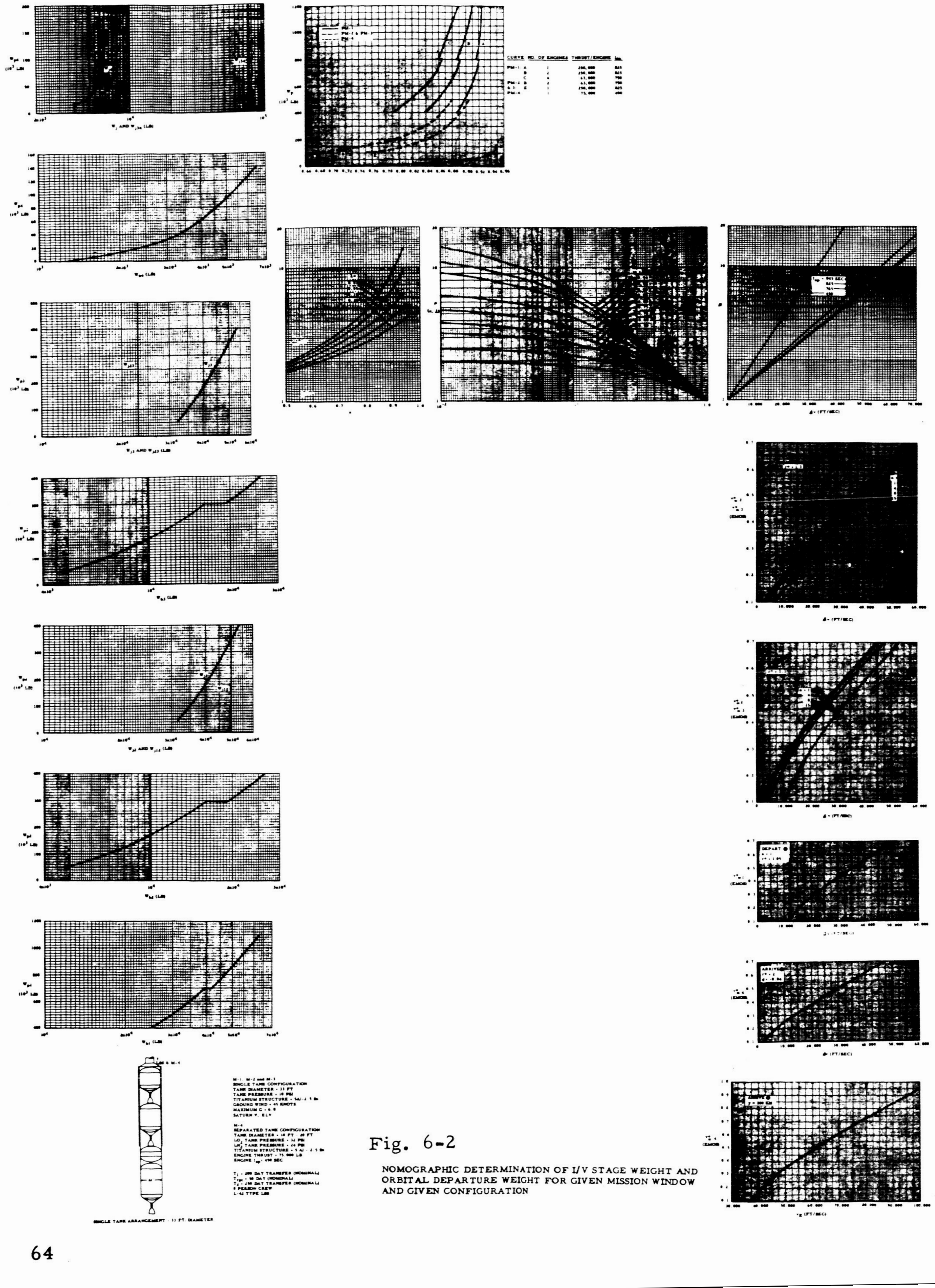


Fig. 6-2

NOMOGRAPHIC DETERMINATION OF I/V STAGE WEIGHT AND ORBITAL DEPARTURE WEIGHT FOR GIVEN MISSION WINDOW AND GIVEN CONFIGURATION

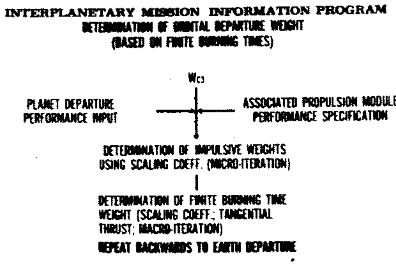


Fig. 6-4

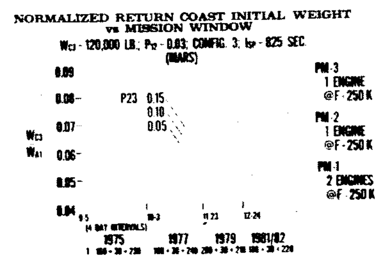


Fig. 6-6

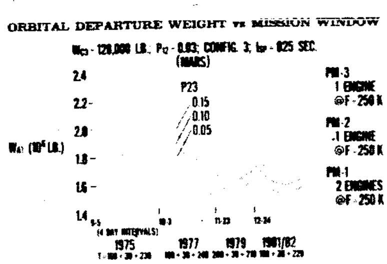


Fig. 6-5

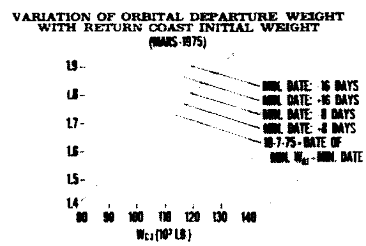


Fig. 6-7

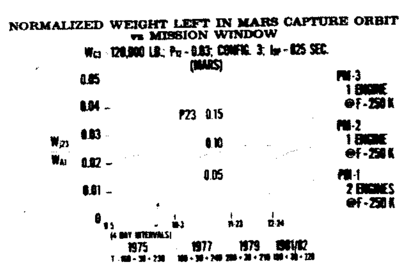


Fig. 6-8

7. EARTH LAUNCH VEHICLES

Planning for the early manned planetary missions is greatly affected by Earth launch vehicle imposed constraints. Three possible launch vehicles have been considered in the study and are defined as (1) two stage Saturn V, (2) Modified Saturn V and (3) Post-Saturn vehicle. These vehicles were chosen not only because of their possible availability during the seventies but also because they represent the range of constraints which may be applied to the program by the ELV.

The role played by the ELV in support of interplanetary operations is only to deliver the spacecraft into Earth orbit. Constraints to the program, due to the ELV, are primarily those due to its inability to place a complete interplanetary spacecraft in Earth orbit with a single operation. When the ELV is too small or the spacecraft is too large, the interplanetary vehicle must be delivered to orbit in sections with orbital operations necessary to perform final assembly and assure vehicle readiness. This means that each section of the spacecraft must be compatible with the ELV. Its diameter must not exceed that of the ELV to avoid hammerhead launch configurations, its weight cannot exceed ELV payload capabilities and its length must fit within the limits set by vehicle bending mode considerations or launch facility limitations. These constraints directly affect spacecraft design and either directly or indirectly affect such other factors as launch facilities, orbital operations and probability of success which in turn are significant cost determination factors.

The Saturn V is assumed to be capable of placing 250,000 lbs of payload into the assembly orbit. Its maximum payload volume of 112,000 ft^3 is based on a payload envelope of 33 ft in diameter by 155 ft long. These payload dimensions result in a launch vehicle which is 375 ft in overall length, considered the maximum due to launch facility considerations.

The hypothetical Modified Saturn is defined as a two stage to orbit vehicle made up of Saturn V subsystems which utilizes uprated F-1 and J-2 propulsion systems and differs from the Saturn only in its structural configuration. Instead of 33 ft, the diameter of this vehicle is 50 ft and its two stage length is reduced from 220 to 112 ft. Assuming that the overall length of the vehicle including payload should not exceed Saturn V, the new configuration allows a payload volume of 500,000 ft^3. A payload weight capability of 250,000 lbs in orbit is unchanged from Saturn V assumptions. The increase in payload volume results in a decrease in the minimum payload density to 0.5 lbs/ft^3. Such a value is now quite compatible with the density of the empty spacecraft (1 lb/ft^3) and therefore the whole propulsion section of the interplanetary vehicle can be delivered to orbit in one launch. (Except in cases where its dry weight exceeds 250,000 lbs.) Such a condition greatly reduces orbital assembly operations and thereby increases the probability of success. The larger diameter of the ELV also lessens the restrictions on spacecraft design by permitting such

allowances as clustered tank arrangements and shorter spacecraft lengths while reducing engine clustering problems.

The Post-Saturn's payload weight to orbit is assumed to be 10^6 lbs. Its payload volume is at least 10^6 ft^3 and its diameter is at least 70 ft. Such a vehicle is assumed to place a minimum of constraints on the interplanetary program. It is capable, in all cases, of delivering the complete spacecraft to Earth orbit in one launch, although in most cases the spacecraft requires additional propellant.

Even in the case where spacecraft modules are fully tanked on the ground, there will be a requirement for orbital fueling if only to supplement boil-off losses or to provide topping requirements. During this study it was found that a much more extensive use of orbital tankers is desirable. In fact, the orbital propellant tanker was found to be one of the primary requirements for preparing interplanetary vehicles in Earth orbit.

To illustrate the differences in launch configurations as well as the number of launches needed to assemble one interplanetary vehicle in orbit, Fig. 7-1 shows the requirements of a 1975 manned mission to Mars. The differences due to the ELV and its payload capability affects not only spacecraft design but ground and orbital operations as well. The number of launches needed to deliver the I/V's to orbit, which is dependent on the ELV payload capability and its reliability, defines a launch rate and the extent of launch facilities necessary to support this rate. The launch rate in turn defines the minimum orbital operation period and will influence the size of the orbital crew.

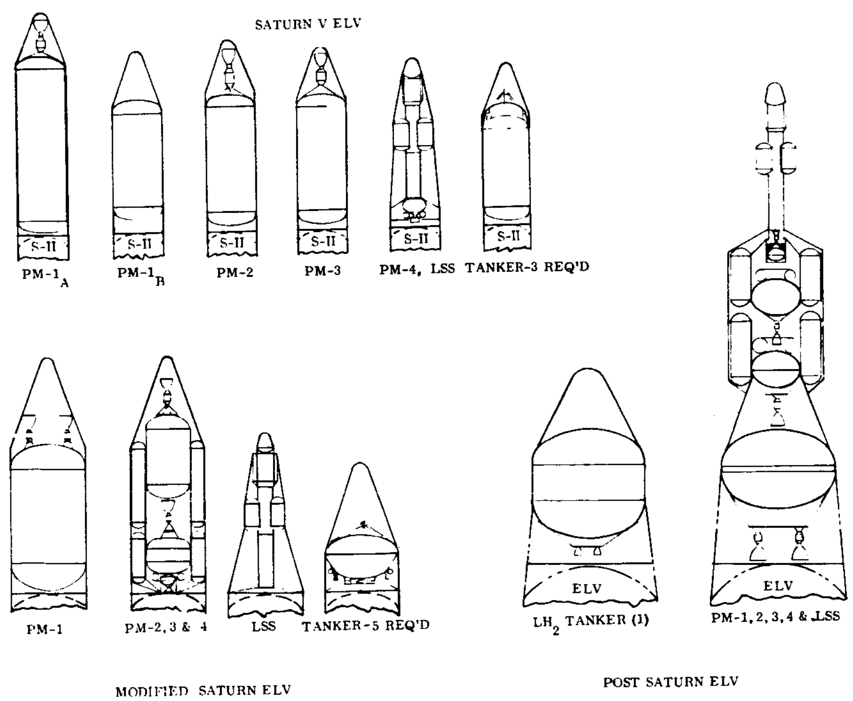


Fig. 7-1 LAUNCH CONFIGURATIONS - MISSION III

8. GROUND HANDLING AND LAUNCH OPERATIONS

The primary functions which must be performed and their dependence on each other are shown in the ground operational schematic in Fig. 8-1. Operational time estimates and a further breakdown of the operations have been determined with a tentative facility schedule. If the launch period is to be fixed at some value, then the amount of facilities must vary with the number of required launches are varied. Because of a variety of interplanetary missions and vehicle configurations, the number of ELVs required to support them also vary. Therefore, the amount of ground facilities listed in Tab. 8-1 are given as a function of the number of launches during a 100 day period. By correlating the number of ELVs for each spacecraft configuration, the extensiveness of the launch facility may be determined for any mission. The orbital operation period becomes the governing criteria for ground operations because it requires all of the ELVs to be launched during this period. The number of launches is dependent on the particular mission and the type of ELV. The primary schedule constriction is at the launch pad. To support the launch of one ELV, the pad is utilized from 16 to 20 days. Therefore during the interplanetary ELV launch period (100 days) each pad can support six launches. At the VAB, the scheduled cycle time per ELV is 40 days and therefore two assembly bays are needed to keep one launch pad busy.

Facilities \ No. of ELVs Required for Mission	6	12	18	24	30	36	42	48
Launch Pads	1	2	3	4	5	6	7	8
VAB High Bay Areas	2	4	6	8	10	12	14	16
Spacecraft C/O and Assembly Test Bays	3	3	3	2	2	2	2	2
Engine-Reactor-Module Assembly Buildings	3	3	3	2	2	2	2	2

Tab. 8-1 LAUNCH FACILITY REQUIREMENTS SATURN V ELV

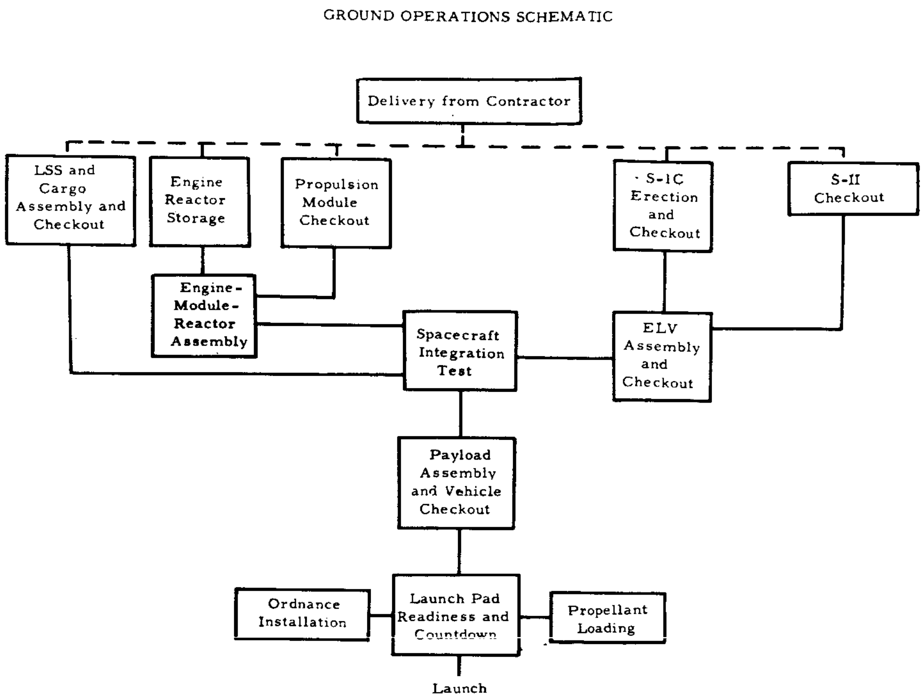


Fig. 8-1 GROUND OPERATION SCHEMATIC

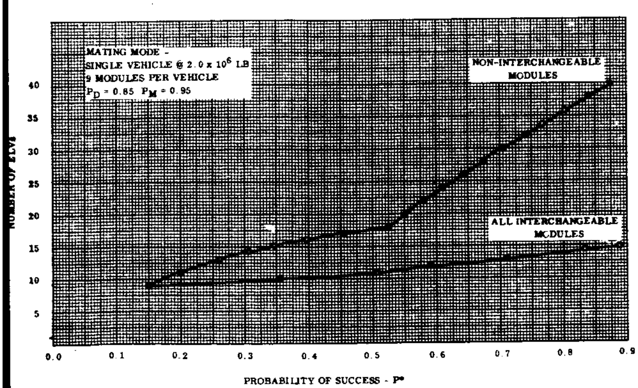


Fig. 9-1 EFFECT OF MODULE INTERCHANGEABILITY ON ELVs

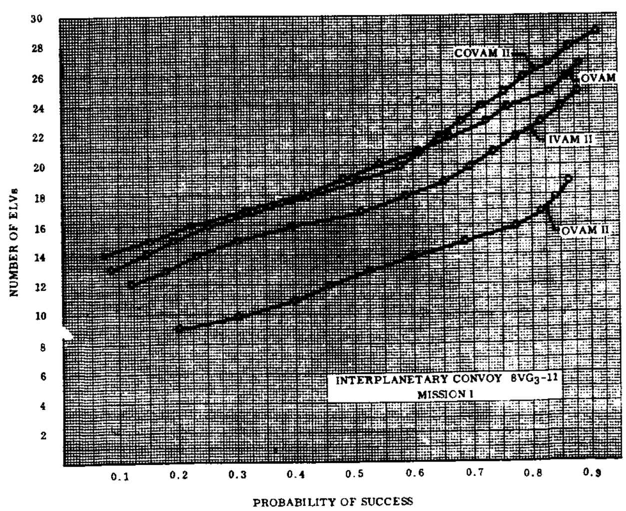


Fig. 9-2 EFFECT OF MISSION MODE ON ELV REQUIREMENTS

9. LAUNCH REQUIREMENTS

The need for a good estimate of the number of launch vehicles required to deliver the interplanetary vehicle to orbit is very important to program planning. Not only does the number of launches have a significant impact on the operating costs, it also has an important bearing on the size of the launch facility and the operations which must be performed there as well as in orbit. Mission velocity requirements, the size of the spacecraft, the mission mode and the mission objectives, each defined by the mission, has an affect on the number of ELVs as does the payload capability and the reliability of the ELV.

The preparation of an interplanetary vehicle for its mission for simplicity has been categorized in one of three primary operations: ELV payload delivery, orbital mating of the spacecraft and orbital tanking of the spacecraft. Because it cannot be realistically assumed that all operations can be successfully performed, reliability factors have been assigned to each main operation and together with knowing the number of times each operation must be performed, it is possible to compute the probability of success for accomplishing orbital launch readiness. If the computed probability of success is unacceptable, redundancies will be necessary to raise its value. But to gain the advantages of a redundant operation, an additional ELV to provide the operation must be acquired. The total number of vehicles required to perform the mission with a certain probability of success is then the summation of the minimum number of ELVs and the redundant ELVs.

One of the most important factors which influence the rate of increase in the probability of success when redundancies are added is interchangeability. This term applies to the modules within the spacecraft as well as the vehicles within the convoy. Redundancies are most effective when all of the modules of the I/V are alike.

Because the propellant requirements for one maneuver are different than the requirements of another, interchangeability of all modules is infrequent. The greatest amount of interchangeability results from use of the tanking mode. Although the modules within each vehicle tend to be unique, the propellant that they require is not. Each tanker which delivers the propellant is interchangeable. With the tanking mode, each module delivered to orbit requires one tanker to supplement its propellant requirement. (A module capable of containing 400,000 lbs of LH_2 will carry only about 200,000 lbs of LH_2 when it is launched because of the payload weight limit.) Therefore half of the minimum number of ELVs needed to place the complete interplanetary in Earth orbit are tankers and that half is completely interchangeable. The effect of interchangeability on ELV is shown in Fig. 9-1.

There are several alternatives for launching the interplanetary vehicle to orbit when, as with the Saturn V, the ELV cannot delivery the complete vehicle in one operation. The sections that are launched may be fully tanked, completely empty or partially loaded with propellants. Because of payload density limits, modules that are fully tanked result in smaller spacecraft sections and therefore more divisions or sections per vehicle are required. By partially tanking these modules on the ground, the tanking mode can tax the weight as well as the volume limits of the ELV payload and reduce the number of tankers needed for orbital propellant delivery.

As already described, several assembly modes have been considered for the interplanetary vehicles. Their different ELV requirements are due to changes in vehicle configurations and to dissimilarities in the ground rules for convoy preparation. A comparison of assembly modes in terms of ELVs is shown in Fig. 9-2. With the exception of OVAM II, the differences due to the assembly modes are not significant.

A comparison of the Post-Saturn and Saturn V shows that the larger ELV is capable of carrying four times the weight and ten times the volume of a Saturn V payload. Because of its almost unlimited volume carrying capability and the resulting reduction in orbital assembly operations, the Post-Saturn ELV requirements are less than 1/4 (the payload weight factor) the number of Saturn V vehicles even though its reliability of .75 is less than the .85 value assumed for Saturn. For the same reason less Modified Saturn ELVs are needed to support an interplanetary mission than Saturn V although neither its payload weight capability nor its reliability is any greater. The comparison of the ELVs given in Fig. 9-3 shows the advantage of the greater payload volume capability.

A variety of missions, mission objectives and vehicle designs resulted in a number of spacecraft configurations. For each, the number of ELVs needed to deliver its convoy to orbit was determined using the various assembly modes and type of ELV as parameters. The results given in Fig. 9-4 are applicable to Mission II or III requirements. A breakdown of Mission I requirements given in Fig. 9-5 shows the various combinations of modules which make up the ELV payload and the number of redundancies needed to achieve .75 probability of success.

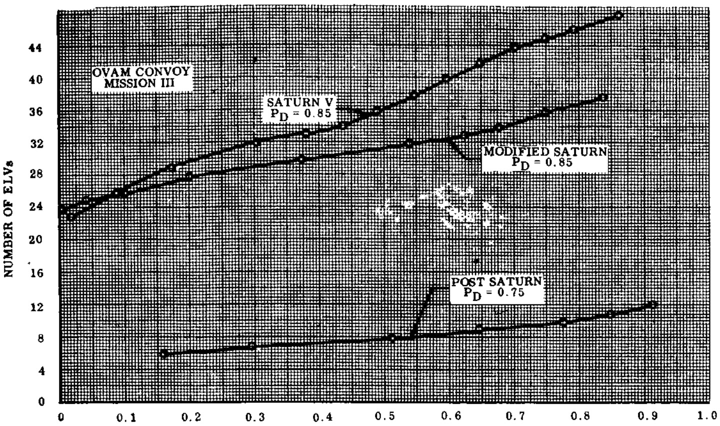


Fig. 9-3 PROBABILITY OF SUCCESS

Fig. 9-3 EFFECT OF ELV ON LAUNCH REQUIREMENTS

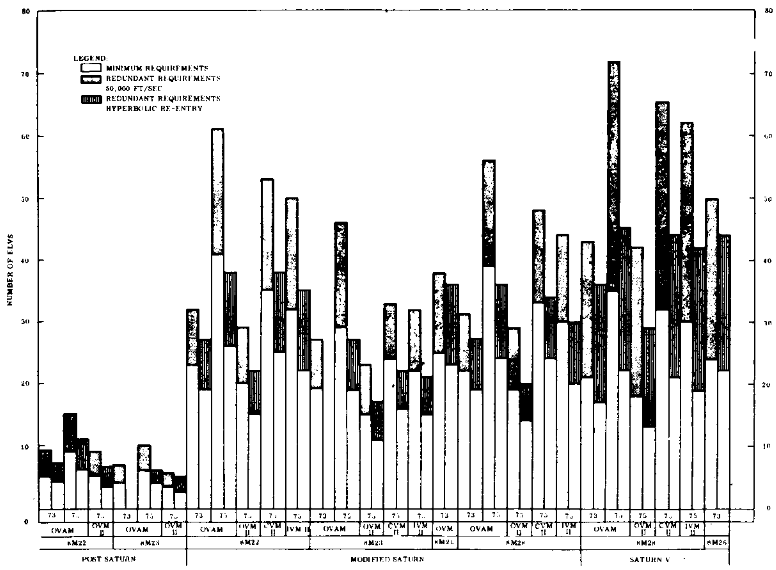


Fig. 9-4 MISSION III E L V REQUIREMENTS

Fig. 9-5 BREAKDOWN OF LOGISTIC REQUIREMENTS FOR ORBITAL READINESS OF 3 INTERPLANETARY VEHICLES (MISSION I) FOR DIFFERENT NERVA ENGINES AND I/V CONFIGURATIONS USING SATURN V OVAM
(Min. - 100% reliability, 75% - probability of success)

	$8VGC_2$-5						8VGC(CG)-10								$8VG_2$(CG)-9				$8VG_2C$-8				$8VC_3$-11		8VGCG-12	
Modules Carried by ELV	5/8		5/9 & 5/14		5/13 & 5/16		10/1		10/2		10/4		10/6		9/1		9/2		8/5		8/6 & 8/11					
	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%	Min	75%
M-1A	3	6					3	6	3	6	3	6	3	7	3	6	3	6	3	7	3	6	3	6		
M-1B, M-2, M-3, LSS	1	2																								
M-1B, M-2, M-3, C	2	4																								
M-1, 2, 3			3	5																						
M-1					3	5																			3	5
LSS			1	2																						
C, C			1	2																						
M-2, M-3, LSS					1	2																			1	2
M-2, M-3, C					2	4																			2	4
M-1B, M-2, M-3C							3	7	3	6	3	7	3	7												
M-3N, LSS							1	2	1	2	1	2	1	2												
M-3N, C							2	4	2	4	2	4	2	4												
M-1B, M-2															3	7	3	7	3	7	3	7	3	7		
M-3 (CG), LSS															1	2	1	2								
M-3 (CG), C															2	4	2	4								
M-3, LSS																			1	2	1	2	1	2		
M-3, C																			2	4	2	4	2	4		
TOTAL MODULES	6	12	5	9	6	11	9	19	9	18	9	19	9	20	9	19	9	19	9	20	9	19	9	19	6	11
LO_2 Tankers	6	8	6	9	6	8	1	3	1	3	4	6	1	2	3	5	2	4	3	5	3	5			2	4
LH_2 Tankers	7	10	8	11	4	7	8	11	7	11	3	5	5	7	6	8	6	9	7	10	6	8	5	7	6	9
TOTAL	19	30	19	29	16	26	18	33	17	32	16	30	15	29	18	32	17	32	19	35	18	32	14	26	14	24

10. ORBITAL ASSEMBLY OPERATIONS

Delivery of the spacecraft modules to the assembly orbit permits orbital operations to begin. In addition to spacecraft assembly, orbital inspection, testing and checkout operations will need be performed in order to verify its readiness for the 300 to 500 day mission. A schematic of the orbital operations given in Fig. 10-1 shows the primary operations and the sequence in which they are performed.

Need for manned performance of orbital operations will be provided by an orbital launch crew which is not composed of mission crew members. The size of the launch crew and the tasks which they must perform are dependent in part on the ELV and its launch rate. More orbital labor is required for Saturn compatible interplanetary vehicles than for Post-Saturn launch spacecraft. But an increase in the number of personnel to perform these tasks will not decrease the orbital preparation period. Due to the greater number of Saturn V ELVs needed to deliver the convoy, the length of the orbital period will be governed by the maximum ELV launch rate. In the event that an OLF is not available, its primary function, that of housing the launch crew, could be provided by the LSS of the interplanetary crew vehicle. It would initially contain special equipment for conducting vehicle checkout and support operations during the orbital period. The LSS would be returned to its mission configuration at the conclusion of the orbital operations. A launch crew of 12 men could be supported by the LSS during the orbital period. Initially the launch crew in the reentry vehicle is delivered into the assembly orbit by a Saturn 1B ELV. Crewmen perform an active part in the mating procedures and serve as override monitors when a hazard or malfunction is imminent. An inspection of the vehicle is then made by the crew. Mobility around the vehicle and access to the other convoy vehicles is provided by space taxis. After visual inspection is completed, damage is assessed and the required replacements and repair kits are requested to the Earth base. The launch crew is then divided into separate teams to perform vehicle checkout, subsystem repair or replacement, and vehicle maintenance operations. The crew must also assist with the orbital tanking operations. When vehicle preparations are completed, the mission crew is delivered to the spacecraft and the launch crew departs via the same carrier.

The period during which the spacecraft is in Earth orbit is dependent on and has a bearing on several factors. The relation between orbital decay and altitude causes a constraint on either the minimum orbital operation period or the ELV payload capability. The amount of propellant boil-off and the vulnerability to meteoroid hazards increase as the orbital period is extended. But shorter orbital preparation times have a great impact on the launch facility due to its need for higher launch rates. In addition, the length of the orbit operations period may dictate the size of the orbital launch crew. All of these factors were considered in arriving at a target value for the maximum preparation time in orbit of 100 days. In most cases this maximum

time period is fully utilized when the Saturn V is the ELV. Saturn V launch facilities, for most interplanetary missions, are either fully extended or overburdened by the required launch rate defined by the number of ELVs and the 100 day preparation period. Its dependence on the ground facilities requires orbital operations scheduling to coordinate both ground and orbit capabilities to provide an efficient interaction. A schedule and sequence of orbital operations has been made for both Saturn V and Post-Saturn type interplanetary vehicles. Scheduling of orbital operations for Post-Saturn launched interplanetary vehicles is not so dependent on ground facilities. Fewer ELVs are required and because of its greater payload capability, the whole convoy can essentially be delivered to orbit completely assembled, on the same day. The orbital period is then defined by the time needed to perform vehicle readiness verification. This in turn is dependent on the size of the launch crew. Orbital delivery of the LSS 30 days prior to launch of the propulsion section and the other two service vehicles, permits the bulk of the test and checkout tasks to be performed without influencing convoy preparation time.

ORBITAL OPERATIONS SCHEMATIC

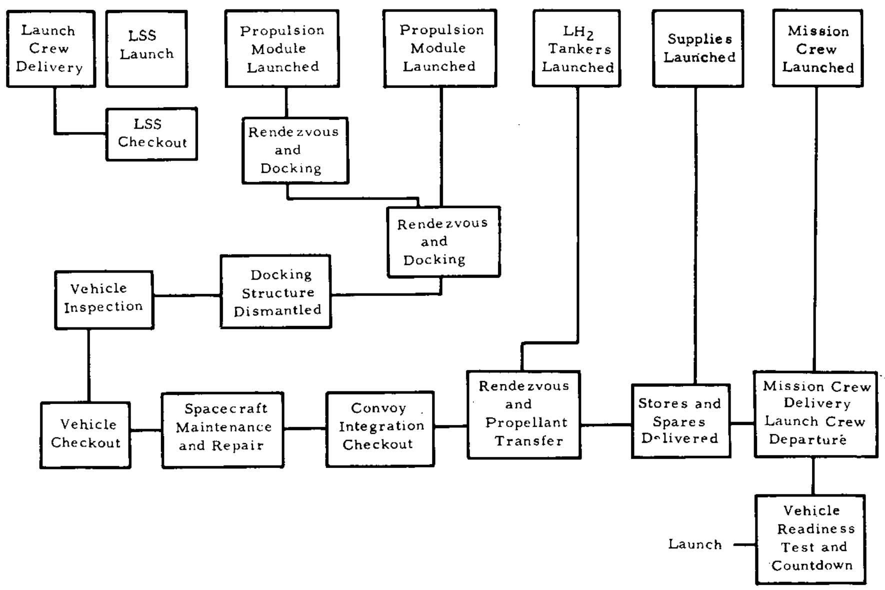


Fig. 10-1 ORBITAL OPERATION SCHEMATIC

26981

PART 3

MANNED INTERPLANETARY MISSIONS

by

B. P. Martin
Lockheed Missiles and Space Company
Contract No. NAS8-5024

FOREWORD

The studies reported herein were prepared by Lockheed Missiles & Space Company for the George C. Marshall Space Flight Center, Huntsville, Alabama, under Contract No. NAS 8-5024 (follow-on). The results of the Early Manned Interplanetary Mission Study are to be presented in three volumes. Volume 1 is this unclassified Summary Volume. Volume 2 will be the complete final report consisting of two parts; Part A the bulk of the Final Report, unclassified, and Part B, a small classified supplement containing nuclear propulsion data. Volume 3 will be a very brief condensed summary. Volumes 2 and 3 are expected to be published during February 1964.

Section 1
MISSIONS

Manned Mission Studies. The first phase of the Manned Interplanetary Mission studies considered three planetary reconnaissance flight profiles: short stopover on a satellite orbit, single launch flyby trajectory passing both Mars and Venus, and flybys passing only one planet. These previous studies established the general mission requirements for several transportation system designs as related to surface and orbit launch booster capabilities, advantageous launch year, and crew life support concepts. The flight profiles had as their basic objective the reconnoitering of the target planet for information not only of scientific value but which would be of significant interest in planning the large scale surface exploration missions to follow. All of the above considerations integrated with state of technology prognostications for the early 1970 time period placed the single planet flyby first in availability with the short stopover and grand tours second.

Numerous other mission mode concepts are available which could satisfy the reconnaissance mission objectives. These, however, were not included in the present studies, for they in general exhibit mission requirements on the order of those derived for the short orbital stopovers. Further they are sensitive to the increasing trajectory requirements associated with the opposition years up through 1980.

The primary focal point on which any planetary mission study must be based is the growth capability and operational availability of both surface launch boosters and orbital launch vehicles. And in addition, in the latter category, the use of either chemical or nuclear propulsion yields ramifications affecting surface boosters and mission utility. The current state of nuclear-engine and vehicle-system development and the consequent necessity of multiple rendezvous in Earth orbit appear to preclude manned planetary reconnaissance more ambitious than the single-planet flyby investigated herein. As is to be expected in preliminary analyses of the nature employed in the second phase system studies, the estimates of spacecraft subsystem

weights will increase with their usual multiplicative influence on booster capability and performance. And so it will be seen, especially in the case of Mars, the flyby missions once considered reasonable now are marginal, and indeed the mission availability now depends more on the development of a nuclear vehicle system.

Nevertheless, the current studies have focused on flyby missions with emphasis on the 1974 Venus conjunction and the 1975 Mars opposition. The basic assumptions prevalent in the booster analysis and spacecraft design are: Saturn V class surface launch vehicles only; orbit launch vehicles of technical sophistication not beyond the S-IVB and nuclear stages projected from current programs; Earth entry system of the modified Apollo type retro-braked to its design entry speed; a three man crew (not by optimum choice but dictated by booster capability and the Apollo); and technological accomplishment reasonably expected of the necessary subsystems by the early 1970's. It should be emphasized that a change in any one of the above assumptions could easily alter not only the spacecraft design but the availability of the flyby mission and its potential level of reconnaissance usefulness. Consideration of these possibilities leads one to the problem of evaluating and justifying the employment of the early manned planetary flyby mission as a significant contributer to an overall National Space Program. Various views concerning this problem will be presented subsequently.

Flyby Trajectory Characteristics. It was initially intended, for the second phase studies, to analyze in as realistic detail as feasible the previously determined desirable missions, which were: a one year Venus flyby, a high and low energy Mars flyby, a two planet flyby,and short orbital stopovers at Mars and Venus. As the study progressed through to the first presentation, it became evident that the flyby missions to either Mars or Venus were highly desirable from the standpoint of surface launch vehicle capability and the requirements on the spacecraft subsystems, especially the Earth entry module, of near term availability. Furthermore, the realities of time phasing between launch vehicle development and trajectory requirements as dictated by nature in succeeding years favored the selection of the 1974 Venus conjunction and the 1975 Mars opposition. In both instances the trajectory requirements remain the lowest of all mission opportunities provided, in the case of Mars, the trajectory is of the low-energy, long-duration class.

Selected Flyby Trajectories. The vehicle system mass that must be placed on Earth parking orbit depends primarily on the Earth departure speed required and secondly on the entry speed. Because the entry system is to be retro-braked by storable liquid or solid propellant rockets, the mass of the entry module is strongly influenced by the entry speed. However, the velocity requirements for Earth departure are such that the mass of the escape boosters mask any mass increases in the entry system caused by operating it at off-minimum speeds. Thus, trip selection is quite simplified and attention may be concentrated on the general trajectory characteristics centered about the minimum Earth departure speed. The actual trajectory selection relies on deeper study involving the relative movements of both the parking orbit and heliocentric trajectory elements.

The regions about the minimum Earth departure speeds for the 1974 Venus and 1975 Mars flybys are depicted in Figs. 1 and 2 respectively. The charts are given for a lightside passage and a close approach altitude of 500 nm, which in terms of planetary radii is 1.15 for Venus and 1.20 for Mars. The values of the flyby characteristics depend on the passage distance that is planned.

For Venus, the speed requirement decreases with a lowering of the passage distance, but of course is limited in principle to a grazing pass and practically by the extent of the atmosphere and the accuracy to which the trajectory can be held. Because of the obscuring dense, cloudy atmosphere, the selection of passage distance for visual and photographic reconnaissance is quite premature. In view of these problems, the choice given here is deemed a reasonable compromise pending further criteria.

The Mars flyby trajectories exhibit tendencies regarding the influence of passage distance on Earth departure velocity in reverse of those described for Venus. In 1975, speeds well below 0.2 EMOS are available provided one wishes to pass at five or more planetary radii. The effect, however, diminishes quite rapidly with increasing distance, resulting in small gains in lower departure speeds for passage distances above three radii. The advantages of lower speed, however, are not available without penalty, for planetary inspection suffers with increasing passage altitude. As in the case of Venus, the optimum close approach distance must await

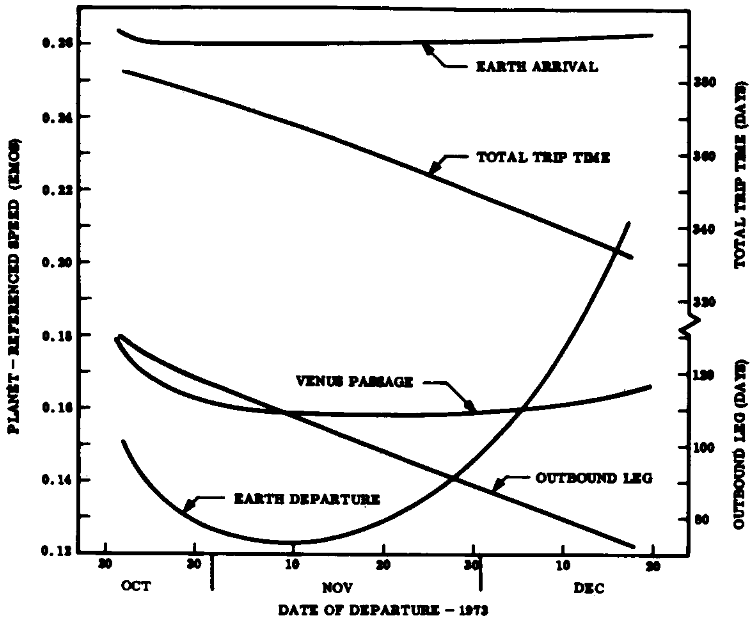


Fig. 1 Mission Characteristics, 1974 Venus Flyby

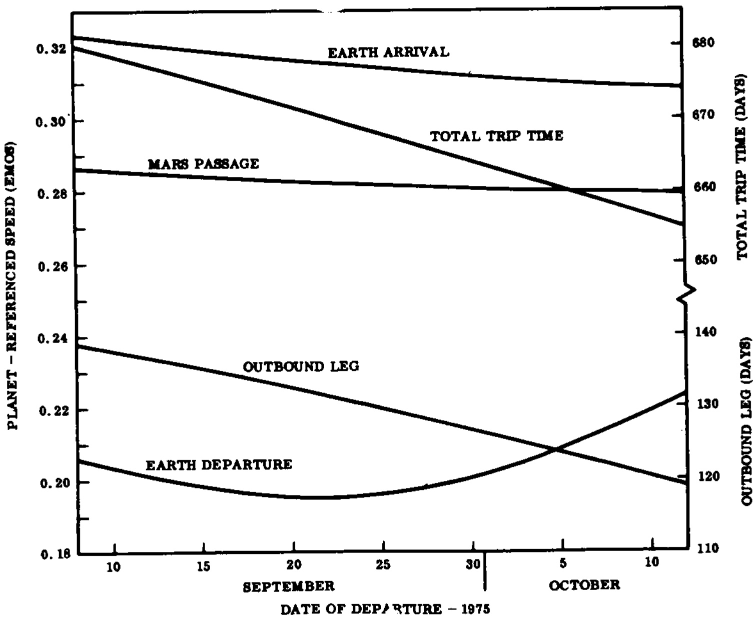


Fig. 2 Mission Characteristics, 1975 Mars Flyby

further definition of reconnaissance effects, length of observation period, and savings in departure speed requirements.

Utilization of Manned Flyby Missions. Because orbit stopover and landing missions require such large masses on Earth orbit, necessitating either post-Saturn launch vehicles or multi-rendezvous techniques, the only manned interplanetary missions which appear to be available in the 1970's are the flybys to Venus and Mars.

These trips would be very useful toward the subsequent manned landing missions, not only because of the planetary data obtained by on-board and probe instrumentation, but also because of the knowledge gained regarding the function of the many spacecraft subsystems, the man-machine relationships and behavioral characteristics of the crew over very extended periods under confinement and strict disciplines dictated by the criticality of their functions.

Section 2

EARTH ORBIT LAUNCH CONSTRAINTS

During preliminary system studies of interplanetary missions, it is generally assumed that launch from Earth orbit occurs with an instantaneous burn at perigee of an escape hyperbola with the plane of the departure trajectory lying in the orbit plane. This is a fair analysis for high acceleration launch stages with launch occurring just as the hyperbolic asymptote passes through the orbit plane.

More likely escape conditions are illustrated in Fig. 3. For any given mission, the required departure hyperbolic asymptote is in motion relative to inertial space. The plane of the parking orbit is in motion due to the asymmetrical gravitational attraction of Earth. Thus, the asymptote and orbit plane are coplanar for only short periods and, for the rest of the time, an angle (shown as ϕ in Fig. 3) will exist between them.

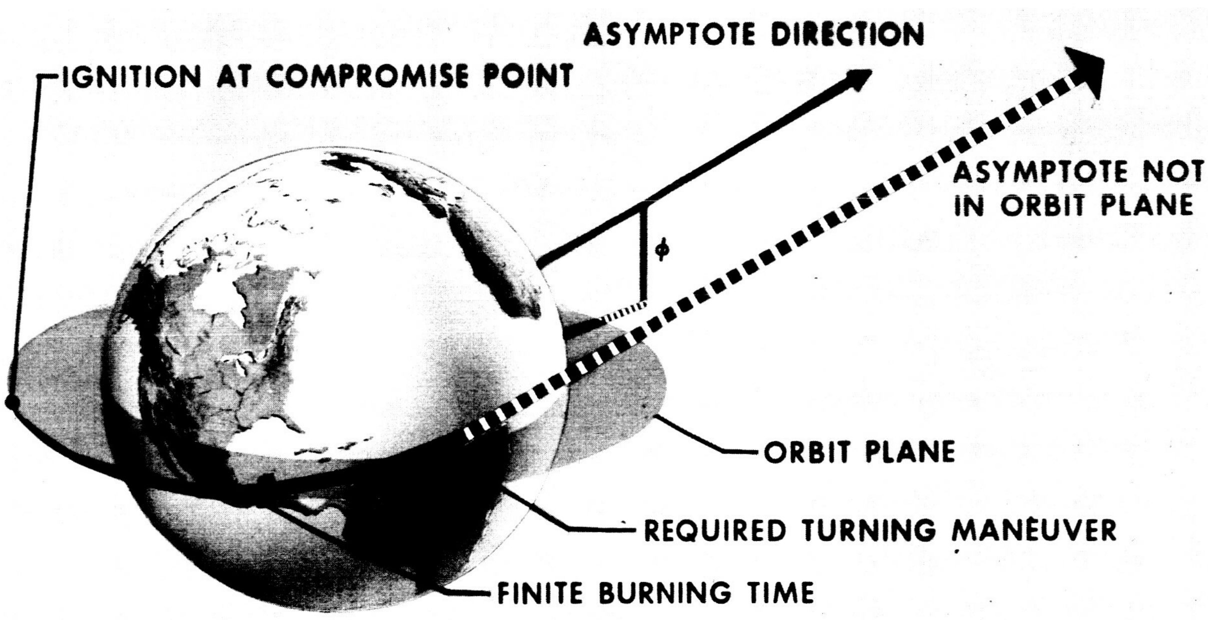


Fig. 3 Earth Escape Launch Conditions

As a result of this angle ϕ and the use of long-burn launch boosters, ignition will generally not occur at perigee of the launch hyperbola, finite burning time must be considered, and a turning maneuver is required. An analysis of this plane change requirement was made during the study since it will greatly affect the mission launch window.

For discussion purposes here, launch window can be defined as that period of time, usually measured in days, in which a vehicle can obtain the necessary energy and direction of travel required for a mission by suitable launch from an established parking orbit. This differentiates from mission window which is concerned with the energy requirement only. "Push-button" window is that period of time, usually measured in seconds, in which proper engine ignition must be initiated within any orbit pass inside the launch window period so as to accomplish the required launch maneuver. The boundaries on each window can be set arbitrarily or phenomenologically.

From this study of orbit launch constraints, the following can be said:

- The problem of launch window and the associated plane change requirement must be a major consideration in the selection of orbit launch systems and operations. Launch windows depend on vehicle capability to absorb the plane change requirement and can range from several days down to several hours.
- A special dual burn technique was uncovered which could be very helpful in reducing the plane change ΔV requirements for launch from orbit. However, more work is needed to obtain the best split of energy input between the two impulses considering both single and two separate propulsion systems.
- The relationship between the declination of the required departure asymptote and the parking orbit inclination profoundly affects the size of the launch and push-button windows.
- Proper selection of a nominal launch date can affect the number of launch windows within a favorable mission window as well as the length of each.
- The total ΔV resulting from a coplanar launch, after maneuvers in orbit to maintain the departure asymptote within the orbit plane, is greater than that which results from absorbing the plane change during the launch itself.

However, it may be advantageous to use such an inefficient scheme in order to reduce the size of the Earth departure propulsion system because the plane change system could be jettisoned in orbit prior to launch.

- Selection of an orbit inclination slightly higher than the declination of the departure asymptote will keep the plane change penalty at a low value over the longest period of time.

A brief study was made of the "push-button" window. It showed a penalty in the order of 5 to 50 feet/second per second of delay around the optimum launch point. Study results in somewhat greater depth will be reported under Contract NAS 8-2469.

No analysis in depth was conducted under the present interplanetary missions study contract. Such a study now would be premature for these reasons:

- Large launch delays in any one orbit pass are not expected. Foreseeable booster systems should be capable of igniting on time. If a large delay developed, the crew could simply wait for a subsequent pass while correcting the difficulty.
- The order of magnitude of the ΔV penalty for launch delays indicates that, for small delays, uncertainties in booster performance might produce more significant ΔV discrepancies.
- Analysis in depth can't be accomplished until many fundamental inputs, such as selected mission and booster system characteristics, are known.

The preceding discussion should not be interpreted to mean that factors influencing the push-button window should be ignored. It does mean that, for this study, there are other items more deserving of attention.

Section 3
CREW UTILIZATION AND REQUIREMENTS

During the second half of the study, ionizing radiation, physico-chemical regeneration, and trace contaminant control were emphasized. Thermal control was also studied in somewhat more detail. Few important changes were made in basic life support and environmental control system weights. Radiation shielding weights were reduced substantially, and power requirements were lowered by about a third, largely due to decreased estimates of the power needed for thermal control. Decreased radiation shield weight requirements greatly lessened the attractiveness of the open metabollic system. These requirements are now considerably less than the weight of an open system water supply, and the use of such metabolic supplies and/or wastes for radiation shielding can no longer justify the high weight of the open system. The concurrent downward trend of the power penalty and slow but steady progress toward increasing the reliability of the physico-chemical closed system also have tended to shift interest to the closed systems.

Figures 4 and 5 illustrate current estimates of the weights and power requirements for representative missions. Weights shown include gas supply, carbon dioxide removal, trace contaminant control, food and water (metabolic and utility), crew support equipment, temperature and humidity control, pressure suits, and sanitation. Container weights, equipment, redundancies and tools were included; power penalties were not. "Nearly open" system weights assume that utility water is recovered; atmospheric water, urine, carbon dioxide, and feces are not recovered. "Nearly closed" system weights assume all of the above but feces are recovered. Leakage is assumed to be 1 kg per day per three men. Metabolism is assumed to be 2820 kcal per man per day, and the latent loss is assumed to be 47% of the total metabolism.

Power requirements are generally applicable to mission durations of 300 to 1000 days. For missions over 400 days add about 25 watts per man per 100 days to the values shown. Peak power is much more difficult to specify than average power.

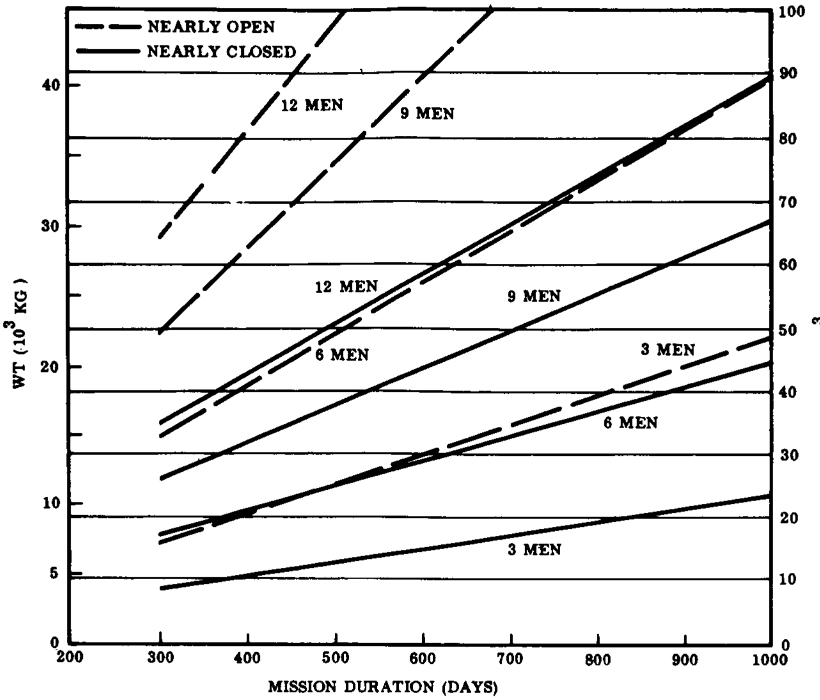


Fig. 4 Life Support and Environment Control System Weights

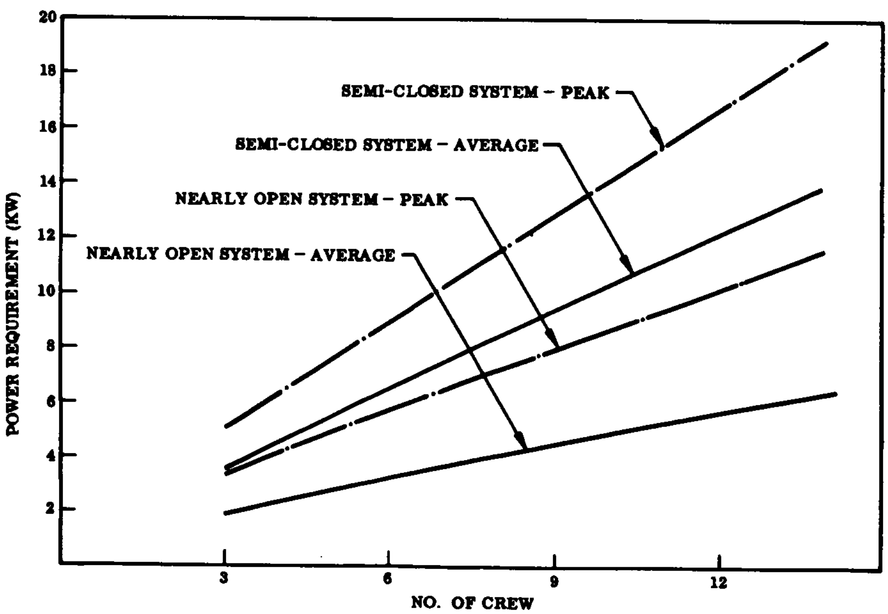


Fig. 5 Life Support and Environmental Control System Power Requirements

Recent developments of importance to life support and crew protection are treated briefly below and followed by a summary of the most important research and development problems still confronting manned space flight.

Atmosphere Selection and Gas Supply. In 1963, two successful tests of the 380 torr, 50% O_2, 50% N_2 atmosphere were achieved. On the pessimistic side, only a few men were involved and the tests did not last more than 30 days. On the optimistic side, no significant problems were encountered. The Johnsville tests included sudden decompression from this atmosphere to 180 torr pure O_2 without incident. Lower thermal power requirements make the reduced atmosphere more attractive. Leakage remains problematical. Servicable pO_2 sensors now appear low in weight and more reliable. Subcritical cryogenic storage, theoretically the preferred storage method for extended missions, is still developing slowly.

Carbon Dioxide Removal. Electrodialysis now definitely appears superior to the molecular sieve for systems involving the hydrogenation of CO_2 to water and subsequent electrolysis of water.

Trace Contaminant Control. This subject is, perhaps, the biggest question mark of all. Simulation experiments which have had notable problems in this area have not exercised anything like proper precautions. Successful simulation tests have not, on the other hand, encountered more than a small part of the problem. Nuclear submarines have now completed over one hundred 60-day cruises with leakage virtually zero. Spaceship contaminant production should be less per unit volume and removal devices at least as efficient even using present technology. Submarine contaminants build up steadily throughout a cruise while leakage holds spacecraft concentrations below 30 days production (EMPIRE nominal leakage) even for contaminants completely unaffected by the removal devices. The longer exposure and fear of synergistic and/or additive effects of other space stresses, and the fear of emergency effluxes quickly loading the small volume of the spacecraft argue for further study of this problem. In spite of the heavy cost, great technical difficulties, and possible danger, a complete simulation of this problem involving man seems a much more feasible solution to this problem than the individual study of every substance which might be a problem

with respect to production, detection, removal and MAC (maximum allowable concentration). The Navy, for instance, quotes one million dollars and 2-1/2 years for the determination of one MAC. Individual substances are, however, obvious candidates for such particular studies.

Thermal Control. Thermal control still seems best effected by space radiators. More precise analyses have reduced the power requirement for this function to about 1/4 of that previously estimated. Several factors contributed: (1) lower and more stable α/ϵ surfaces are available or foreseeable; (2) finer analysis showed 70% of the electronic and 50% of the ECS power losses can be cold plate cooled and kept from reaching the cabin; (3) by modest increases in the sizes of fans, heat exchangers, and ducting, the fan power required was reduced to some 300 watts (3 man system); (4) favorable radiator orientations (away from the sun) are practical; (5) use of vacuum distillation instead of vapor compression for urine recovery; and (6) original estimates were generous simply because they were rough. The power savings were achieved at the expense of a relatively trivial 30-lb increase in weight. Heating systems seem to have little application even where heat sources such as reactors are available. Heating systems should, however, be investigated – especially if the meteoroid problem takes a turn for the worse.

Diet and Metabolism. Though some arguments have been made during the last year for 4000 kcal plus diets based on the demonstrated difficulty of working under weightlessness, much too little evidence has been adduced to expect such high energy requirements throughout a mission even at zero g and certainly not for a g vehicle. Dehydrated food technology has taken great steps forward in recent years as have packaging techniques. A wide variety of tasty foods should be available by 1970 with virtually no spoilage within up-to-three-year missions. Preparation of such foods for consumption requires minimum power and facilities. Zero-g consumption techniques are still problematical.

Physico-Chemical Regeneration. Atmospheric water recovery is still considered feasible by condensation-separation and filtration. Definitive tests with trace contaminants in the atmosphere have not, however, been reported. Utility water and

urine recovery now seem best accomplished by vacuum distillation under gravity and also under weightlessness if surface tension separators prove practicable. Vapor compression techniques also look good but require more power. 95% recovery looks possible at the moment. Though many methods for O_2 recovery from CO_2 are being studied, Sabatier reduction, electrolysis of the water produced and pyrolysis of the methane to recover hydrogen still looks best tentatively. The methane pyrolysis step is currently the most frought with difficulties. Other methods with potential advantages – especially at zero g – include MSAR's Fused Carbonate Salt, MRD's Solid Amino Salt, and P_2O_5 solid electrolyte systems.

Biological regeneration continues to be studied but still suffers from most of the disqualifying problems pointed out previously.

Weightlessness. With the exception of blood pooling in the legs of astronaut W. Schirra recent Russian and American flights, of durations up to five days, have been encouraging with respect to man's ability to endure weightlessness. Theoretical considerations of muscle biophysics incline many to believe a simple exercise program will maintain muscle tonus over extended missions at zero g. However until studies have been conducted for at least 30 days and preferably much longer periods under weightlessness it will be impossible to state with any confidence the effects of weightlessness for much more extended missions.

Ionizing Radiation. Figure 6 shows skin dose vs. aluminum shield weight for missions of 300 to 700 days duration. The curves for three probabilities of exceeding the calculated dose are shown.

Fluxes were taken from the NASA document quoted. Shielding calculations were based on total ionization. This method is probably conservative for shield thicknesses up to 30 gm/cm^2. The method for calculating the probabilities was not stated, but it is interesting to note that the fluxes given for particles with energies greater than 100 Mev for 365 day exposures were 30, 10, and 2 times, for probabilities of 0.001, 0.01, and 0.1, respectively, the total flux greater than 100 Mev for 1959 reported in the NASA Solar Proton Manual – the heaviest year well studied to date. The dose to the blood forming organs is roughly half of the skin dose.

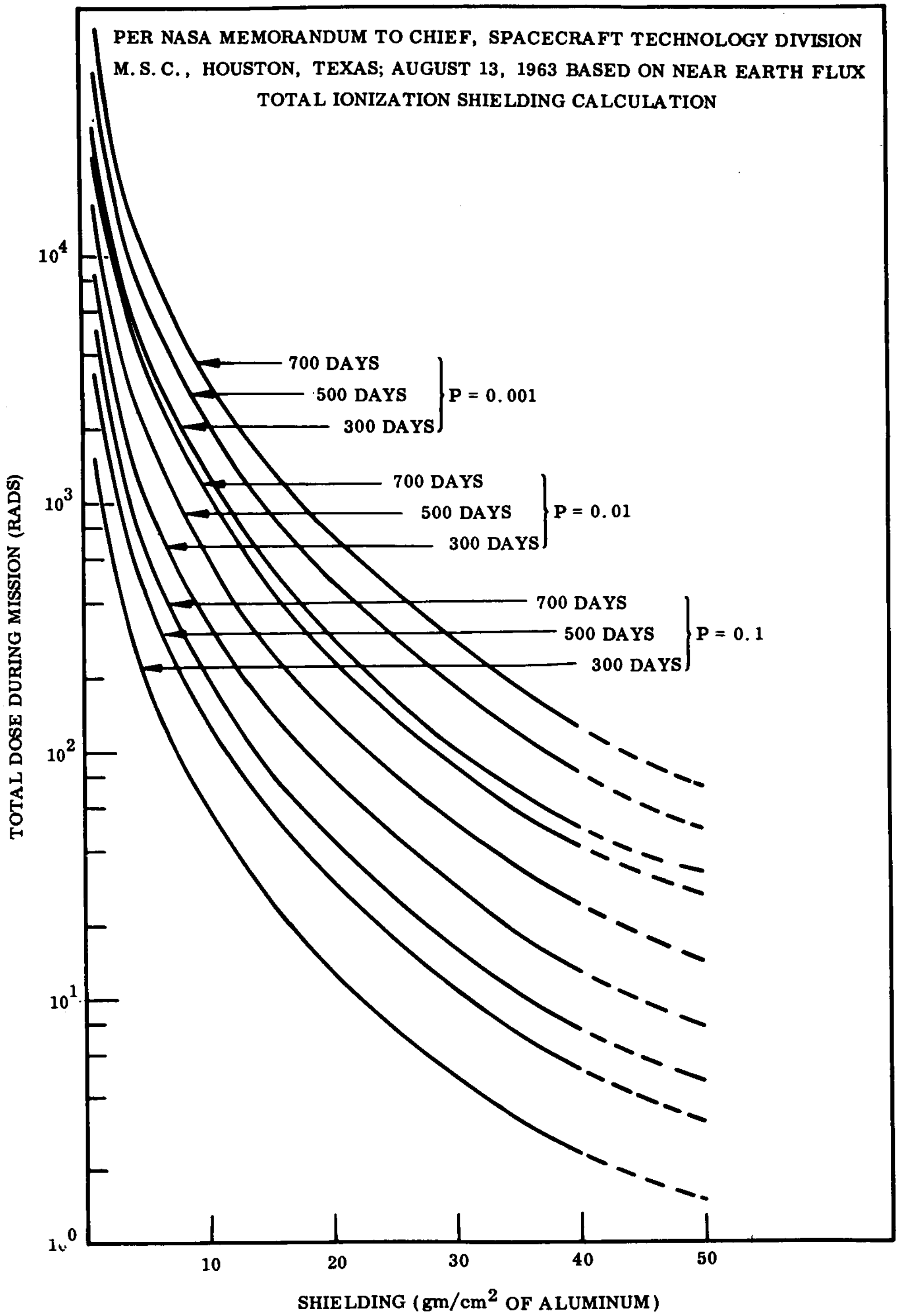


Fig. 6 Solar Proton Skin Dose Vs. Shield Weight

Table 1 shows the possible effect of distance from the sun upon total dose for five typical missions. $(1/R^2)_{mean}$ refers to the time average over the trajectory. For the Venus mission, $(1/R^3)_{mean}$ is substituted since it would lead to higher values. Values are in astronomical units.

Table 2 (based on $(1/R^2)_{mean}$ for the Mars missions and $(1/R^3)_{mean}$ for the Venus mission) shows the shield weights required to have a probability of less than 0.001 of receiving over 200 rads to the blood-forming organs (assumed to be five gm/cm^2 inside the skin). Cosmic and reactor doses are both assumed equal to 20 rads/year. Composite shields of aluminum and polyethylene are preferable for the shield thicknesses, shielded volumes, and configurations now envisioned. Raising the required probability to 0.01 would cut shield weights by about 40%.

Reentry. A recently completed LMSC reentry body study determined the life support and crew protection system to weigh some 600 kg for 6 men. Present acceleration restraint systems seem barely adequate for the 9.5-g peaks encountered. Present cooling methods including ventilated reflecting suits seem adequate to protect against 93°C interior walls if skin pain from suit conduction heat leaks can be eliminated.

Remaining Research and Development Areas. (Not necessarily in order of importance). (1) Extended exposure to exotic atmospheres, (2) effect of N_2 upon g tolerance, (3) development of subcritical gas storage, (4) toxicity problem definition including production, detection, removal, and MAC's – particularly study of new problems added by catalytic burner, (5) more sophisticated thermal control analyses, (6) selection of urine, utility water, and carbon dioxide regeneration methods for intensive development, (7) as always – the effects of weightlessness alone and in combined stresses, (8) physiological problems in the rotating vehicle, (9) time, space, and energy distribution of solar protons, interaction of such particles with shielding, and RBE's of these and cosmic primaries, (10) centrifuge runs of proposed high peak, slow decay reentry accelerations, and (11) reentry heat protection systems – simulated profiles.

Table 3 shows the development status of important life support components.

Table 1

THE EFFECT OF DISTANCE FROM THE SUN UPON RADIATION DOSAGE

Mission	$\left(\frac{1}{R^3}\right)$ Max	$\left(\frac{1}{R^2}\right)$ Mean	SKIN DOSES WHICH WOULD HAVE BEEN RECEIVED FROM ALL 1959 – 60 SOLAR EVENTS COMBINED† – RADS: If all Events Occurred When the Vehicle was at Perihelion $(1/R^3)_{Max}$: Under 2 gm/cm^2 Al Plus 5 gm/cm^2 Polyethylene	If all Events Occurred When the Vehicle was at Perihelion $(1/R^3)_{Max}$: Under 4 gm/cm^2 Al Plus 10 gm/cm^2 Polyethylene	If all Events Occurred at the Mean Value of $(1/R^2)$: Under 2 gm/cm^2 Al Plus 5 gm/cm^2 Polyethylene	If all Events Occurred at the Mean Value of $(1/R^2)$: Under 4 gm/cm^2 Al Plus 10 gm/cm^2 Polyethylene
Mars, Light, Hot, 1973 578 Days	2.8	1.03	263	65	97	25
Mars, Light, Cool, 1973 664 Days	1.0	0.39	94	24	37	10
Mars, Light, Hot, 1975 585 Days	4.3	1.02	405	101	96	24
Mars, Light. Cool 1975 739 Days	1.45	0.49	137	35	46	12
Venus, Light 1974 370 Days	2.68	1.18‡	252	63	111	28

† Flux from NASA TR R-169, "A Summary of Solar Cosmic Ray Events," by H. H. Malitson and W. R. Weber in Solar Proton Manual, ed. by F. B. McDonald, pp. 12-13, Washington, D.C., Sep 1963

‡ $\left(\frac{1}{R^3}\right)_{Mean}$

Table 2

RECOMMENDED DOSAGE AND SHIELDS†

	Reactors 20 Rads/ Year	Cosmic 20 Rads/ Year	Total To Blood Forming Organs P = 0.001 – Rads	Shield Weight (gm/cm^2) Al + Polyethylene
♂ Light, Hot, 1973 578 Days	31.5	31.5	200	2 + 18
♂ Light, Cool, 1973 664 Days	36.5	36.5	200	2 + 14
♂ Light, Hot, 1975 585 Days	32	32	200	2 + 18
♂ Light, Cool, 1975 739 Days	40	40	200	2 + 16
♀ Light, 1974 370 Days	20	20	200	2 + 13

† Solar proton flux at Earth extrapolated using $(1/R^2)$ Mean for ♂ missions and $(1/R^3)$ Mean for ♀ mission.

Table 3

LIFE SUPPORT AND ENVIRONMENTAL CONTROL DEVELOPMENT STATUS

FLIGHT-TYPE UNITS PRESENTLY QUALIFIED OR IN DESIGN, FABRICATION, OR TEST	PROTOTYPES BUILT AND EVALUATION UNDERWAY OR COMPLETE	PROTOTYPES UNDER DEVELOPMENT	CONCEPT FEASIBILITY ESTABLISHED BY LABORATORY MODEL	LABORATORY INVESTIGATION IN PROGRESS	CONCEPT ANALYTICALLY SOUND, DESIGN FEASIBILITY YET TO BE ESTABLISHED
• High pressure and supercritical gas storage • LiOH for CO_2 removal • Condensation-separation water removal • Dew point and relative humidity sensor humidity control • Charcoal odor removal • Evaporative and space radiator thermal control • Chemical disinfectant waste storage • Dehydrated food storage • Zero-g food consumption	• Liquid cryogenic storage • Sabatier reactor • One-step CO_2 hydrogenation • Fuzed salt CO_2 electrolysis • Rotating water electrolysis cell • Regenerative molecular sieve CO_2 removal • Atmospheric water reclamation • Catalytic burners (trace gas) • Vacuum and vapor compression urine distillation • Electrolysis of urine • Vacuum drying, heat sterilization and incineration of feces	• Solid electrolyte CO_2 reduction • Electrodialysis CO_2 removal • Selective solid sorbent CO_2 removal (charcoal, amino salt) • Chemical O_2 sources (KO_2, N_2O_2) • Thermoelectric distillation and electrodialysis urine processes • Various water electrolysis units, e.g., vortex cell, ion exchange, P_2O_5 vapor cell • Biological cell exchange	• CO_2 freezeout • Platinum diffusion water electrolysis • Urine evaporation • Cold sterilization of wastes • Zero gravity personal hygiene equipment	• Selective solid sorbent CO_2 removal (Ag_2O, MgO, CaO) • Biological waste processing • Carbohydrate synthesis • Photosynthesis for food production • Thermal switch • Radiation shutters	• Methane reduction • Long chain hydrocarbon products of CO_2 hydrogenation • Waste heat thermal control • Biological trace gas control

Section 4

GUIDANCE AND CONTROL

The interplanetary mission, for the guidance and control systems study, has been divided into the following phases: (1) injection into the heliocentric trajectory; (2) midcourse; (3) planet approach; and, where stopover missions are considered; (4) the stopover orbit. Some phases, depending on the mission, are repated on the return leg.

During the first contract period, navigation techniques, accuracy requirements, guidance procedures, and the conception of an integrated spacecraft guidance system were developed for the injection, midcourse, and planet approach phases. The present study has been directed toward a refinement of techniques, extending the spacecraft guidance to include stopover orbit capability, and development of an instrument landing probe concept to aid in defining stopover orbit guidance requirements and surface reconnaissance techniques.

Midcourse Navigation. Interplanetary spacecraft designs currently being proposed generally include rotation for gravity simulation. When navigation measurements are made during rotation, star and planet tracking instruments must be isolated by mounting them on an inertially stabilized platform if accurate angle readings are to be obtained. An alternate method is to despin the spacecraft when angle readings are to be made.

Results of recent studies by Breakwell et al. indicate that measurements made in the vicinity of the two terminal planets (departure and arrival) provide enough information, and midcourse measurements may be deemphasized. This technique could eliminate the need for special instrument mounting. The midcourse trajectory could be established by measurements during the first 15 to 30 days after departure, and final corrections during the last 30 to 50 days before arrival.

Navigation measurements, as previously defined, consist of star, sun, and planet angles. The process of guiding the vehicle with these measurements is outlined in Fig. 7. The method outlined is based on the use of a nominal trajectory and linearized perturbations. Star and planet angles, measured at a preselected time, are compared with precomputed angles for the nominal trajectory. Angle differences are then multiplied by precomputed matrices to obtain the velocity impulse required to return the vehicle to the nominal trajectory.

Flyby Navigation. Trajectory conditions during planet passage are determined by the corrections made during planet approach. Altitude measurements obtained with a horizon scanner during the flyby will be used to determine the time of passage through pericenter (minimum altitude). Following planet passage, as the planet recedes from the spacecraft, the approach phase horizon scanner and star tracker will measure planet range and direction to establish the departure trajectory asymptote. Data smoothing with linear filter techniques may be used over several days before making a trajectory correction.

Stopover Orbit Guidance Concepts. If a planet stopover mission is selected, impulse requirements to correct the hyperbolic approach trajectory, and the retrograde impulse required for transfer to the stopover orbit, will be determined from the vehicle optical sensors and computer. The spacecraft inertial system provides control of the thrust vector during orbit transfer, and commands thrust termination. Sensor data processing techniques, such as linear filtering will be useful in reducing trajectory errors.

Stopover orbit characteristics will be obtained with measurement techniques defined in Fig. 8. A Mars mission has been selected for illustrative purposes because the Mars surface environment is suited to a wide variety of instrument probes and optical reconnaissance techniques. A horizon scanner and star tracker are provided to determine the local vertical, and a pulsed radar altimeter is provided for altitudes below about 10^3 miles. An accurate determination of the orbit ephemeris is required to relate surface reconnaissance information to the planet surface coordinate system, and to furnish initial conditions for entry into the Earth return trajectory. With

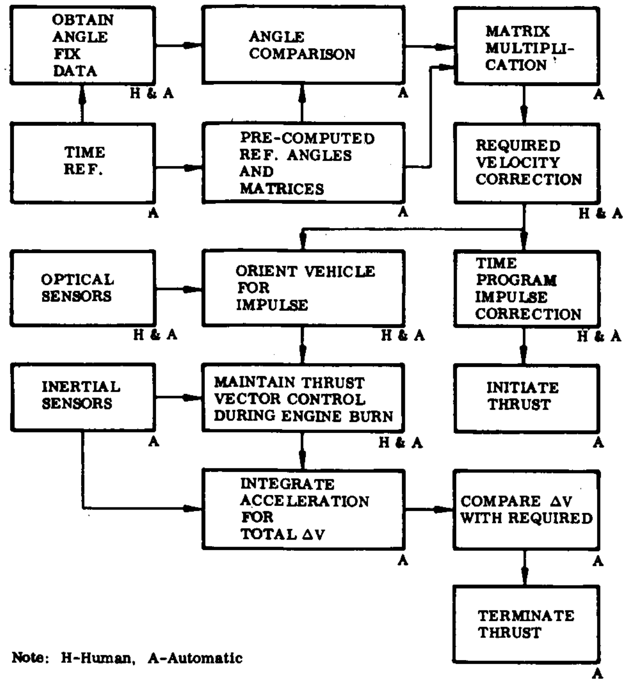


Fig. 7 Navigation Sequence Using Nominal Trajectory and Perturbation Scheme

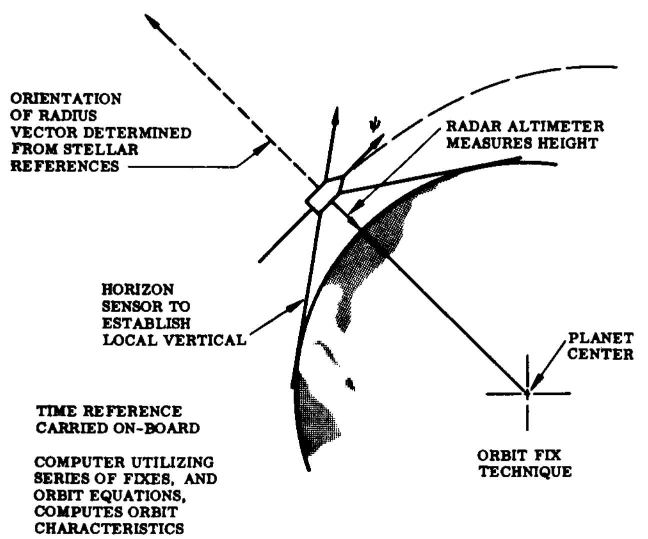
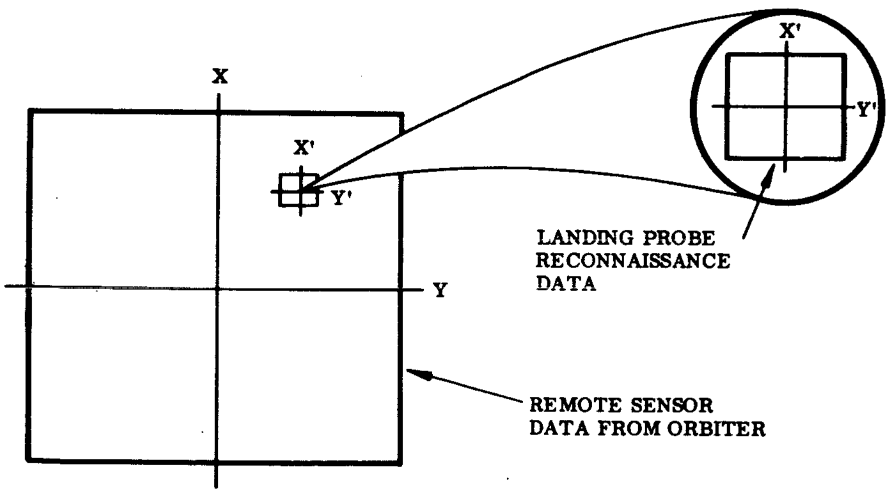


Fig. 8 Establishing the Mars Orbit Ephemeris

suitable data smoothing techniques the estimated residual position error could be reduced to the order of 10^3 feet for a 500-mile orbit altitude.

Guidance for landing probes is closely related to the probe functions. The probe candidate considered here includes a TV system for obtaining surface image data during the atmospheric descent. Inertial sensors are provided for retro-thrust control and atmospheric entry steering. At a preselected altitude determined with a radar altimeter, a parachute or drag chute is deployed. During the final parachute descent, TV ground images are relayed back to the manned orbiter and terminal corrections commanded. Image correlation techniques illustrated in Fig. 9 may be used to provide landing point control within a few feet for subsequent probes to the same area. The radar altimeter is also useful to determine TV image scale factor.



- LOCATE COORDINATES OF HIGH RESOLUTION PROBE DATA WITHIN REFERENCE FRAME OF ORBITER DATA.
- CORRELATE METEOROLOGICAL, SOIL & ENVIRONMENTAL DATA WITH SPECIFIC LOCATION AND TERRAIN IMAGE DATA
- DEVELOP GUIDANCE TRAJECTORY FOR SUBSEQUENT PROBES OR MANNED LANDER.

Fig. 9 Landing Site Reconnaissance Data Correlation

Equipment Physical Characteristics. Diagrams of the spacecraft and landing probe guidance systems are shown in Figs. 10 and 11. The earlier spacecraft configuration has been augmented by a horizon scanner, and radar altimeter for stopover orbit ephemeris determination. The probe system includes a parachute to stabilize the TV sensor attitude during the terminal descent and to provide lower velocity impact. The TV imaging sensor and data link transmitter permit image data correlation to be used for final corrections commanded from the orbiting spacecraft. Estimated weight and power requirements of the guidance system are summarized in Table 4.

Table 4

SPACECRAFT GUIDANCE AND CONTROL SYSTEM
(Estimated for 1970 State-of-the-Art)

	Weight (lb)	Volume (ft^3)	Power (W)
Trajectory Determination System	145	5	200
Attitude Control System	50	2	25
Computer System	70	3	150
Command Link Equipment	5	0.3	5
Mars Orbit Determination	75	2.0	90
Totals	345	12.0	470

NOTE: 1) Environmental control equipment and attitude torquers are not included.

2) Computer memory and clock are continuous; other items operate intermittently.

Human Functions in the Navigation System. Functions logically assigned to the human navigator include manual sextant observations, assistance to automatic devices by initiating fix sequences, and elimination of readings based on false targets. Complex decisions, such as those involved in the image correlation technique proposed for the impact probe, are obviously enhanced by human participation. System and component calibration checks, repairs, and resolution of anomalies in system performance, and programming of automatic sequences, can all benefit from the improvisational character of human intelligence.

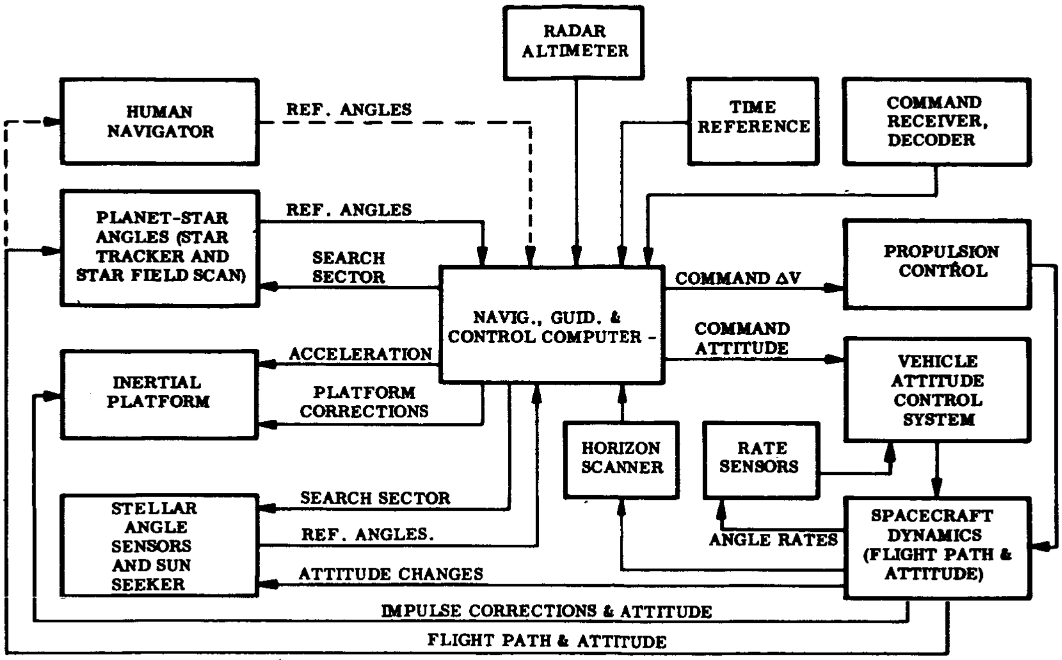


Fig. 10 Spacecraft Guidance & Control System

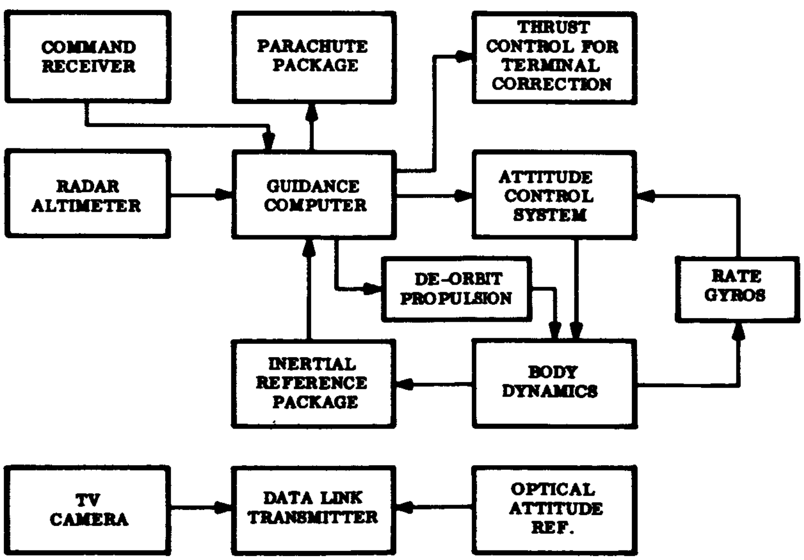


Fig. 11 Landing Probe Guidance System

Section 5
RECONNAISSANCE

Interplanetary reconnaissance requires a total, integrated system of data sensing, processing, correlation, and evaluation geared to supply planet description and/or next-step design data sufficient, ultimately, for manned landing and exploration of Mars. Hopefully, these data contributions will supplement a logical sequence of earlier Earth-astronomical and unmanned reconnaissance missions. However, the intimate knowledge of detail site and landing environment necessary to commit a manned lander appears attainable only through the coordinating efforts of an onboard space crew, applied to the repetition and refinement of specific site reconnaissance. The crew can perform this role only if it is possible to design an adequate, onboard data-processing, automatic-comparator, and display system to serve them. (Similarly, the crew is needed to reduce the masses of collectible data to storable/transmissible proportions.)

Preliminary review has been performed on the reconnaissance goals, capabilities, and problems of the three interplanetary missions: flyby, orbiting, and landing. The conclusions (much simplified) are summarized in Fig. 12. Out of this effort emerge the following relative mission concepts:

- The flyby is a limited, one-shot, narrow-swath reconnaissance tool, subject to continuous scale and angular-rate variation detrimental to optimum sensor design and to data interpretation. Its best use may be for operational test of future-mission equipment and possibly for accurate insertion of a reliable unmanned reconnaissance orbiter and/or lander.
- The manned orbiting mission provides a stable reconnaissance platform, capable of mapping the complete planet, onboard sensing of vast quantities of detail surface data at near-uniform scale, and if properly equipped, collating probed air and surface data. Major problems are mass-data-handling – automatic processing and display to aid crew tracking, correlation, sensor control, and evaluation.

	FLYBY	ORBITING	LANDING
GOALS: MANNED LANDING & EXPLORATION	OPERATIONAL TEST LIMITED RECONNAISSANCE	TOTAL MARS MAP ENVIRONMENT (DESIGN DATA) LANDING SITE CHOICES	FINAL SITE CHOICE SAFE LANDING SURFACE EXPLORATION RENDEZVOUS & RETURN
CAPABILITIES: DATA – SENSING ABILITY EXCEEDS DATA – HANDLING ABILITY	10^{12} BITS NARROW 1-SHOT RECON LIMITED PROBE DATA	TO 10^{16} BITS MAP TO 60 m OR BETTER MULTI-SITES TO 1.5 m MULTI-AIR/SURFACE PROBES	TO 10^{14} BITS AREA & SITES TO 1.5 m PROBED AIR/SURFACE DATA LANDING ALTITUDE & TOPO MANNED SURFACE EXPERIMENTS
PROBLEMS: LOW ALBEDO CORRELATION & PHYSICAL INTERPRETATION DATA – HANDLING & DISPLAY	SCALE & ANGLE VARIATION	PROBE-TRACKING & CORRELATION EXCESSIVE STORE/TRANSMIT REQT	PROBE GUIDANCE/DATA CORRELATION PHYSICAL INTERPRETATION LANDER GUIDANCE
CREW ROLE: OPTIMIZE SYSTEM – FILTER DATA	TEST FUTURE EQPT'	ADJUST MONITOR READJUST EVALUATE STORE TRANSMIT	SELECT LANDING AREA REFINE SPECIFIC SITE DATA CORRELATE DECIDE LANDING
EQUIPMENT WEIGHT: TRADEOFF PROBE NO. VERSUS SOPHISTICATION	1600–2200 lb (with 2 Probes)	Onboard: 2000 lb + Store/Transmit On Probe: 200–500 lb ea	Onboard: 2000 lb Store/Transmit On Probe: 200–500 lb ea On Balloon: 20–200 lb ea

Fig. 12 3-Mission Mars Recon Resume (Current Estimate)

- The landing mission requires an order higher of data-handling and correlation sophistication than the orbiter. In the course of repetitive orbits, preferred exploration areas must be located, detail landing sites and environment closely inspected by means of all sensors — onboard and probe-borne — spatial correlation of data must be established, and the safety of manned landing judged on-orbit. Landing and return-rendezvous sensor systems must be provided, along with surface exploration aids.

Sensor candidates include high- and low-resolution photography, infrared, hopefully laser radar,as well as the variety of physical particle and field detectors such as air, weather, chemical, and biological instrumentation, along with probe TV heads. Sample payloads and performance capabilities are treated in other EMPIRE reports. A major reconnaissance tool of all missions is the long-focal-length camera (3 meters or longer), with a design goal of 1.5 meters of surface resolution assumed. This stringent goal, coupled with the low-light conditions of Mars (commonly cited as one-third that of Earth), calls for fine system stability plus film high in both resolution and speed.

Unlike our Earth reconnaissance experience, straightforward physical interpretation of the unknown planet surface cannot be performed readily from the reconnaissance record. Here, multiband spectral reconnaissance may prove a useful tool, as in synchronous picture-taking with several film-filter combinations, assuming adequate laboratory experimentation has preceded the mission. However, final physical judgment must depend on on-the-spot probe data. The reliability of judgment of the safety of a particular landing trajectory and landing site must depend, first, on the success of the repetitive orbit-by-orbit process of reconnaissance, evaluation, and improved reconnaissance; and, finally, on the achievable refinement of data correlation through tracking, probe guidance, and final data collation.

Figure 13 blocks out the elements of an onboard data-handling system and shows complementary automatic and manual control loops, sensors, computers, reference aids, display, etc., to support the central crew functions of equipment and data control, evaluation, decision, and action.

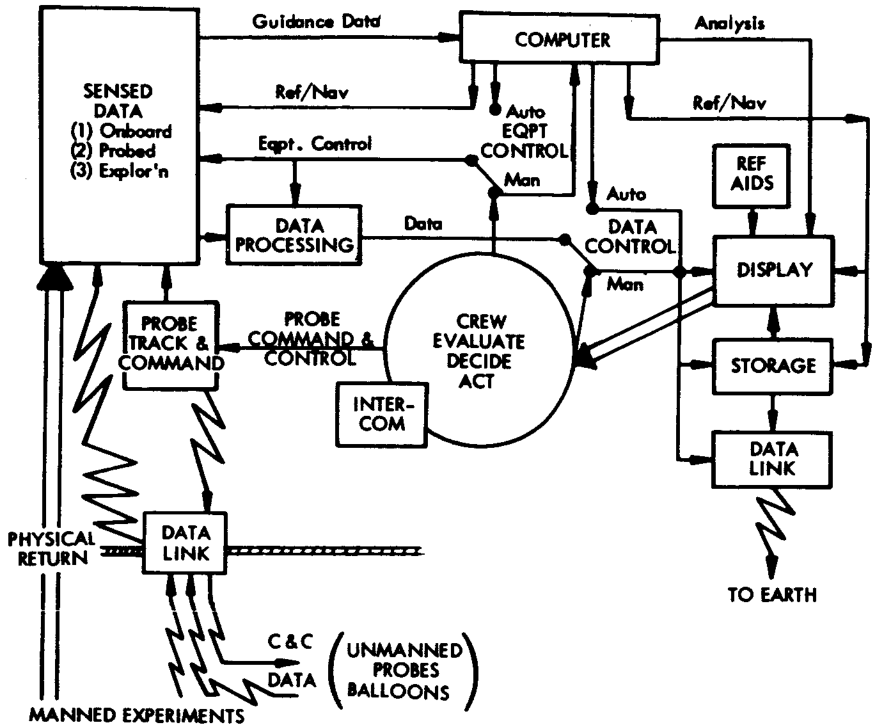


Fig. 13 Orbital Data-Handling System (On-Board)

The onboard reconnaissance evaluation capability, as noted earlier, is sought in order to permit the crew to fulfill two major functions:

1. To optimize reconnaissance on specific sites of interest by interorbital refinement and improvement of onboard sensing, by selective assignment and guidance of supplementary air and surface probes, and, finally, by accurate spatial correlation of all data. Time-scale constraints for the 350-nm orbital altitude include a 20-minute extreme maximum of viewing – or line-of-sight-transmission – time for any surface point (considered horizon-to-horizon), and a 138-minute maximum effective orbital period to monitor and evaluate past results and plan future orbits. Needs are most critical in the mission requiring a manned landing decision.
2. To act as a data filter, thus reducing data-storage and transmission requirements to a feasible volume – first, by collecting only the most needed data; and second, by selecting those segments to be stored or transmitted. Here,

the most drastic requirements occur in the pure orbital reconnaissance mission, where the extreme data-collection capability for complete-planet, fine-scale reconnaissance has been estimated at 10^{16} bits per mission. For coverage an order lower in refinement, IBM has estimated a storage weight of 16,600 pounds, assuming no transmission. Conversely, for complete transmission (i.e., only pre-transmission storage), communication of 10^{16} bits of data at 2 cycles per bit within a 100-day period would require the exorbitant transmission bandwidth of 2.3 gigacycles (2.3×10^9 cycles). Data reduction to the extreme feasible bandwidth of, hopefully, 1 megacycle would thus require a selection ratio by the crew of 1-to-2300 bits transmitted to bits collectible.

Feasibility studies on the system requirements for implementing onboard crew evaluation must include sophisticated estimates of requirements for rapid data-processing; storage and retrieval of reference-frames (both pre-mission and past-orbit); automatic aids to the comparison of past and current reconnaissance shots and the correlation of many-source data; display/control techniques to integrate the crew into the man-machine loop; computer size to perform the access and retrieval function, as well as more complex calculations. Feasibility studies of the rapid reconnaissance-evaluation function – still problematic for giant ground systems – are still in the embryonic stage for onboard space-vehicle application.

Section 6
POWER SYSTEMS

Electrical power subsystem studies have culminated in the recommendation of on-board power plants for three-man Mars/Venus flyby spacecraft configurations. As previously reported, nuclear and solar activated primary systems are considered leading candidates. The scope of recent studies has been expanded to include auxiliary or emergency systems and tradeoffs with the primary power plant.

Investigations have included the feasibility of the substitution of Brayton gas-cycle conversion equipment for the Rankine cycle turbomachinery associated with the SNAP 2 and SNAP 8 programs.

Auxiliary power plant investigations have included Gemini and Apollo type hydrogen-oxygen fuel cells as well as recent progress in reciprocating engine development. A power level of 5 to 8 kw was considered for each power system. Among the nuclear configurations, the primary power plant selected is an 8 kw nuclear/dynamic system utilizing SNAP 2/8 reactor technology and Brayton gas-cycle energy conversion.

Primary Power. Two primary power sources have emerged from the detailed studies. These are nuclear/dynamic and solar/dynamic systems.

The nuclear/dynamic technology reference is the current family of SNAP units: SNAP 10A, -2, -8 and SNAP 50/SPUR. Specific weight estimates are conservatively based upon current SNAP-10A, -2 and -8 epithermal spectrum reactor technology and SNAP 2/8 temperature levels and turbomachinery concepts. Unshielded specifc weights for nuclear Rankine systems are projected to be 300 lb/kw in the 5 to 8 kw range.

Nuclear/dynamic system investigations have been extended to include advanced SNAP reactor-activated Brayton cycle conversion systems. Because of the extensive operating experience with SNAP 10A and SNAP 2 reactors which will be available well in

advance of manned interplanetary flight, these are considered as potential Brayton cycle heat sources. Significant heat-source and heat-exchanger weight savings would be possible with the development of compact high temperature gas-cooled reactors which could be integrated directly into the Brayton cycle ducting, eliminating liquid metal to gas heat exchange. The status of current gas-cooled reactor development programs has been reviewed. Brayton cycle system unshielded specific weights are projected to be 350 lb/kw in the 5 to 8 kw range.

Biological shield weight, allowable biological dose, separation distance, structure and spacecraft stability tradeoffs have been computed for nuclear power plants. It has been determined that a composite shadow shield consisting of depleted uranium gamma shield, borated zirconium hydride gamma/neutron shield and lithium hydride neutron shield in conjunction with separation distances of 80 to 120 ft and allowable biological doses of 25 to 30 rads per mission (from power plant reactor) results in an acceptable configuration.

Calculated shield weights vary from 3000 lb for a 5-kw Brayton cycle power plant 120 ft from the nearest manned compartment to 6000 lb for an 8 kw Rankine cycle system 80-ft away.

Previous studies have dealt primarily with solar/Rankine power plants represented by Sunflower and ASTEC technology. Because of launch packaging constraints, the solar concentrator configurations of the most interest are the compact inflatable-rigidized concepts currently under development rather than stowed rigid sections of a metal reflector or one-piece systems. Earlier studies have been augmented by the consideration of the substitution of Brayton cycle conversion equipment for Rankine cycle counterparts. The advantages cited for nuclear-heated Brayton cycle systems apply equally to heat sources derived from incident solar flux.

An additional area of study concerning solar/dynamic power plants has been variable input flux design philosophy. Estimated average solar fluxes of 52, 130, and 240 w/ft^2 are estimated for near-Mars, near-Earth and near-Venus conditions, respectively.

Several optical approaches appear to be feasible including mechanical manipulations such as aperture variations and possible defocusing in high flux environments. A second approach involves designing concentrators for the local fluxes. A third approach makes use of multiple solar/dynamic subsystems operating in parallel. Projected solar/dynamic system characteristics vary over a wide range of values. The 3-kw Sunflower Rankine system will weigh approximately 275 lb/kw and require a 44-ft diameter erectable aluminum concentrator. AiResearch estimates 110 lb/kw and a 32-ft diameter inflatable concentrator for their proposed 5-kw Brayton cycle unit.

Owing to the diversity of design approaches proposed for inflatable rigidized solar concentrators, it is difficult to assess the structural integrity of these units. The extent to which additional structure is required to prevent deleterious concentrator distortion must be studied for the most promising reflectors. On the basis of current funding and flight test planning, the SNAP systems enjoy considerably higher development priorities and funding than the various solar/dynamic power plant systems.

Auxiliary/Emergency Power. An attempt has been made to define auxiliary and/or emergency power requirements. Owing to the unique nature of the mission, hypothesis of an early primary powerplant failure results in auxiliary powerplant performance criteria virtually identical to the primary plant, suggesting 100% redundancy.

However, definition of emergency powerplant criteria hinges upon consideration of chronological order and duration of an "emergency" during which primary power is interrupted. With the Gemini hydrogen-oxygen fuel cells nearing operational status and the Apollo-LEM systems in an advanced state of hardware development, these systems are obvious candidates for auxiliary/emergency powerplant concepts. Advanced hydrogen-oxygen fuel cells with cryogenic reactant storage are estimated to have fixed weights of approximately 8 to 12 lb/kw and reactant consumption of 1.4 lb/kw-hr where hydrogen demand is 0.1 lb/kw-hr, oxygen demand 0.8 lb/kw-hr and cryogenic tankage, insulation and controls comprising the remaining 0.5 lb/kw-hr. Although direct electrochemical conversion utilizing storage liquid propellants (fuels and oxidizers) is theoretically possible, the majority of development effort to date has been devoted to hydrogen-oxygen activated systems.

Since storable liquid propellants are leading candidates for spacecraft attitude control, midcourse trajectory correction and spin/de-spin propulsion, the possibility of combining auxiliary/emergency powerplant reactant and onboard propellant tankage is attractive. Recent developments in reciprocating engines have been reviewed. Compression ignition systems operating on storable liquid propellants are expected to achieve fixed weights of 12-14 lb/kw and fuel/oxidant consumption rates as low as 1.8 lb/kw-hr, 0.7 lb/kw-hr of which is tankage.

Spacecraft Integration. The energy sources associated with the leading powerplant candidates pose radically different spacecraft integration problems which defy direct comparison. The primary powerplant energy source candidates are limited to solar and nuclear systems. The sole auxiliary powerplant energy source proposed is chemical. The spacecraft integration constraints inherent in primary and auxiliary powerplant energy source and conversion systems were examined. Resulting configurations for Solar/Brayton and Nuclear/Brayton power systems are shown in Figs. 14 and 15.

Operational Constraints. A tentative sequence of events for the flyby mission, as related to powerplant selection, was outlined to isolate operational constraints inherent in the various candidate systems.

Conclusions and Recommendations. The conclusions reached during the course of interplanetary flyby spacecraft powerplant selection studies were necessarily based upon current programs and projected advances in technology. The electrical power systems under consideration represent a diversity of development priorities, funding levels, development risks, schedules, man-rating, flight testing, etc.

CONCLUSIONS

1. Nuclear power systems are currently the leading candidates for EMPIRE spacecraft. 2. No firm requirement for auxiliary power has been established in conjunction with the nuclear powerplants. 3. There are weight penalties inherent in the nuclear powerplant. 4. There are several operational constraints inherent in solar/dynamic powerplants. 5. Multiple solar/dynamic systems may be required. 6. An auxiliary powerplant is required in conjunction with the solar/dynamic primary powerplant for Earth orbit escape.

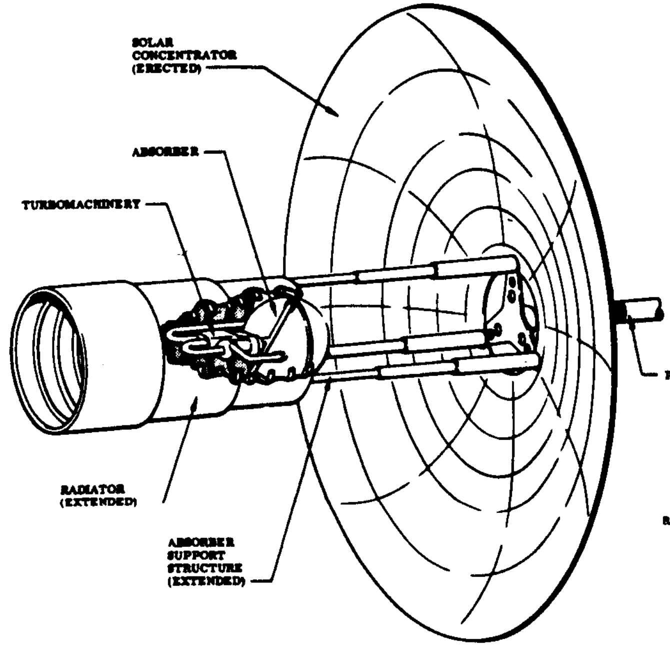


Fig. 14 Solar/Brayton Power System

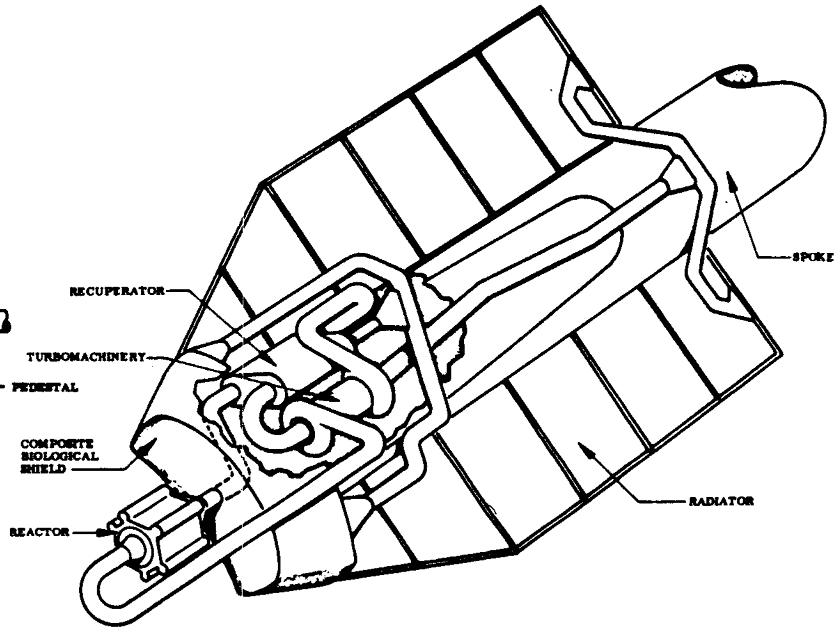


Fig. 15 Nuclear/Brayton Power System

RECOMMENDATIONS

Several recommendations relative to the particular powerplants selected and power requirements in general are offered. 1. More detailed powerplant selection studies should be carried out. 2. A solar/dynamic reference design should be established and more detailed trade offs performed with nuclear reactor/dynamic systems. 3. Inflatable/rigidized solar concentrator development efforts should be reviewed to determine redirection required to assure structural integrity of concentrators. 4. Detailed investigations should be undertaken to determine the feasibility of developing a single solar/dynamic powerplant providing 5-8 kw with incident solar flux variations from 52 to 240 watts/ft^2.

Section 7
SPACECRAFT DESIGN

Introduction. Two detailed spacecraft configurations were designed for the interplanetary flyby mission, one employs a solar concentrator heat source for its electric power system, the other a nuclear heat source. The first configuration, which is a modification of the spacecraft described in the initial EMPIRE study report, incorporates a larger, more spacious solar shelter in the hub and provides for a heavier Earth entry system; the second configuration is a more detailed study of a spacecraft mentioned briefly in the initial report. Both configurations rotate about a central hub to provide a gravity field for their three-man crew and are designed for planetary flyby missions to be accomplished within current state-of-the-art or with projected Apollo technology.

Earth Entry Module. For the 1974-75 period, Earth entry speed is 14.02 km/sec (46,000 fps) for the Venus mission and 14.63 km/sec (48,000 fps) for the Mars mission. One of the study ground rules is that the Apollo module is to be used as the final Earth entry module. Since it is capable of only 11.00 km/sec a rocket braking system has to be added to provide the initial ΔV, bringing the Apollo to its design speed. This arrangement, however, does not produce an optimum weight system. For this study, three entry systems were considered, (a) the present Apollo plus a retro-propulsion system, (b) a modified Apollo capable of a higher entry speed plus a retro-propulsion system, and (c) a new high-speed direct entry capsule.

System (a) requires only a minimum Apollo modification but the development of the largest retro-module. This is the heaviest of the three systems considered and would be used if the Apollo heat shield were not to be modified.

System (b) is a weight optimization of system (a). The initial Apollo weight is 4,450 kg (10,000 lb) for an 11.00 km/sec (36,000 fps) entry speed,but with the addition of heat shield mass, insulation, and structure this entry speed can be increased. This reduces

the retro-system ΔV requirements and therefore its mass. The sum of these two masses is the weight of the entry-system and is shown plotted against the Apollo design speed in Fig. 16. The minimum system weight occurs for both the Venus and Mars mission when the Apollo ΔV is about 11.90 km/sec (39,000 fps) with the retro system providing the initial impulse. Thus, by optimizing these ΔV's it is possible to save 2,720 kg (6,000 lb) in the Venus spacecraft and 4,536 kg (10,000 lb) in the Mars spacecraft. For this reason, system (b) has been selected as the entry system for these spacecraft.

The third possibility, system (c), is the new NASA/MSC direct-entry body. This is the lightest entry capsule system, but it is only in the design-concept stage and may not be readily adaptable as a command/control center for the interplanetary spacecraft. This body weighs about 6,800 kg (15,000 lb) for an entry speed of 19.81 km/sec (65,000 fps) and its effect on the basic configuration is discussed in the sections pertaining to specific configurations.

Major Spacecraft Components. The spacecraft consists of the following major components; command module, mission module, power system, and the midcourse propulsion system. The command module for the spacecraft with the solar power system is a modified Apollo module with an internal volume of about 8.5 m^3 (300 ft^3). It serves as the crew's launch vehicle, as the emergency escape vehicle during launch, as the Earth entry vehicle during an emergency entry from Earth orbit, as the command/control center during the interplanetary mission, and as the Earth entry vehicle at the completion of the mission. For one configuration using the nuclear power system, the command module is a special purpose module which integrates the command/control center with the solar shelter. For the other configuration the command module is similar to that for the solar heat source configuration.

The mission module houses the crew's living quarters, the dining and recreation area as well as the environmental control equipment, food, water, and the spares for the spacecraft. Internal volume of the module is 113 m^3 (4,000 ft^3).

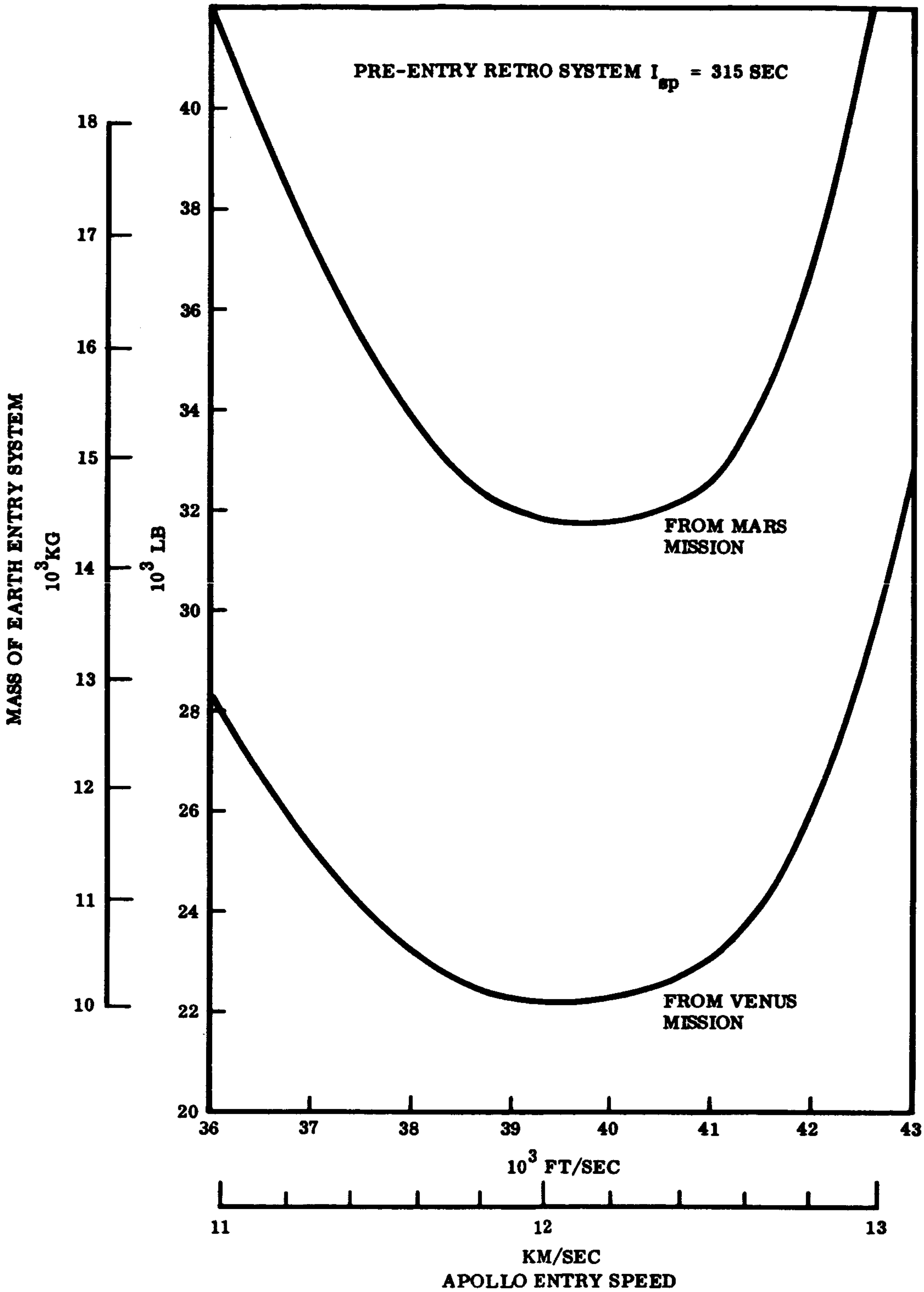


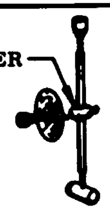
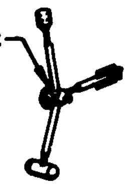
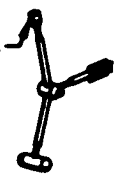
Fig. 16 Total Entry System Mass Vs. Apollo Entry Speed

Spacecraft Configuration: Solar Power System. The rotating spacecraft consists basically of two modules, the "command" and the "mission" modules connected by rigid structure, spinning about a central hub (see Fig. 17). The centrifugal force produced by this rotation acts outward radially and the crew within these modules experience this force as gravity. This configuration is basically the same as that shown in the initial Empire report except for changes in two areas, the command module and the solar shelter. The command module and its retro-system are heavier to account for the entry speeds, the shelter is larger to give the crew more living room.

In this configuration the retro-propulsion module remains attached to the Apollo command module throughout the entire mission, their combined weights counter-balance that of the mission module. Both modules are always occupied, one crewman is in the command module at the primary duty station, the other two are in the mission module. The heavier command module is mounted 25 m (82 ft) from the spin axis on the hub centerline and at the 0.4 rad/sec spin rate has a 0.4 gravity field. Balancing this, the lighter mission module is mounted at a distance proportional to its weight. The minimum 25-m spin radius and the 0.4-rad/sec (3.84-rpm) spin rate have been selected as being within the "comfort-zone" for rotating spacecraft based on the study of artificial gravity in rotating vehicles by B. J. Loret.

A new body such as the NASA/MSC direct-entry capsule could be incorporated into this same basic configuration to serve as the command module. This new module would be attached to a small "supply-module" and these two modules would be connected by rigid structure to the mission module at the opposite side of the central hub. Internal equipment would be distributed between the mission and supply modules to equalize the mass balance on both sides of the hub.

A second major change in the spacecraft is in the solar shelter. Initially the shelter had an enclosed volume of 1.9 m^3 (67 ft^3) but the human engineers consider this is too confining for the three man crew especially since their entire activity is restricted to a relatively small spacecraft for an extended period of time. Information compiled on confinement volumes relative to periods of confinement within civil defense shelters, deep sea vessels, high altitude gondolas, and ground test chambers cannot be used to establish a specific volume for this shelter.

	VENUS		MARS		
	.001†	.01†	.001†	.01†	
SOLAR SHELTER  SOLAR POWER SYSTEM	37,838 kg 83,420 lb	35,706 kg 78,720 lb	48,278 kg 106,440 lb	46,055 kg 101,540 lb	MODIFIED APOLLO
	44,770 kg 98,700 lb	42,412 kg 93,500 lb	56,774 kg 125,161 lb	55,407 kg 122,149 lb	REGULAR APOLLO
SOLAR SHELTER  NUCLEAR POWER SYSTEM	41,593 kg 91,700 lb	39,665 kg 87,450 lb	50,691 kg 111,750 lb	48,513 kg 106,950 lb	MODIFIED APOLLO
	48,848 kg 107,690 lb	46,735 kg 103,033 lb	60,789 kg 134,014 lb	58,397 kg 128,740 lb	REGULAR APOLLO
SOLAR SHELTER  NUCLEAR POWER SYSTEM	43,502 kg 95,910 lb	41,370 kg 91,210 lb	53,170 kg 117,220 lb	49,511 kg 109,150 lb	MODIFIED APOLLO
	52,145 kg 114,960 lb	48,582 kg 107,104 lb	63,524 kg 140,043 lb	60,042 kg 132,367 lb	REGULAR APOLLO
DIRECT ENTRY BODY, SOLAR SHELTER  NUCLEAR POWER SYSTEM	39,510 kg 87,110 lb	37,514 kg 82,710 lb	44,054 kg 97,120 lb	40,789 kg 89,920 lb	

† PROBABILITY OF RECEIVING MORE THAN 200 RADS TO THE BLOOD FORMING ORGANS

Fig. 17 Injection Payload

The crew within the interplanetary spacecraft is in a very unusual situation. First they are confined for 300 to 600 days in a 7000 to 8000 ft^3 pressure vessel in which they experience a variable g-field as they traverse the hub. Next, there is no immediate abort capability where they can initiate a retro-maneuver to be back on Earth in a few hours. Finally, during the solar flares the crew has to confine itself further within the local storm shelter. Psychologically, this "confinement within confinement" will be difficult to duplicate on Earth and the specific volumes for the shelter can only be estimated.

The larger shelter is compatible with the basic configuration and the booster capability. It is cylindrical; its thick wall forms a major section of the hub itself. Internal volume is about 5.6 m^3 (200 ft^3). A smaller "cage" assembly is mounted within this massive shelter and the crew sits around the sides of this cage 120 degrees apart facing the center. During a solar flare, the crew enters the shelter, closes the hatches, and activates the motor driving the cage in the direction opposite that of the spacecraft, and at the same rate. The net effect is a stationary shelter in inertial space, with the crew in a zero-g field. The cage roller system is adjustable so that its centerline can be aligned with the actual spin axis of the spacecraft. Primary purpose of the cage is to permit the crew to live and work in the hub in a zero-g field during a solar flare without having to despin the entire spacecraft. It also encloses the crew and eliminates the visual cue of walls rotating around them in the zero-g field.

Shielding requirements for this shelter are discussed in Section 3 and are summarized in Table 5. The shelter is made of polyethelene with a thin aluminum shell, rather than solid aluminum as formerly shown, to save about 35 percent in structural weight. The original Empire weights were based on more conservative estimates; 70 gm/cm^2 for Venus, and 56 gm/cm^2 for Mars. For this reason it has been possible to increase the shelter volume by 300 percent while decreasing the weight by 250 to 300 percent.

All systems within the spacecraft would be checked out in Earth orbit before proceeding with the interplanetary phase. Power during this checkout phase is supplied by the auxiliary system, either an internal combustion engine which uses storable propellants from the midcourse propulsion system or by fuel cells. This is discussed more fully

in Section 6. The solar concentrator is not erected in Earth orbit because this large diameter reflector cannot withstand the escape loads during the injection phase.

Table 5

SHIELDING REQUIREMENTS

	Probability of Receiving More Than 200 Rads to the Blood Forming Organs	
	.001	.01
Venus	24 gm/cm^2 alum or 2 gm/cm^2 alum + 13 gm/cm^2 poly	12.5 gm/cm^2 alum or 2 gm/cm^2 alum + 6.3 gm/cm^2 poly
Mars	32 gm/cm^2 alum or 2 gm/cm^2 alum + 18 gm/cm^2 poly	17 gm/cm^2 alum or 2 gm/cm^2 alum + 9 gm/cm^2 poly

Spacecraft Configuration: Nuclear Power Systems. Basically the modular arrangement is the same as that of the solar powered spacecraft except that an additional third spoke supporting the nuclear power system and radiation shield is added to the spacecraft in the plane of rotation.

The nuclear power system with its radiation shield is dynamically balanced by the two manned modules so that the center of mass and spin axis lie on the hub centerline. A change in the distance of the power system from the hub or in the shielding mass requires only a small displacement of the two manned modules from their present position. During a solar flare the crew moves into the shelter in the hub and exercises control from this module. The cage within the shelter permits the crew to operate from the shelter without despinning the entire spacecraft.

A variation of the above configuration was examined. The object in this study was to generate a configuration which could use one of several entry systems, e.g., the Apollo command module or the direct-entry capsule without a major change in the basic configuration, and one which would incorporate the command/control center with the solar shelter. The spacecraft is similar to that described above except that the entry body is docked at the side of the hub opposite the nuclear power system. The power system

and its shield are balanced by the entry body plus the other two manned modules so that the center of mass and the spin axis lie on the hub centerline. A change in the mass of the entry body requires a small displacement of two manned modules from their present position.

The second change is that the solar shelter serves as the command center. One crewman is always stationed here at the primary duty station and then during a flare he is joined by the other two crewman. In this way the command/control function is undisturbed throughout the mission. Internal volume is 10.5 m^3 (370 ft^3) which is larger than the 8.5-m^3 Apollo module and since it would serve specifically as the command center for the interplanetary mission it could be designed for maximum mission effectiveness.

The Earth-escape booster payload weights for the four configurations are shown in Fig. 17. Two entry columns are listed for each configuration for two probabilities of receiving more than 200 rads to the blood-forming organs. For those spacecraft which use the Apollo module, weights are listed for the standard and the modified Apollo modules.

Section 8

ORBIT LAUNCH VEHICLES

This section reports on a first in-depth iteration made to ascertain with greater confidence the applicability of future or possible Saturn-class orbit and ground launch boosters to interplanetary missions. Earth-to-orbit trajectory studies of the Saturn V launch booster were combined with orbit launch booster performance calculations and analysis of probable orbit operations to obtain a detailed mass breakdown for various launch modes.

Study Method. The orbit launch boosters were limited to combinations of four possible Saturn-class stages. The pertinent characteristics of these are given in Table 6.

Table 6

ORBIT LAUNCH BOOSTER STAGES

Name	Engine	Propellant	Propellant Capacity lb	Propellant Capacity kg
S-IVB	J-2	LOX/LH_2	230,000	104,000
S-NA1	Mod 1	LH_2	80,000 to 140,000	36,300 to 63,500
S-NA2	Mod 1	LH_2	>140,000	63,500
S-NB	Mod 2	LH_2	Any	Any

The basic ground rules and assumptions used in the study:

- Launch to orbit was assumed to be accomplished by the Saturn V booster vehicle only. This two-stage vehicle was defined by the C-5 Launch Vehicle Design Data Book – Revision h (IN-P & VE-V-62-6)
- S-IVB mass and J-2 engine characteristics were based upon data in the C-5 Launch Vehicle Design Book.

- Saturn Nuclear (S-N) vehicle masses were based upon estimates from RIFT sizing studies at LMSC and from the Nuclear Lunar Logistics Study (NAS 8-5600) conducted by LMSC for MSFC. The masses used were generally more conservative than those reported in the lunar study to allow for additional shielding, insulation, and weight growth. Close cooperation and liaison with the RIFT Follow-On Applications Group at LMSC was maintained throughout the study to assure prompt and accurate input of pertinent data.
- Nuclear engine design data were obtained from the input of the REON Division, Aerojet General Corporation, to the Nuclear Lunar Logistics Study.
- No more than two Saturn V launches, with one rendezvous in orbit, were considered.
- Only simple mating orbital rendezvous was considered (i.e., no propellant transfer). The MSFC study on orbit rendezvous (Ref. Earth Orbital Operations Tanking Mode, MTP-CD-62-1, June 1962) provided data on rendezvous and mating operations and their mass requirements.
- The orbital altitude was selected at 250 nm. This altitude was chosen because, for a 30-degree orbit inclination, a vehicle in orbit is in approximately the same point in its orbit each day that Cape Kennedy comes into the orbit plane. Thus, a launch opportunity is assured each day while avoiding dog-leg launches, elaborate phasing orbit operations, or other techniques that would be necessary if other altitudes were used.
- No elaborate splitting of the interplanetary payload was desired. It was considered ideal to have the payload placed into orbit in one piece. However, some splitting of the payload appears mandatory for many of the launch configurations to obtain reasonable performance output.

Study Results. The results of this study are summarized in Table 7. The injected payloads shown can be considered as the approximate maximum for each configuration. Only a more detailed look, with better weights for the design concepts involved, can ascertain more accurate payload numbers. Several modifications to current design concepts were found to be required. Because of the long stay time in orbit, the S-IVB and all nuclear stages must be provided with additional insulation. Figure 18 shows a possible concept for the S-N stages involving a honeycomb structure, filled with helium prior to tanking and covered with super insulation, inserted in the space between the longitudinal stiffeners. Figures 19 and 20 show the design modifications to the S-IC

Table 7

ORBIT LAUNCH VEHICLE PERFORMANCE COMPARISON
(Limited to Orbital Rendezvous of Two Saturn V Booster Payloads)

OLV CONFIGURATION		MARS FLYBY (V_∞ = .20 EMOS)			VENUS FLYBY (V_∞ = .15 EMOS)		
FIRST STAGE	SECOND STAGE	1ST STAGE IMPULSE PROPELLANT	2ND STAGE IMPULSE PROPELLANT	INJECTION PAYLOAD	1ST STAGE IMPULSE PROPELLANT	2ND STAGE IMPULSE PROPELLANT	INJECTION PAYLOAD
S-IV B	–	180,000* (81,600)*	–	53,000 (24,000)	180,000* (81,600)*	–	71,000 (32,200)
S-NA 2	–	154,000* (69,900)*	–	100,000 (45,400)	154,000* (69,900)*	–	141,000 (64,000)
S-NB	–	154,000* (69,900)*	–	111,000 (50,400)	154,000* (69,900)*	–	153,000 (69,500)
S-NA 1	S-NA 1	97,000 (44,000)	97,000 (44,000)	121,000 (55,000)	86,000 (39,000)	86,000 (39,000)	143,000 (64,900)
S-NA 1	S-NA 2	47,000 (21,300)	144,000* (65,300)*	120,000 (54,400)	–	–	–
S-NA 2	S-NA 1	144,000* (65,300)*	54,000 (24,500)	113,000 (51,200)	144,000* (65,300)*	36,000 (16,300)	131,000 (59,400)
S-IV B	S-IV B	104,000 (47,200)	180,000* (81,600)*	98,000 (44,400)	100,000 (45,400)	162,000 (73,500)	120,000 (54,400)
S-IV B	S-NA 1	100,000 (45,400)	133,000 (60,400)	126,000 (57,200)	–	–	–
S-NA 1	S-IV B	123,000 (55,800)	133,000* (60,400)*	106,000 (48,100)	–	–	–

Masses given in pounds; kilogram equivalent in parentheses.
***Propellant loading limits the capability of this configuration.**

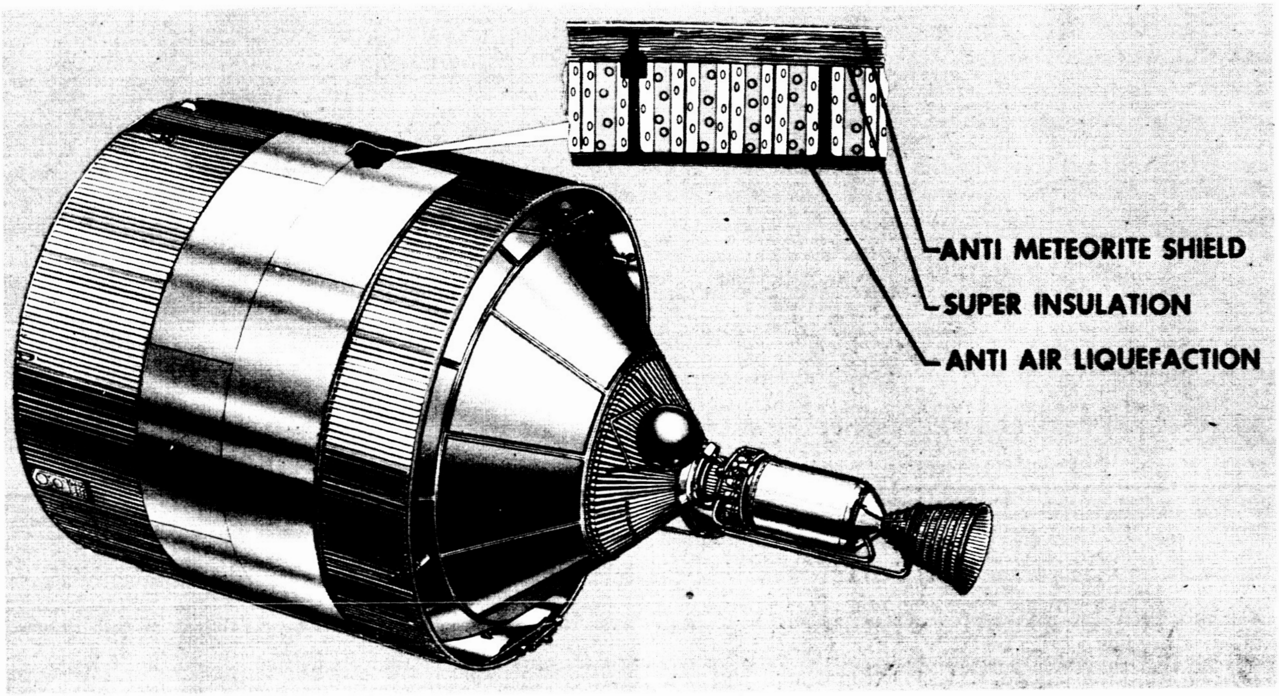


Fig. 18 S-N Stage Insulation Configuration

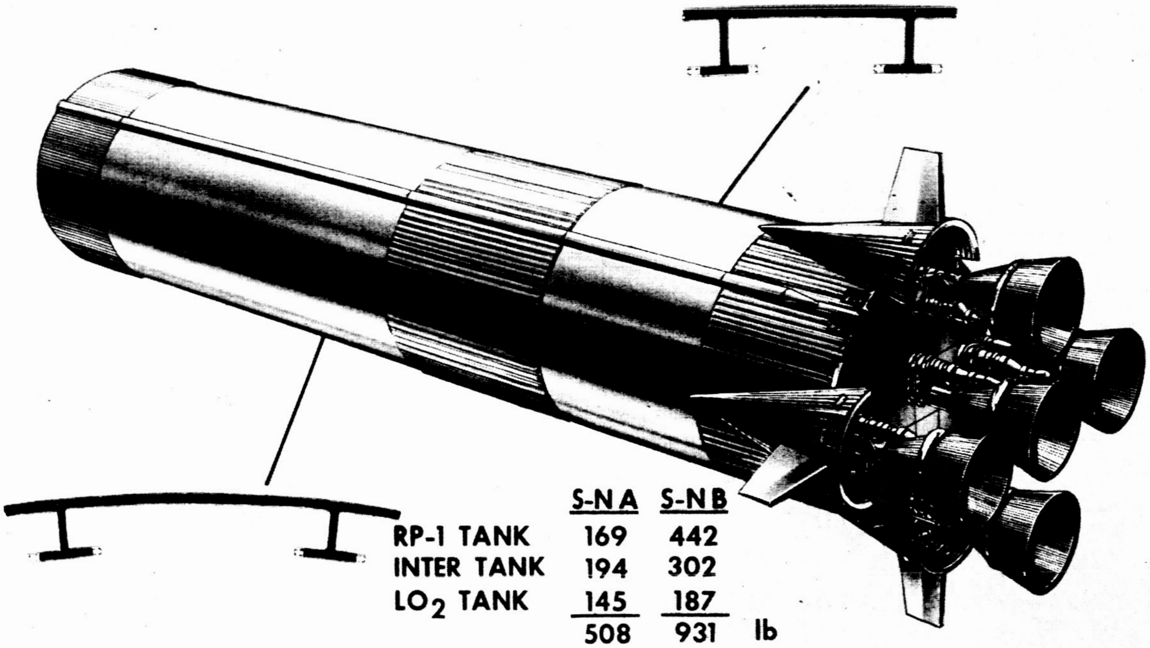


Fig. 19 S-IC Modifications

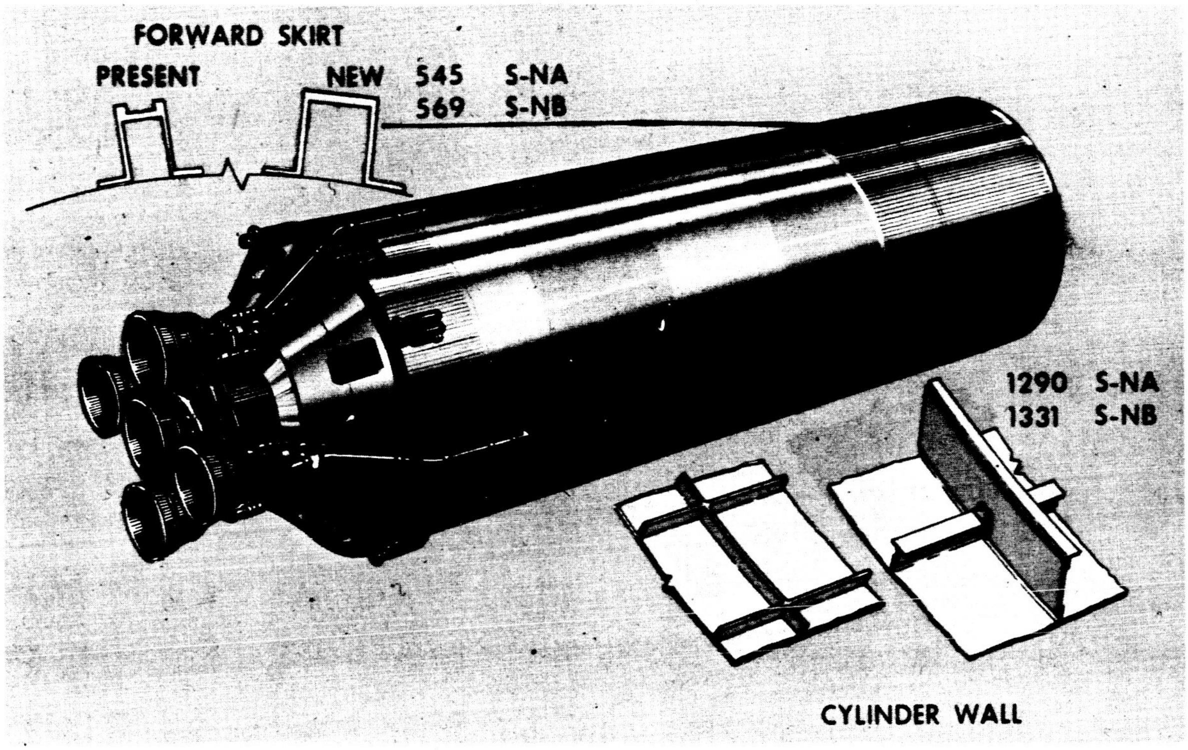


Fig. 20 S-II Stage Modifications

and S-II stages found by the RIFT Follow-On Applications Group to be required in the orbiting of nuclear stages.

For the S-IC, it is suggested that the "T" stiffeners be left at their full 1.8-in. width rather than be machined down to 1.3 in. The intertank structure requires some small gage changes for stiffening. The net weight increases shown include a 10 percent design reserve. For the S-II, changing the present extrusion of the longitudinal stringer in the forward skirt to the configuration used in the aft skirt and adding a "T"-shaped extrusion to the machined longitudinal stiffeners through the present rings and baffles will add the needed strength.

The S-II was also analyzed to determine the possibilities of its use as an orbit launch booster. The brief examination indicates that, with certain modifications, the two-stage Saturn V booster could orbit an S-II third stage loaded with liquid hydrogen but with empty LOX tanks. The modifications that appear desirable are summarized in Fig. 21.

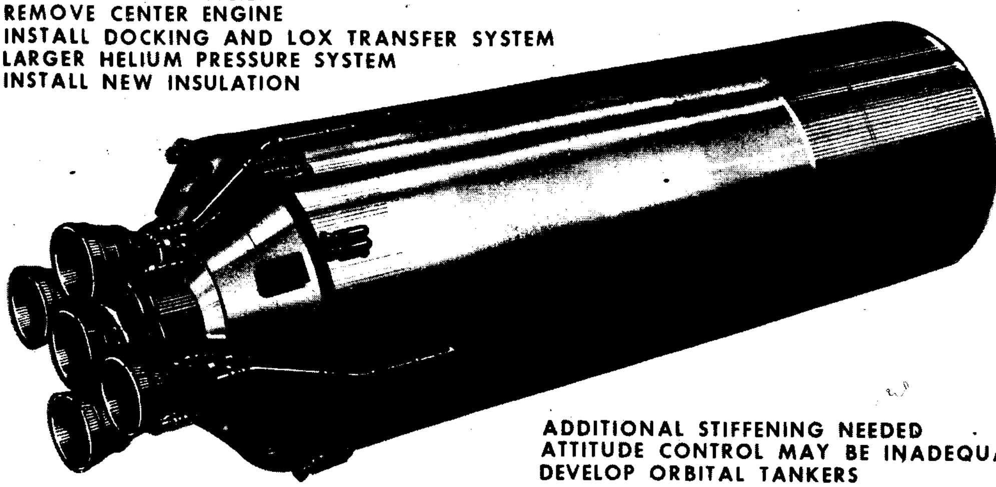


Fig. 21 Modifications to Make S-II Into Orbit Launch Vehicle

Tentative Study Conclusions. This detailed look at the applicability of various Saturn-class booster stages and launch vehicles raises more problems than it resolves. At least another two iterations between design concepts and performance calculations are needed before any real determination can be made. However, on the basis of this work (Table 7) and the information presented in Section 7, certain tentative conclusions follow:

- A manned Venus flyby, with one launch of a Saturn V/N (Saturn V with a nuclear upper stage) cannot be completely ruled out. Recent reductions in estimated shielding requirements (see Section 7) have lowered the masses of some spacecraft concepts below 39,000 kg (86,000 lb). As long as these low estimates hold, a single launch flyby appears possible.
- The Venus flyby appears definitely possible with two Saturn V launches (one rendezvous). This can be launched from orbit two ways: (1) staging of either two S-IVB's, two nuclear stages, or a S-IVB/nuclear combination; or (2) one large nuclear stage (S-NA2 or S-NB).
- The LOX/LH_2 propellant required for the Venus flyby mission is beyond that amount that can be orbited inside an S-IVB by the Saturn V. Therefore, a flyby initiated with one S-IVB is not possible unless propellant is added through refueling, additional tanks, or another suitable propulsion stage.

- The ability to perform the Mars low energy flyby with two Saturn V launches to orbit cannot be resolved at this time. Some of the conservative ground rules and assumptions under which this study was conducted (e.g., Apollo-type, Earth entry system, large midcourse propulsion system, no on-orbit refueling, no Saturn V performance increase) leave little margin in performance capability.
- Three two-stage booster configurations show the highest performance: S-NA1/S-NA1, S-NA1/S-NA2, and S-IVB/S-NA1. Substitution of a Mod 2 or better engine would provide even greater capability. However, the S-NA1/S-NA1 requires close to a 50-50 split in the payload launched to orbit – which would greatly complicate the rendezvous operation and systems. The S-NA1/S-NA2 requires two different nuclear stage sizes. The S-IVB/S-NA1 achieves high performance because the thrust-to-weight ratio for each stage is sufficient to reduce gravity loss to a minimum. However, it presents configuration problems arising from a first stage of smaller diameter than the second stage. Finally, no detailed analysis has beeen performed to support the adequacy of the nuclear engine shielding assumptions.
- The Saturn V appears incapable of launching a single nuclear stage into orbit with sufficient propellant to accomplish the Mars low energy flyby. Thus, either the payload mass must be reduced or some other configuration or operation change is needed.

There are, of course, numerous suggestions of ways to overcome the discrepancy between payload requirement and booster capability:

- Three Saturn V launches, with two of these orbiting S-NB nuclear stages and a third orbiting the intact payload, virtually guarantees flyby capability past Mars. This is probably one of the simplest and least costly methods with the greatest chance of success. This two-stage escape system could deliver over 170,000 lb (77,000 kg) payload to a hyperbolic excess speed of .20 EMOS (the nominal low energy flyby requirement) and over 130,000 lb (59,000 kg) to .25 EMOS (the maximum expected requirement).

- Uprating the Saturn V capability could have a decisive effect in requiring only one orbital rendezvous for the Mars mission. To assure success, an increase of over 10,000 lb (over 4,500 kg) delivered payload per launch would be a minimum requirement.
- A high-speed aerodynamic entry body would greatly reduce the entry system mass on the payload. An LMSC concept being studied under contract with MSC (NAS 9-1702) could reduce the required mass of the heaviest payload configuration to less than 110,000 lb (50,000 kg) — well within the capability of several booster configurations shown in Table 7.
- Orbital refueling appears as a means to overcome the inability of the Saturn V to orbit a booster with sufficient propellant to perform the Mars flyby. Indications are that such a system could deliver 170,000 to 180,000 lb (77,000 to 82,000 kg) of liquid hydrogen to orbit. This amount, when injected into an already partially loaded S-NB, would be sufficient propellant to launch about 130,000 lb onto the flyby mission.
- Modular tankage, coupled on orbit, is another possible scheme to provide adequate propellant.
- Suborbit start of the nuclear stage would increase the orbital payload delivery of the Saturn V an amount to provide a nuclear booster of sufficient size to perform the mission. A means would be needed to overcome the radiation hazard of the "hot" nuclear engine during rendezvous.

These are all additional items requiring investigation before any firm commitment could be made on a configuration or on required launch and orbital operations.

Section 9
SCHEDULE OF OPERATIONAL EVENTS

Because of the interlocking nature of the various events in a flyby mission, it is instructive to delineate the sequence of activities involving the general and major operations related to the spacecraft. This was studied briefly and a schedule of operational events was made to organize the salient flyby mission milestones on a general time scale.

Model Vehicle System and Variations. The various vehicle designs being considered for the manned flyby mission are discussed in Section 7, Spacecraft Design. For establishing the general sequence of events, a model transportation system was selected. This consists of the spacecraft and a two-stage escape booster utilizing as one of the stages a nuclear propulsion system. A nuclear power system is used to provide primary electric power for the spacecraft. The Earth entry system is a modified Apollo using retro-rockets. A separate command module is provided for the spacecraft control functions and is integrated with a solar flare shelter. Because of the radiation emanating from the nuclear propulsion system, the crew must be located near the central hub during the escape phase. This is accomplished by using the Apollo module.

The schedule of operational events is based on the Mars flyby mission. Because of the general approach utilized here, the sequencing is believed applicable also to the Venus mission.

Schedule of Operational Events. Taken in chronological order, the overall operations quite naturally group themselves into the near Earth operations, the outbound midcourse phase, the planetary operations, the homebound midcourse phase and the Earth entry phase. These in turn can be separated into major headings, each of which contain the actual events. These are depicted in Figs. 22 and 23, the outbound and homebound legs respectively.

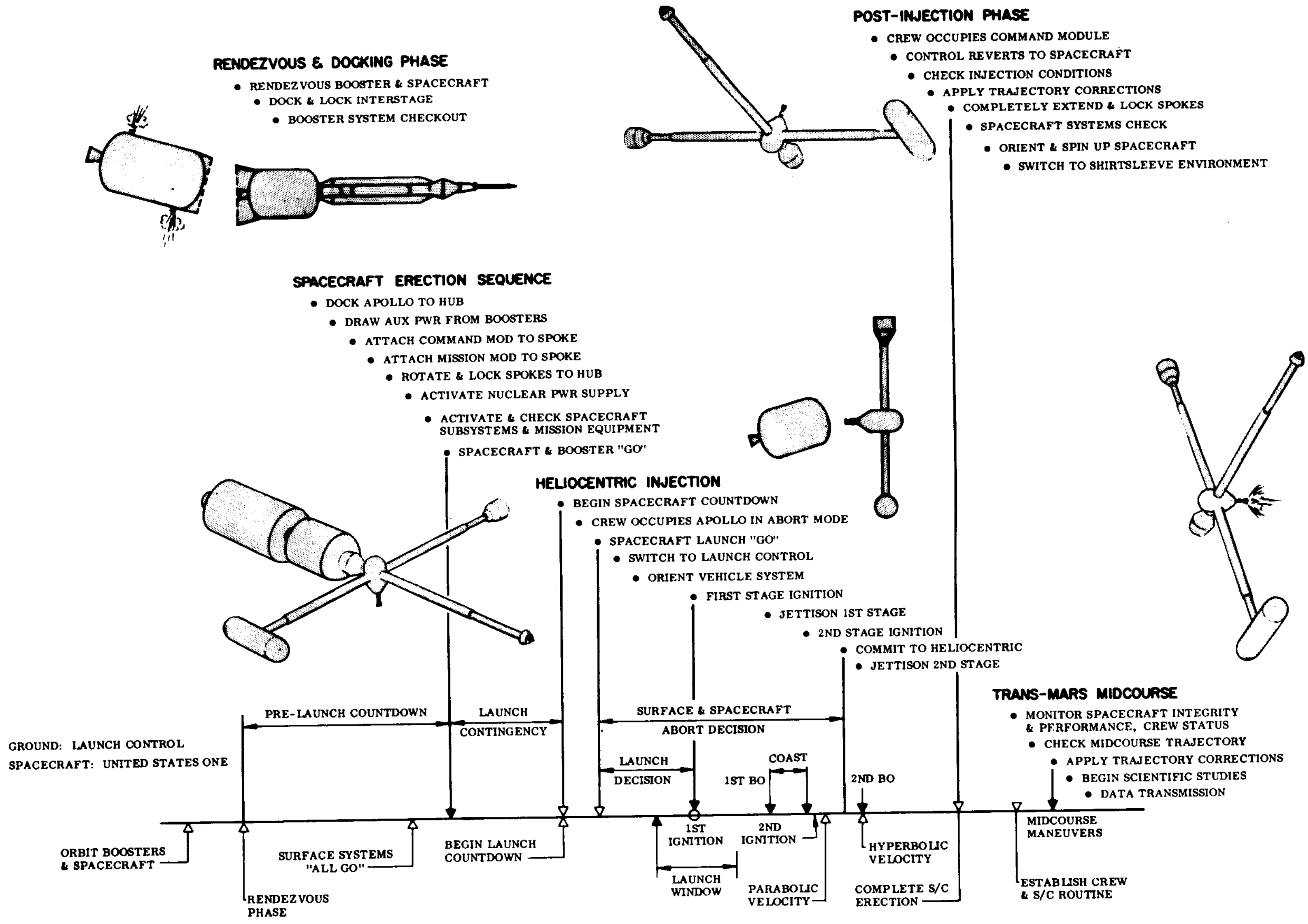


Fig. 22 Mars Flyby Schedule of Operational Events, Earth to Mars Leg

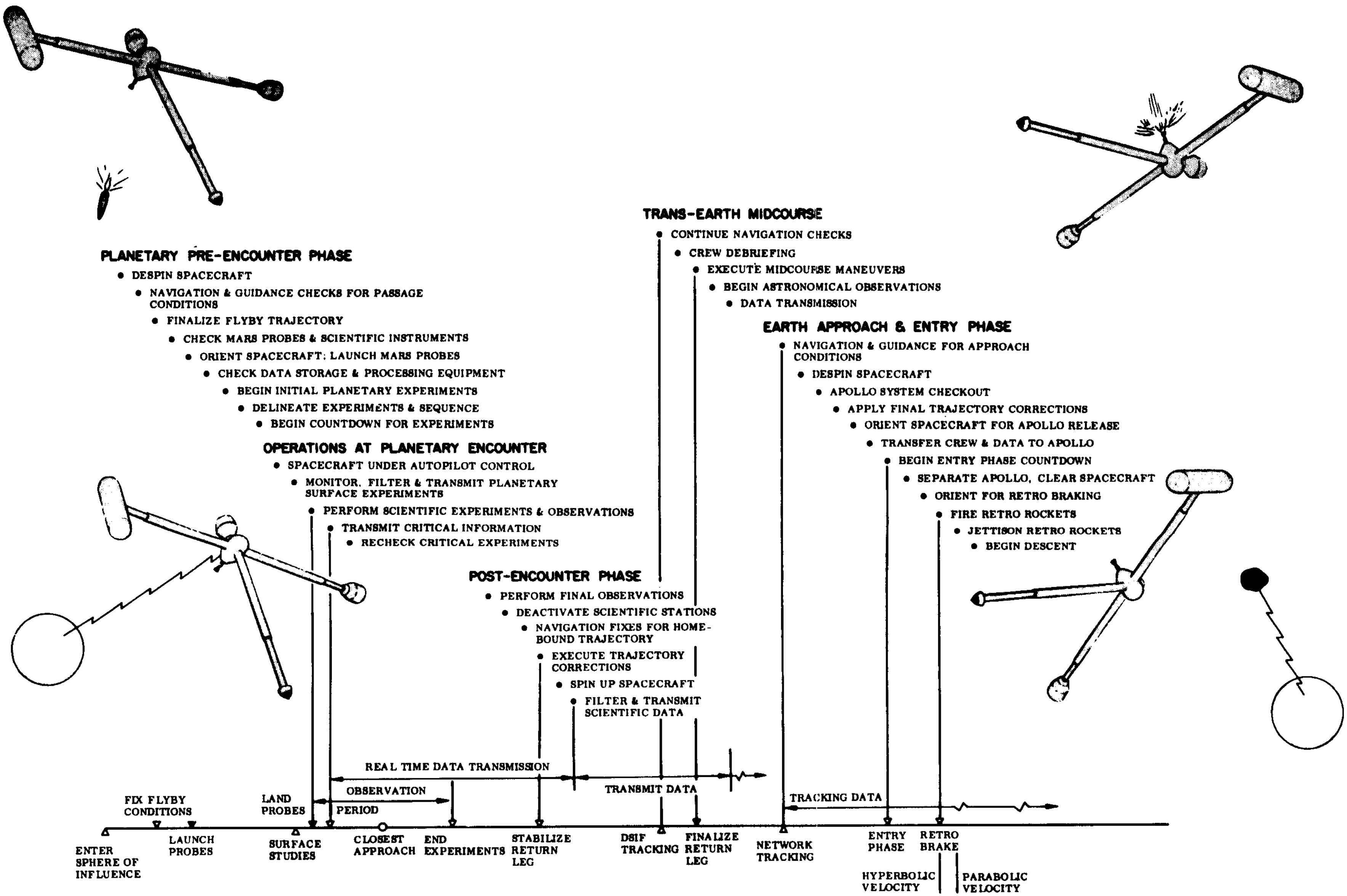


Fig. 23 Mars Flyby Schedule of Operational Events, Mars to Earth Leg

Rendezvous Docking Phase. Throughout the rendezvous phase the crew remains in the module. The actual joining and locking of the interstages will be accomplished through remote television and control. If necessary one crewman may be given the capability to perform low level external tasks such as securing the locks, connecting electrical cables and making visual checks.

In the sequence shown the spacecraft and a booster is to rendezvous with a second booster, each item having been placed on orbit separately. Surface launch of the spacecraft and escape stage is initiated with proper phasing to permit rendezvous of the two pieces. At this point the Rendezvous and Docking Phase begins coincident with the prelaunch countdown. After locking is accomplished the entire booster system is given a complete verifying checkout to determine any loss in booster capability which may have occurred during the surface launch.

Spacecraft Erection Sequence. The detailed erection sequence is explained in Section 7. Power for moving the mission module, spokes, and command module is drawn from auxiliary power packs stored in the boosters. The startup procedure of the nuclear power supply will involve coordination of power output control and readiness of the spacecraft internal subsystems to "on the line." Once the spacecraft is operating in the self-contained mode, the on-board mission equipment and planetary probes are subjected to system tests.

With this final check the prelaunch countdown comes to an end and the spacecraft and booster are declared "Go." Sometime prior to this announcement, all the surface systems (e.g., rescue and recovery groups, tracking, communications, etc.) have been brought to "All Go" status. The launch countdown for heliocentric injection now begins.

Heliocentric Injection. The actual countdown for launch will not necessarily commence at the "all systems go" point. Ideally this would be the case if both the spacecraft vehicle system and all surface systems reach their final status together. However, the scope of this undertaking will quite naturally involve holds for both; therefore, some period of launch contingency is provided (Fig. 22).

At the start of the heliocentric-injection phase, the spacecraft undergoes a relatively short countdown wherein it is made ready for the acceleration phase. The crew returns to the Apollo entry system and brings the module to abort status. This is necessary not only for abort but for protection from radiation of the nuclear propulsion system. The entire vehicle system is then pronounced ready for launch and command is taken over by Launch Control. The decision whether to launch or not is nominally indicated between "launch go" and ignition.

Precise thrust vector alignment and time of ignition is determined by the ground facilities. Launch Control orients the vehicle properly with verification by the crew. The countdown continues through engine bootstrap until main-stage occurs, at which point the vehicle is underway.

The crew monitors all the necessary data which could lead to an abort decision. Launch Control, with information from telemetry, spacecraft communications and network tracking also processes data for an abort decision. Either may terminate booster operation and abort the mission; however, it is expected that a prior emergency mode list will require several preliminary decisions to be made before a drastic measure such as booster cutoff is initiated.

If a mission abort is declared, the Apollo is disengaged from, and clears the spacecraft, and is oriented so that the retro-rockets may be fired to reduce the velocity to below parabolic speed. Such a reduction is desirable if immediate return to Earth is necessary. Note from Fig. 22 that the duration of an abort decision nominally lasts from the launch go point to when the vehicle is committed to the heliocentric orbit. This latter event is determined by the incremental velocity capability of the entry system retro-rockets to bring the velocity below escape.

Post-Injection Phase. After the final stage has been jettisoned, the crew deactivates the Apollo and transfers to the command module where spacecraft control is assumed. Precision determination of the injection conditions is provided by the surface tracking facilities in addition to frequent on-board navigation checks. The vehicle is oriented for thrust vector alignment and the initial trajectory correction is applied.

The erection of the spacecraft is completed by fully extending the spokes and securing the locks and seals. A spacecraft systems check prepares the vehicle for artificial gravity. The desired spin plane is selected and the spacecraft oriented and spun up. The crew verifies that the environmental control system is providing a shirtsleeve environment, then doffs spacesuits. After this last major event in the post-injection phase the crew establishes the normal work-rest cycles.

Trans-Mars Midcourse. There will be a continuous monitoring of the spacecraft integrity and performance and also of the crew's behavior and status. The entire flight is considered a scientific experiment in itself since information on the spacecraft and crew performance will be used to establish or modify design criteria for further manned missions.

Scientific studies are started once the spacecraft routine is established. The information from the experiments and observations are relayed to Earth.

Application of the trajectory corrections is the major event of the midcourse leg. Information again would be supplied by Earth DSIF tracking and by the on-board navigation system. Not more than two, possibly only one, major midcourse correction would be applied during this part of the flight because precision tracking information will be available.

Planetary Pre-Encounter Phase. At approximately a day before planetary encounter or near the point where the planet's sphere of influence is entered, the planetary pre-encounter phase begins, Fig. 23. From this point until after the closest approach is made, the crew will be under free fall for approximately three days.

Frequent navigation sightings are taken during the early portion of this phase in order to satisfy the desired flyby trajectory, i.e., passage distance and illumination conditions. System tests and checkout are performed on the scientific instruments and in particular on the unmanned planetary surface probes.

The timing of probe launch and choice of landing site must allow the maximum viewing time between the probe and the spacecraft. The priority of experiments and observations

is established once the instrument tests and checks have been performed. With the start of the countdown for experiments, the crew assumes scientific stations and prepares to execute the operations at planetary encounter.

Operations at Planetary Encounter. The actual useful observation period expected from the planetary flyby missions insofar as the instrumentation is concerned would be not more than about two hours. During this brief period the crew spends all of their time performing experiments and observations in addition to monitoring and controlling the activities and results of the surface probes.

Because of the volume of information to be expected, a priority schedule is set up to assure that the information critical to later planetary exploration is transmitted as soon as possible. Another source of information is the crew themselves. Their impressions and observations of any unusual characteristics should be recorded in addition to their comments in general. These recordings can be used to debrief the crew later and to make a log of their impressions and observations.

Post-Encounter Phase. During this portion of the flight, the major activity of the crew is navigation and guidance because no precise externally provided aids are available and further, because the perturbations of the encounter may not be accurately known. After the return leg is reasonably stabilized, the spacecraft is prepared for artificial gravity and spun up.

Now begins the continuous job of relaying the acquired scientific information to Earth. The most important experimental evidence is relayed to Earth first; or possibly preliminary information on all the experiments are transmitted initially. The results of secondary experiments are transmitted later along with supporting and cross checking information.

Trans-Earth Midcourse. The navigation checks are by now intermittent and the data relay is automated under manual control. Thus crew debriefing may commence. The purpose of the crew self-interrogation session is to derive and correlate additional planetary information as seen through the eyes of these trained observers. The primary aid will be the voice recordings of each crew member taken during the passage,

augmented by a specially prepared question list. These discussions and crew impressions and reflections are then summarized in a spacecraft log book.

After the debriefing and when data storage space becomes available, the crew initiates the astronomical observations and experiments scheduled for the trans-Earth phase. Later in the midcourse phase Earth's DSIF will be in contact with the spacecraft and will provide additional tracking and ephemeris data for finalizing the homebound leg.

Earth Approach and Entry Phase. Upon approach to Earth, the surface tracking network and on-board navigation system will provide the necessary information for determining the entry conditions and the spacecraft deactivation countdown sequence. As the approach conditions are finalized, the vehicle is despun and one crew member dons a spacesuit and enters the Apollo to activate and check the entry module subsystems.

The final velocity impulses are applied and the spacecraft oriented for release of the entry module. Control of the spacecraft orientation and midcourse propulsion systems is assumed by the Apollo while the remainder of the crew dons spacesuits and prepares to abandon the spacecraft. All necessary log books, data, records, and other important material are transferred to the entry module.

Any last orientation corrections and trajectory maneuvers will be executed by the spacecraft under control of the entry module. The module is disconnected from the parent vehicle. The spacecraft is moved slowly away from the module by use of the spacecraft spin-up and orientation system. Once clear of the spacecraft, the Apollo is oriented for retrorocket ignition. Upon ground command and with on-board backup, the retrorockets are fired and the module is slowed sufficiently to provide for its design entry speed.

Section 10
DEVELOPMENT PLANNING

The scope of the development planning activities was restricted to the consideration of early manned Venus flyby systems and the early manned Mars flyby systems, utilizing a multi-Earth-launch mode of operation and considering no new launch vehicle technology beyond the Saturn V program.

Planning objectives were centered on establishing development program requirements, preparing system development program plans, and identification of key development milestones and critical development problem areas.

The planning approach initially established basic premises and reviewed factors in related national space programs that are applicable to the development programs for the selected early manned Venus flyby systems and Mars flyby systems concepts. The significant aspects of related national space programs are:

- Launch vehicles
- Spacecraft
- Support systems
- Facilities and logistics

Master development program plans were prepared in the form of integrated development schedules. Plans were prepared and major features reviewed for early Venus flyby systems concepts utilizing an all chemical Saturn V propulsion with a multilaunch mode for early orbiting. One Venus flyby Earth escape would utilize multi S-IVB chemical propulsion while another approach for Venus flyby Earth orbit escape would utilize a single S-NA2 propulsion system. The Mars program plan for an Earth orbit escape system was assumed to use a multistage nuclear launch vehicle.

Key development milestones for the 1975 Mars flyby are shown in Fig. 24 with a more detailed data on the spacecraft and nuclear stages in Fig. 25.

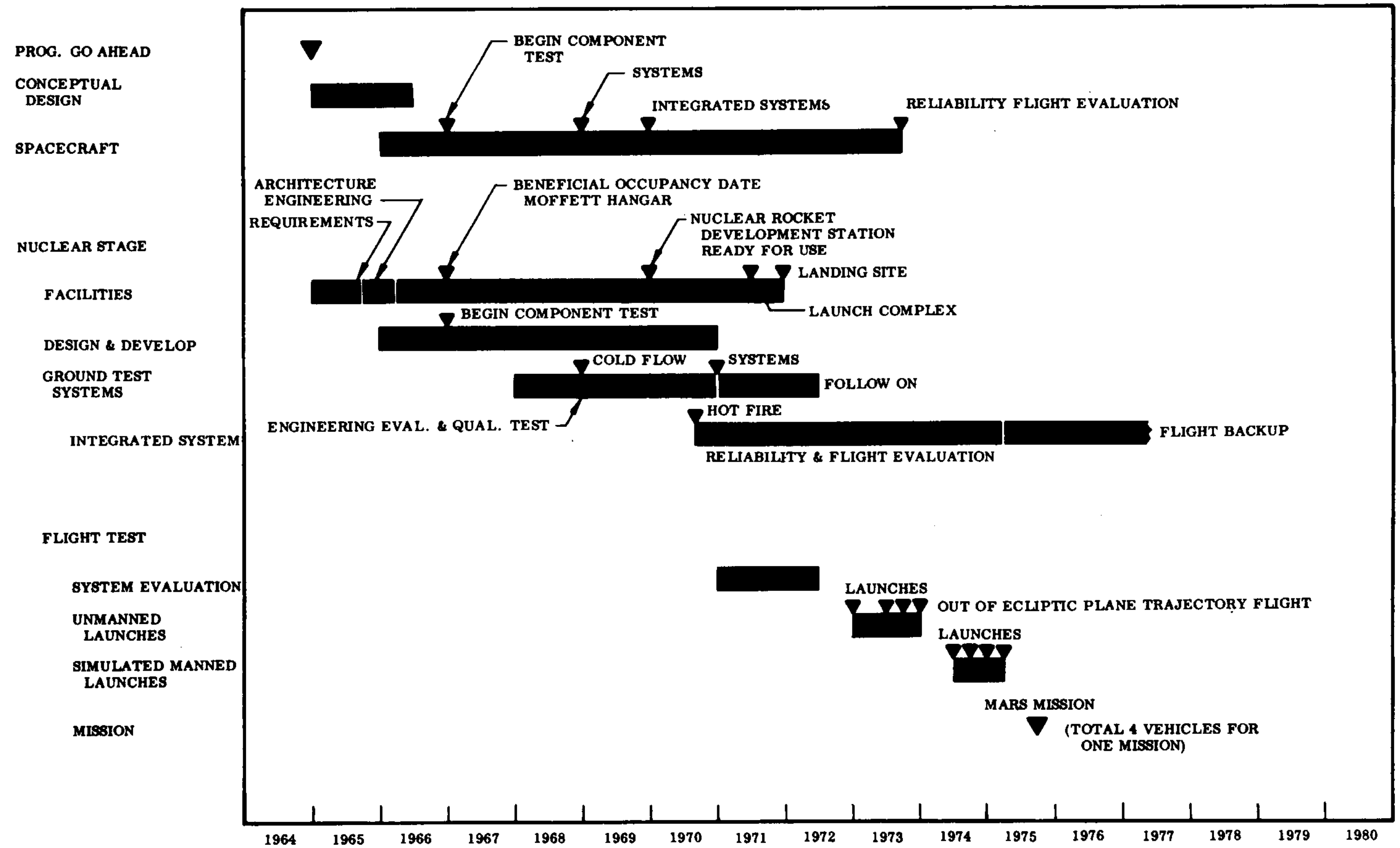


Fig. 24 Empire Program Mars Nuclear

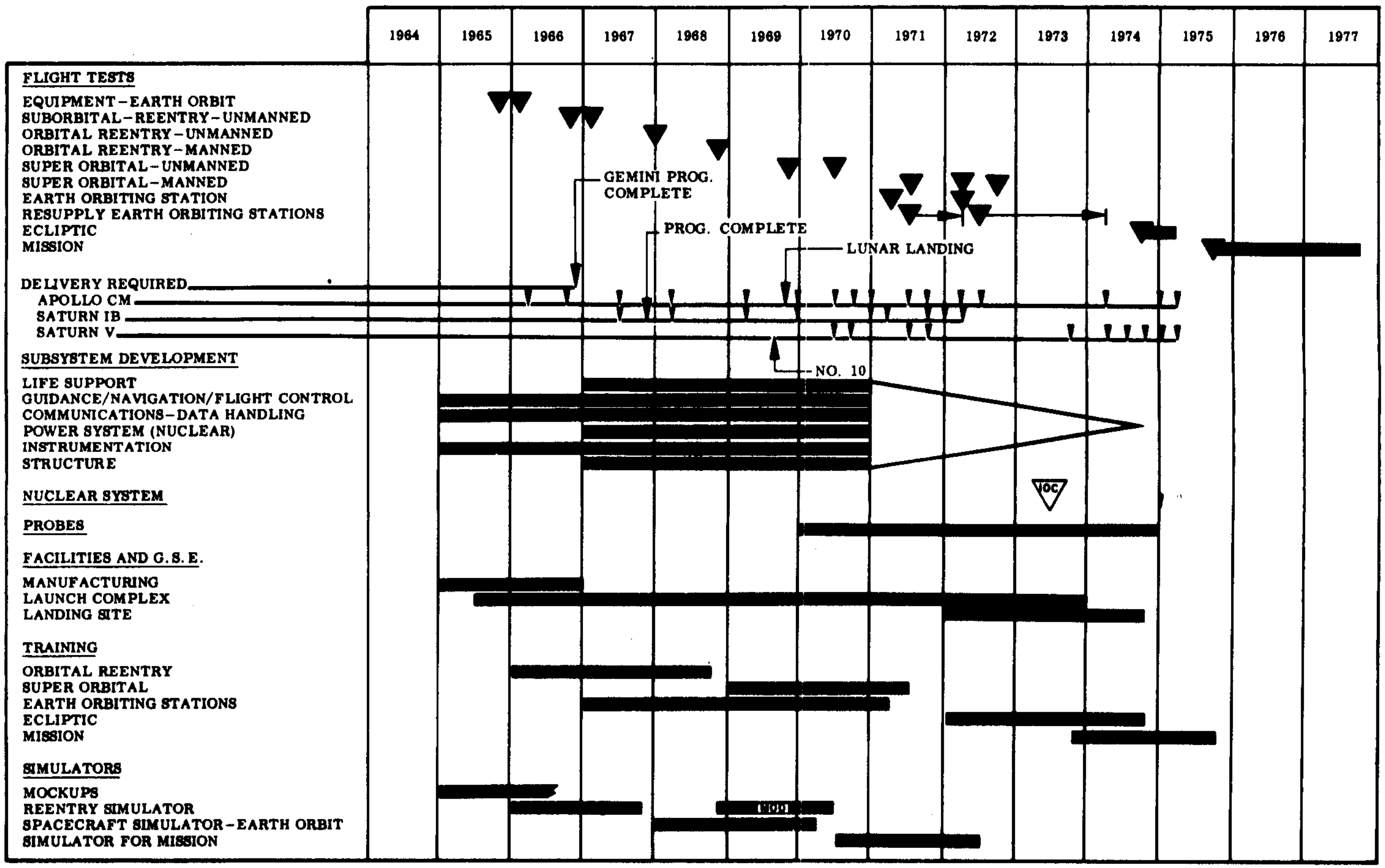


Fig. 25 Empire Program – Development Plan

Section 11
COST AND FUNDING SCHEDULE

A preliminary cost analysis of an early manned Mars and Venus flyby mission concept has been conducted. It must be emphasized that although fairly extensive operations research data and techniques have been utilized by experienced personnel, the accuracy of the results using concept information cannot be comparable to a study where component and systems definition have been established. Since this analysis has been conducted, the nuclear propulsion and staging system program has been redirected. If the nuclear system development cost is to be included in the present program, the estimate may have to be increased by one to two billion dollars. Although the study is necessarily cursory, the results, when used for planning purposes will be found to be quite adequate.

It appears that a Mars or Venus flyby mission will require from \$3.5 to \$3.8 billion depending on the nuclear stage used. The concept of using the all-chemical launch system (for Venus missions) was examined and was found to be within the estimating accuracy range of the \$3.5 billion figure. The estimate is based on conducting one Mars flight in 1975. Two launches are required to inject the spacecraft and nuclear propulsion system into Earth orbit. A complete backup of launch vehicles and spacecraft has been provided in case of a launch or rendezvous failure.

Due to time limitation, the cost analysis was concentrated on a Mars mission. As the development effort essentially is the same for either Mars or Venus as the objective, an additional flight could be conducted for approximately the cost of hardware procurement, some instrumentation development peculiar to the mission, and some additional crew training.

The spacecraft cost includes the modified Apollo command module, fairings and adapters, probes, retrorockets and nuclear power system. (Preliminary analysis of the solar version does not indicate a significant cost difference.)

The major system and subsystem development requirements were determined by analysis of the expected availability of off-the-shelf items, the possibility of modifying or repackaging items assumed to be developed, and evaluation of the items that must be developed for this program.

Procurement of Atlas, Little Joe II, Saturn IB, and Saturn V launch vehicles are required for component and system tests, Earth orbit and reentry tests. Launch base operations, G. S. E., and facility modifications are also included. Allowance has been made to bring the development of the nuclear stages to the man-rated condition required for this program.

Analysis of the development plan indicates that the maximum funding requirement in any calendar year would not exceed $600 million, as shown in Fig. 26, based on development of either the SN-A or SN-B nuclear stage. Itemized cost estimates are presented in Table 8.

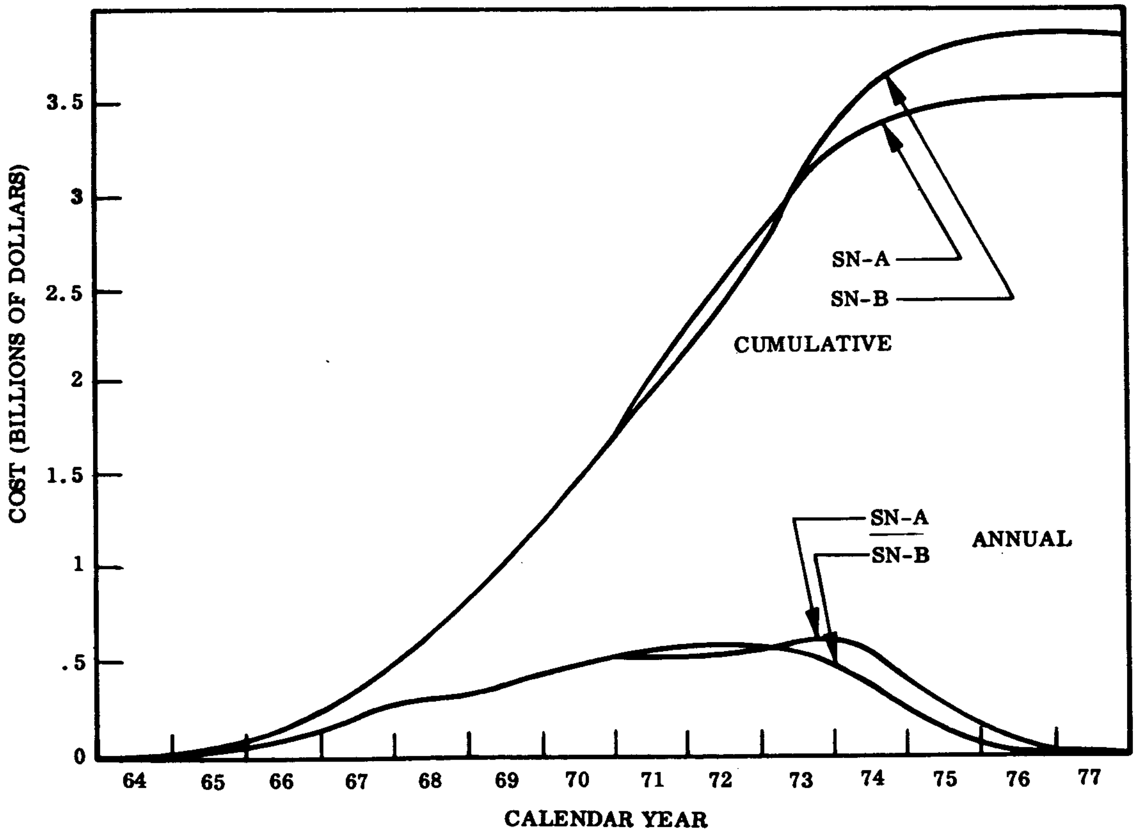


Fig. 26 Funding Estimate

Table 8

STAGE DEVELOPMENT COST ESTIMATE
(Millions of Dollars)

	System	
	SN-A	SN-B
Engineering and Systems Integration	$ 700.0	$ 700.0
Spacecraft Hardware	788.0	788.0
Launch Vehicles	631.0	642.0
Apollo	280.0	280.0
Launch Operations	98.1	98.2
Subsystems Development	443.0	443.0
Probes	250.0	250.0
G.S.E. and Facilities	50.0	50.0
Development of SN-B Stage	–	540.0
Man Rating of SN-A Stage	194.2	–
Crew Training	8.0	8.0
Communications During Mission	5.0	5.0
Maintenance	30.0	30.0
Transportation	20.0	20.0
Total	$3497.3	$3854.2

26982

PART 4

AMES RESEARCH CENTER MARS MISSION STUDIES INTRODUCTION

by

C. A. Syvertson
Ames Research Center

AMES RESEARCH CENTER MARS MISSION STUDIES

INTRODUCTION

By

C. A. Syvertson, Ames Research Center

At the present time Ames Research Center has three studies which are sufficiently far along to be reported at this symposium. These three studies are listed in Figure 1. Two of the studies are parallel and cover the subject of systems concepts and effectiveness. One of these is contracted to the Space Technology Laboratories (NAS 2-1409) and will be reported by Mr. R. L. Sohn. The other is contracted to North American Aviation (NAS 2-1408) and will be reported by Mr. A. L. Jones. Both of these are nine-month studies and the material to be reported here will cover cover approximately the first seven months of work. The third study is a trade-off study of descent systems. It is contracted to Northrop Ventura (NAS 2-1411) and will be reported by Mr. R. N. Worth. This study was originally a nine-month study but was recently extended to 11 months. Mr. Worth will cover approximately the first seven months of the study. Within the time restrictions of the symposium, results of these studies cannot be covered in great detail. For this reason only two or three selected points will be examined in each presentation. In addition, an attempt will be made to summarize results of each study that have been obtained to date.

For both of the Ames Mars studies, the objectives are essentially the same and are listed in Figure 2. The first objective is a common one, to examine the technical feasibility of the landing and return mission. In the studies managed from the Ames Research Center, this objective is achieved through the use of trade-off studies involving the principal parameters and modes applicable to the mission. From the trade-off studies, the more promising concepts are determined and these concepts are given additional study to establish their characteristics in greater detail. The last objective listed is one of the primary reasons for Ames activities in the advanced study area; namely, the studies expose technical problem areas and thus assist in the planning of in-house research.

A-31876-1

I - MANNED MARS SYSTEMS CONCEPTS AND EFFECTIVENESS

a. SPACE TECHNOLOGY LABORATORIES
R.L. SOHN

b. NORTH AMERICAN AVIATION
A.L. JONES

II - DESCENT SYSTEM TRADE-OFFS

a. NORTHROP—VENTURA
R.N. WORTH

Figure 1.- Ames studies.

A-31876-2

- DETERMINE TECHNICAL FEASIBILITY OF LANDING AND RETURN MISSION
- CONDUCT TRADE-OFF STUDIES OF PRINCIPAL PARAMETERS AND MODES
- ESTABLISH CHARACTERISTICS OF MOST PROMISING CONCEPTS
- EXPOSE TECHNICAL PROBLEM AREAS

Figure 2.- Study objectives.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER, MOFFETT FIELD, CALIFORNIA

For the two parallel mission studies the guidelines are listed in Figure 3. First, the contractors were asked to examine not only landing missions but Mars orbiting missions as a limit to the Mars orbital rendezvous mode. They were asked to study the use of both Venus and the Moon to reduce energy requirements associated with the missions. In the human factors area trade-off studies were requested in the closure of the life-support system and in the interplay between the environmental protection and the life-support systems. Tasks required of the crew were also to be analyzed. In these studies it was suggested that the contractors use the Schilling No. 3 atmosphere for Mars. Number 3 is the mean atmosphere defined by Schilling. There has been, of course, considerable discussion recently as to the probable atmosphere on Mars. This subject is covered in more detail in the Northrop descent systems study.

As indicated earlier, the studies are essentially parametric in form. The parametric constraints applied to the manned Mars studies are shown in Figure 4: Crew sizes from 3 to 10 men, mission duration from 12 to 18 months, and stay times from 7 to 60 days. At Earth, atmospheric braking is treated for vehicles with lift-drag ratios of 1/2 and 1. The contractors were asked to examine both propulsive and atmosphere braking at Mars. The propulsion systems to be examined were chemical systems using hydrogen and oxygen and hydrogen and fluorine for the period 1971 to 1986 and nuclear for the period 1975 to 1986.

For the descent-system study contracted to Northrop Ventura, the guidelines differ from those for the overall mission studies. These guidelines are shown in Figure 5. The systems to be studied were to touch down on land, not water. The entry vehicles for use at Mars were to be blunt bodies, essentially of the Apollo type. For Earth entry the vehicles were to be half cones of the M-1 lifting body type. At the start of the study, attention was to be restricted to the Schilling estimates of the atmosphere for Mars. This guideline developed to be a particularly poor choice since, shortly after the start of the study, estimates of the Martian atmosphere began to change rapidly. This problem was solved by changing the atmosphere of Mars from a guideline to a parameter. This and other parametric constraints are shown in Figure 6. When the Martian atmosphere situation developed, the study was extended for two months and Northrop was asked to examine various atmospheres from the lowest surface pressure suggested to the highest.

A-31876-3

- MARS ORBIT MISSION AS LIMIT TO MOR MODE
- EXAMINE USE OF VENUS AND MOON
- HUMAN FACTORS — LIFE SUPPORT, ENVIRONMENTAL PROTECTION, TASK ANALYSIS
- DEPARTURE POINT — 100 AND 300 N.M. EARTH ORBIT
- SCHILLING NO. III ATMOSPHERE FOR MARS

Figure 3.- Manned Mars: guidelines.

A-31876-4

- CREW - 3 TO 10 MEN
- MISSION DURATION - 12 TO 18 months
- STAY TIME - 7 TO 60 days
- ATMOSPHERIC BRAKING AT EARTH-L/D=0.5 AND 1.0
- PROPULSIVE AND ATMOSPHERIC BRAKING AT MARS
- PROPULSION: H_2/O_2 AND H_2/F_2 1971 - 1986
NUCLEAR 1975 - 1986

Figure 4.- Manned Mars: parametric constraints.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER, MOFFETT FIELD, CALIFORNIA

A-31876-5

- TOUCHDOWN ON LAND NOT WATER
- BLUNT BODIES FOR MARS ENTRY
- HALF CONES FOR EARTH ENTRY
- SCHILLING ATMOSPHERES FOR MARS

Figure 5.- Descent systems: guidelines.

A-31876-6

- MARS ATMOSPHERE
- RANGE OF PARACHUTE L/D
- RANGE OF PROPELLANTS AND ENGINE ARRANGEMENTS
- VEHICLE WEIGHTS - 20,000 TO 200,000 lb
- DESCENT SYSTEM - UP TO 15 percent OF VEHICLE WEIGHT

Figure 6.- Descent systems: parametric constraints.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER, MOFFETT FIELD, CALIFORNIA

Other parametric constraints included a range of lift-drag ratios, a range of propellants and a range of engine arrangements. It was left to the contractor to select the bounds for each of these ranges. The vehicle landing weights to be examined were from 20,000 to 200,000 pounds. Descent-system weights in excess of 15 percent of vehicle landing weight were not to be considered. The contract covers systems for both Earth and Mars but in the paper to be presented by Mr. Worth attention is restricted to Mars landing.

Following the three presentations, Mr. H. Hornby of Ames Research Center will summarize very briefly what we at Ames feel are the most important results obtained to date from these studies.

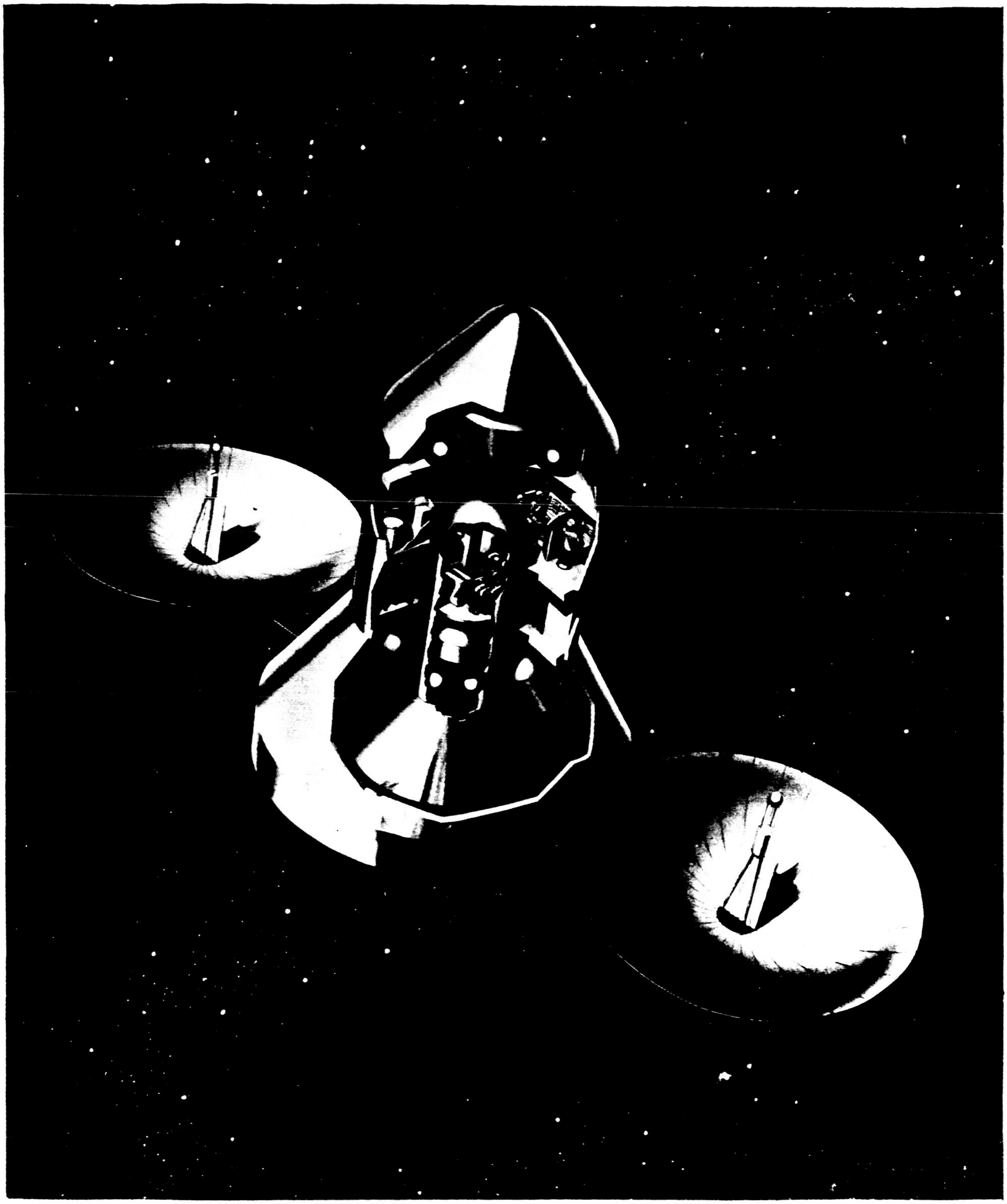
26983

PART 5

SUMMARY OF MANNED MARS MISSION STUDY

by

R. L. Sohn
Space Technology Laboratories



FRONTISPIECE

SUMMARY OF MANNED MARS MISSION STUDY

By

R. L. Sohn, Space Technology Laboratories

FOREWORD

Manned exploration of the planet Mars is the key mission in interplanetary space flight. Man must play a dominant role in the exploration of Mars because the planet is relatively complex, remote, and less amenable to exploration by unmanned probes than is the Moon. Important and necessary data gathering tasks will be accomplished by the unmanned probes and landers, but full realization of the interplanetary space sciences programs can only be reached through the manned missions, and in particular, the Manned Mars Mission.

For these reasons, serious interest in the Manned Mars Mission is springing up throughout the space sciences and engineering community, with many planning studies being performed by several study teams within the National Aeronautics and Space Administration, and within industry. The studies supported by NASA and industry are yielding valuable results in terms of providing a thorough understanding of the key technical problems and the comparative effectiveness of possible solutions to these problems, and a comprehensive summary of mission mode and system design trade-offs. For example, the development of the Venus swingby return mode, as described in this paper, offers a means for greatly reducing the excessive Earth entry velocities encountered with the direct return mode, and makes possible the use of the Apollo entry system. Thus one of the major problem areas previously associated with the manned Mars stopover mission is avoided. Other advances undoubtedly will be made as the planning studies progress. The time scale for mounting this mission is such as to permit thorough and complete planning to be performed with resultant savings to the overall program.

Perhaps the most important result emerging from the present studies is the indication that the Manned Mars Mission can be performed in the relatively near future with equipment and techniques that will for the most part be brought into operation by the Apollo project. In summary, the Mars Mission is rapidly taking shape as the direct follow-on to the Apollo project.

SUMMARY

A study of Manned Mars Missions has been completed for the NASA Ames Research Center. The primary objectives of this study were to establish system trade-offs between mission modes, to evaluate and compare general approaches to key technical problems, and to identify and recommend technological areas for long-range research. A brief summary of results achieved is given below (the Mars orbital rendezvous technique was followed throughout).

Weight Scaling Laws

Weight scaling laws were derived for the spacecraft subsystems, including propulsion (nuclear-hydrogen and hydrogen-fluorine), life support system, propellant storage, communications and power. Estimates were made both for open and closed life support systems (closed water and atmosphere only), including parametric variations for mission duration and crew size.

Mission Evaluation Techniques

Two approaches were developed to aid in optimizing and comparing mission modes. First, nomographs were prepared based upon weight scaling laws and characteristic mission velocities. Optimum trajectories could be determined quickly by this technique when used in conjunction with graphs of mission velocities versus launch date.

The second approach was based upon analytical optimization techniques that use differential equations containing vehicle weight-velocity exchange ratios and derivatives of mission velocities with respect to calendar date and leg time duration, in selecting best launch dates and trip times. These techniques have been developed for a large variety of mission modes, and have been programmed on the IBM 7094. Computing volume is reduced by a factor in excess of 200 compared to direct computations.

Mission Characteristics

Comparisons of various mission modes were made for a range of mission opportunities from 1971 to 2000. The following results were obtained (no launch hold, 10 days dwell at Mars):

	Minimum	Maximum
Velocities (1000 fps)		
Earth depart,	13.3	14.9
Arrive Mars,	24.0	27.7
Mars depart,	13.5	16.6
Arrive Earth,	46.0	68.1
Trip Duration (days)	400	440
Spacecraft Weight (million lb)		
H_2F_2 - H_2F_2	1.04	1.43
Nuc - H_2F_2	0.68	0.87

A two-man Mars lander weighing 25,000 pounds is included in the spacecraft weights. A Mars orbit rendezvous technique was assumed.

Increasing Earth hold to 50 days increased the spacecraft gross weight from 1.43 to 1.53 million pounds for the 1975 mission; extending dwell time at Mars to 30 and 60 days increased spacecraft gross weight from 1.53 to 1.69 and 2.00 million pounds, respectively, assuming a 30-day hold capability at Earth. Propulsion deceleration at Mars increased the gross weight from 1.43 to 7.15 million pounds for the H_2-F_2 system.

Reducing aerodynamic entry speeds at Earth return can be accomplished in several ways in the direct return mode, all of which result in increased spacecraft gross weight. Biasing the trajectory optimization to reduce the entry speed from 66,500 to 60,000 fps increases the gross weight from 1.43 to 1.77 million pounds. Use of a retrorocket to reduce speed by the same amount increases gross weight to 1.61 million pounds.

In special cases it was found that Class I outbound trajectories (greater than 180° transit angle) resulted in slightly lower overall gross weights. Earth departure velocity requirements increased markedly, but were traded off against slightly reduced Mars departure velocities. An advantage was obtained only in the case in which a high performance nuclear-hydrogen Earth depart stage was used in combination with a low

performance storable propellant Mars depart stage, and only for the 1975 opportunity. Earth entry speeds were reduced to below 50,000 fps, but trip time was extended by more than 100 days.

Venus Swingby Modes

A very promising mission mode was examined in which the spacecraft returns from Mars to Earth by way of Venus, using the gravitational field of Venus to decelerate the spacecraft and (thereby) greatly reduce the Earth entry velocities. In a typical unfavorable year, 1975, Earth entry velocity is reduced from 66 to 46,000 fps. No increase in spacecraft gross weight results, although trip time is extended slightly. From 1971 to 1999, which covers two cycles of oppositions, the Venus return mode is achievable in 9 of 14 opportunities. The fact that more than two out of three opportunities can use the Venus return mode seems fortuitous; however, the stopover missions are found to be very flexible in achieving a rendezvous with Venus (because the angular rate of travel of Mars is small compared to that of Venus, by a factor of about four), with no sacrifice in propulsion requirements. Cases wherein the spacecraft arrives late for the rendezvous are more restricted, but can be accommodated if the delay is small.

Venus swingby also can be achieved during the outbound trip from Earth to Mars. This mode is normally precluded because the Earth departure propulsion requirements are excessive; however, these can be reduced to near-optimum values if Venus is used to accelerate the spacecraft toward Mars (the gravitational whip effect).

In summary, it was found that the Venus swingby mode can be used in all opportunities during the thirty-year period, with the result that Earth entry velocities are reduced to below 45-50,000 fps. In most cases no adverse effects have been noted. Earth and/or Mars launch-holds can be accommodated, and navigation requirements for the Venus gravity turns are less stringent than for aerodynamic entry into the Mars or Earth atmospheres. The overall contribution to the Manned Mars Mission achieved through use of an existing Apollo entry system as compared to the development of a 70,000 fps Earth entry system is readily appreciated.

Aero Entry

Analyses were made of the aerodynamic corridors required at Mars and Earth as a function of entry velocity and entry vehicle lift-to-drag ratio. At Mars the undershoot boundary was determined by the requirement to decelerate the spacecraft to escape velocity or less. For a nominal entry speed of 27,500 fps, an entry corridor of 31 km was obtained for a vehicle with L/D = 0.1.

For Earth entry, the undershoot boundary was determined by a maximum deceleration limit (assumed to be 10 g), and the overshoot boundary by the requirement to capture the spacecraft, i.e., decelerate it to escape speed or less. Entry corridors at Earth diminish from 22.2 and 16.7 km at 60,000 fps entry speed for vehicles having L/D = 1.0 and 0.5, respectively, to 15.4 and 10.2 km at 70,000 fps (relative to the atmosphere). A Mars entry corridor analysis was run using the JPL model atmosphere (15 mb mean); Mars corridors were reduced by less than a factor of 2, and pose no new navigational problems.

Estimates of entry vehicle heat shield weights were made (the weight of the Mars entry body weight was based upon latest estimates of the Mars atmosphere). The Mars entry vehicle heat shield weights varied from 30 to 40 thousand pounds as entry speed increased from 25 to 30 thousand feet per second. Earth entry system weights increased from 12.2 to 19.8 as entry velocity increased from 50 to 70 thousand feet per second. The presence of CO_2 in the Martian atmosphere gives rise to strong radiation heating over blunt nose shapes, and leads to the selection of a relatively pointed Mars entry body. Heat soak after entry into the Mars atmosphere gives rise to large insulation weights, making it desirable to jettison the heat shield immediately after exit from the atmosphere.

Navigation

A computer simulation of an onboard navigation system has been made, using typical outbound and inbound trajectories for the 1978 mission. The sensors were assumed to observe the Sun, Canopus, and the target planet. With the state of the art error models chosen, entry corridors at Mars and Earth, respectively, are 0.7 and 20 km. Use of a second star sensor in place of the sun sensor reduces the entry corridors to 0.7 and 7.0 km at Mars and Earth, respectively, and offers the additional advantage of reducing the uncertainties at a

considerable distance from the target planet. Use of DSIF radio tracking at Earth entry reduces the uncertainties to less than 1 km. Navigation propulsion requirements were 350 fps outbound and 300 fps inbound (assuming direct return from Mars).

Earth Depart Windows

Earth depart windows have been computed, and, as noted by prior study groups, the parking orbit precision rate is excessive, and results in restricted launch windows. For the 1975 mission, opportunities exist about eight days before, and 40 and 48 days after the nominal launch day (launch can be effected from about ten orbits per opportunity). The opportunities occur more frequently in the "good" years, with less severe propulsion penalties. The results can be summarized as follows:

One-minute delay	120 fps
One-day delay	950
50-day launch season	1000
	1385 RMS

Solar Radiation Shielding

Hydrazine was selected for solar radiation shielding material, and subsequently used for terminal navigation and retro propulsion prior to entering the Earth's atmosphere. No significant weight difference was noted between water, polyethylene and hydrozine materials, and hydrazine was selected because it can be used to reduce Earth entry speeds by several thousand feet per second.

Artificial Gravity

An artificial gravity version of the spacecraft using empty propulsion tanks as counter weights was studied. A single cable was used to suspend the tanks from the main spacecraft; cable lengths of about 500 feet are required. This configuration was analyzed by the Langley Research Center, where a nine-degree freedom of motion dynamic model analysis routine is available on a computer. The results indicate that the single cable design is stable (long and short

term), and poses no problems with cable lengths up to 500 feet. The cable system weighs less than one percent of the spacecraft gross weight.

INTRODUCTION

A study of the Manned Mars landing and Return Mission has been completed for the NASA Ames Research Center under Contract NAS 2-1409.[1] The general goals of this study are to establish system trade-offs between mission modes, to evaluate and compare general approaches to key technical problems and to identify and recommend technological areas for long-range research.

The first phase of the study was devoted to the definition of basic mission modes, including the types of trajectories associated with the various launch opportunities, description and characteristics of maneuvers and operations near the planets (capture, orbit parking and rendezvous, descent and return of landers), determination of propulsion and aerodynamic entry system requirements, probable mission durations, and approximate spacecraft gross weights. These considerations served to define the general scope and limits of the Manned Mars Mission, and indicated the trade-offs and comparisons necessary to establish best approaches to the mission.

To implement these trade-offs, preliminary weight scaling laws were developed for the major elements of the mission spacecraft, aero entry system weights were derived for a range of entry velocities for both Mars and Earth. Propulsion system weights were estimated for nuclear-hydrogen and hydrogen-fluorine combinations, and for storables used during midcourse and retro maneuvers. Crew size and mission duration were retained as parameters in the mission module scaling laws. The overall weights were based on general design concepts developed by STL in connection with company sponsored studies of the Manned Mars Mission.[2]

---

[1] This contract was performed under the cognizance of Mr. C. A. Syvertson, Chief, Mission Analysis Division, and Mr. H. Hornby, Staff, Mission Analysis Division of the Ames Research Center. The comments and suggestions of the staff at Ames have been most helpful to the study team.

[2] R. L. Sohn, et al, "Feasibility Studies of Manned Mars Mission." STL Memo 9860.6-1, 25 March 1963.

The initial studies also included analyses of crew requirements and utilization, Mars landing party functions and operations, and a computer simulation of an onboard navigation system, which gave an accurate evaluation of achievable entry corridors at Mars and Earth along with midcourse corrections necessary to achieve the corridors.

Succeeding activities were devoted to analyzing and comparing the various mission modes. Special emphasis was placed upon the development of mission evaluation techniques that reduced the large volume of computing usually required to optimize each mission mode. Nomographs were prepared based upon the weight scaling laws, and were combined with the characteristic mission velocity requirements so that rapid computations of spacecraft gross weight could be made for any given trajectory mode. It was found that by graphing characteristic mission velocity as a function of calendar date (Mars arrival date), near-optimum trajectories could be selected by inspection.

The second approach to mission optimization was based upon analytical techniques using differential equations which contain vehicle weight-mission velocity exchange ratios and derivatives of mission velocities with respect to launch date and trip duration, in selecting best launch dates and trip leg times. These equations, which eliminate the need for repeated calculations of lengthy mission performance equations, have been programmed for the IBM 7094. These analytical techniques are especially beneficial in optimizing missions which contain provisions for launch holds.

Special attention was given to the analysis of interplanetary guidance and navigation, particularly the terminal phases which require accurate steering into a relatively narrow corridor, especially in the aerodynamic capture mode. A complete simulation of the problem was performed, using the TAPP (Tracking and Program) modified for onboard sensing. Corridor estimates and navigational propulsion requirements were derived by this technique for optical and ground-based radio tracking modes.

Selected design problems were analyzed, including provisions for artificial gravity, and economical arrangements for achieving acceptable solar radiation protection.

MISSION CHARACTERISTICS

The primary purpose of the study was to define the basic mission modes, including types of trajectories, descriptions of maneuvers and operations near the planets, propulsion and aerodynamic entry system requirements, probable mission durations, and approximate spacecraft gross weights. From these considerations, trade-offs were established between major elements in the mission, and between mission modes. The basic weight scaling laws used in developing these trade-offs are based on spacecraft designs described in a following section.

Mission Opportunities

First, it is of interest to note the opposition dates of Earth and Mars because the launch opportunities generally precede the oppositions by three to four months. Figure 1 illustrates the oppositions. It can be seen that the oppositions of 1978 and 1980 occur when Mars is near its apofocus, whereas the opposition of August 1971 occurs when Mars is near perifocus. The opportunities of 1971 and 1978 to 1980 correspond roughly to the favorable and unfavorable years for Mars missions for the 1971 to 1984 cycle of oppositions.

A factor entering into the trajectory selection is the inclination of the Mars orbit with respect to that of Earth. This inclination angle is small (1.85°), but moderately increases the required departure velocities when intercept at the target planet does not occur near the line of nodes. Also in the special case of near-180-degree transfers, the inclination of Mars causes the transfer trajectories to rotate upward normal to the ecliptic. This effect is confined to a small range of cases and was not found to constrain the selection of optimum trajectories, even with provisions for launch holds. In any event this phenomenon can be avoided by means of two-impulse trajectories that lie near the ecliptic but rotated near midcourse to accommodate the out-of-planeness of the target planet; moderate propulsion requirements are involved in this maneuver.

Mission Optimization Techniques

A serious problem encountered in previous study efforts was that of selecting the trajectory paths and calendar dates that tend to optimize the spacecraft design for a given mission mode and opportunity.

Usually repeated calculations of lengthy weight scaling laws were required for a range of trip durations and calendar dates to establish optimum combinations of these variables. Computational volumes become excessive if large numbers of cases are being examined, and particularly if launch-hold considerations are introduced.

Two approaches were developed in an effort to reduce the burdensome computational procedures involved in the mission optimizations. One of these approaches was based upon analytical techniques which use differential equations containing vehicle weight-velocity exchange ratios and derivatives of mission velocities with respect to launch date and trip duration, in selecting best launch dates and trip leg times. These equations have been programmed for the IBM 7094, and are found to reduce computational volumes by factors in excess of 200. A description of the method is given in Appendix A. Applications to onboard computers are possible.

In addition to the analytical optimization techniques, it has been proved useful to develop mission analysis nomographs, which contain the spacecraft weight scaling laws, and the mission characteristic velocities plotted as a function of calendar date. Leg times are treated as parameters. The basic nomograph computes component spacecraft weight as a function of mission velocities. By tracing through the complete nomograph, a value of spacecraft gross weight in Earth orbit can be achieved for a given set of mission velocities. The nomograph based on STL scaling laws is presented in Figure 2, velocities for the 1973 opposition are shown.

Representing the mission velocity requirements in the manner shown in Figure 3 yields considerable insight into the basic velocity trade-offs involved in the overall optimization. This is achieved by graphing the velocities as a function of calendar date at Mars. An important advantage is gained in that the outbound and inbound legs can be optimized independently for a given calendar date.

It is noted also that the propulsion mode rather than the aerodynamic mode dominates the optimization of a given leg, assuming aerodynamic capture at the target planet. In this case, optimum leg time can be found simply by graphing the lower envelopes of the velocity curves. The nomograph is then used to calculate the overall spacecraft gross weight variation along the envelopes over a range of calendar dates to find the best calendar date. Near-optimum conditions

can be selected by inspection. If propulsion deceleration at Mars is specified, the outbound leg times do not fall along the depart-Earth velocity envelopes, and it is necessary to examine a range of leg times at each calendar date.

Launch Holds

A brief review of missile launch holds indicates that delays tend to fall into general classes according to the seriousness of the malfunction: minor malfunctions give rise to delays of a few seconds or a few minutes, more serious problems delay the launch by periods of several hours or a few days, and major problems cause postponements of one to several weeks.

Launchings from orbits can be made with compatible frequencies, and indications of possible performance penalties can be obtained by assuming delays of the above magnitudes and calculating the changes in the burnout velocity vectors. In general, the performance penalties due to delays are much more severe when firing from parking orbits because of the rapid motion of the spacecraft.

For delays of a few minutes, the following types of penalties are encountered. Nominally the Earth burnout velocity vector will produce a heliocentric velocity vector that guarantees a given arrival date at Mars. If the spacecraft fires late, the burnout velocity vector rotates beyond the nominal and must be corrected if the Mars arrival date is to be met, giving rise to propulsion corrections of several hundred feet per second. The magnitude of these corrections can be reduced if the Mars arrival date is adjusted slightly (which gives rise to very small "non-optimum" penalties), and if the delay is "centered" about the nominal, that is, 30 seconds before nominal to 30 seconds after nominal for a total delay of one minute. The resultant propulsion correction for a one-minute delay is reduced to 120 fps.

Postponement of the launch to the next opportunity, which occurs on the succeeding orbit, would result in no additional penalties were it not for precession of the parking orbit plane due to the equatorial bulge. Unfortunately, the precession rates are relatively high for the cases of interest, of the order of several degrees per day. As a result, parking orbit plane changes must be made if launch is not accomplished on the nominal orbital pass. The magnitude of the performance penalty is a function of the inclination of the parking orbit

with respect to the equator, and of the number of orbits missed. The inclination of the parking orbit is lower in the favorable years (because of the reduced heliocentric out-of-plane corrections) and is such that holds of one full day (17 orbits) can be accommodated with a performance penalty of 720 fps. This penalty increases to 950 fps in an unfavorable year, such as 1975.

If launch is not accomplished on the nominal day, postponement must be made until the parking orbit precesses to the next launch opportunity. If the parking orbit is inclined slightly above its minimum value, this opportunity occurs several days after nominal, when the corresponding ascending (or descending) node firing point rotates into position. Additional opportunities do not occur until the parking orbit rotates a full 360 degrees, which requires about 41 to 48 days over the cycle of favorable to unfavorable years. Hence the launch season for a Mars mission is of the order of 1.0 to 1.5 months. A delay of 1.5 months moves the mission past the optimum calendar dates, and results in an additional 1000 fps performance penalty.

The three performance penalties derived above can be summarized as follows (45-day season with a total of 68 firing passes):

1.	One minute delay in firing	120 fps
2.	One day delay in launch	950 fps
3.	Non-optimum penalty over 45-day season	1000 fps
		2070 fps
		1385 (RMS)

Launch hold delay penalties are summarized in Figures 4 and 5.

Launch delays off Mars are less severe although the precession of the parking orbit plane is relatively rapid, leading to the same types of delays encountered at Earth. If the non-optimum penalties are absorbed in the normal allowance for dwell time on Mars, assuming a 30 to 40 day stopover, Mars launch delays result in a 575 fps penalty (one minute delay in firing, and one day delay in launch, 40-day season, giving 30 firing passes). An elliptic parking orbit greatly eases the Mars departure maneuver because of reduced plane change penalties: a continuous 10-day departure window (52 firing passes) can be obtained for a penalty of 275 fps.

General Mission Characteristics

General mission characteristics have been established for Mars orbiting stopover modes using aerodynamic and propulsion braking at the target planets. The aerodynamic deceleration mode requires the least weight in Earth orbit for manned landing missions (compared to direct landings, or to modes involving propulsion deceleration). A stopover of 10 days is assumed, with a Mars orbital altitude of 500 km. Using the given weight scaling laws, a complete optimization was carried out to select the combination of Julian date, total trip time, and leg time to minimize total spacecraft weight on Earth orbit. The results obtained are shown in Figure 6.

Figure 6a indicates the optimum launch dates and leg times for the various launch opportunities from 1971 through 1980. Overall trip times (for the aerodynamic deceleration modes considered herein) vary from 400 to 440 days. It is interesting to note that optimum Earth launch dates occur approximately one hundred days before the opposition dates.

Figure 6b indicates the characteristic mission velocities. Surprisingly little variation in velocities occurs over the range of favorable to unfavorable opportunities, with the exception of the Earth arrival velocities, which increase rapidly during the late 70's when opposition occurs as Mars nears its apofocus. The fact that aerodynamic entry system weights are relatively small compared to equivalent propulsion system weights causes the mission to be biased heavily in favor of reduced propulsion system requirements. Use of other aerodynamic entry system weight scaling laws would not change these results appreciably. An upper bound on Earth entry speeds will have to be determined by feasibility of guidance and control limitations, rather than by system weight optimization.

Total weights of the spacecraft in Earth orbit are shown for nuclear-hydrogen and hydrogen-fluorine Earth depart propulsion systems (hydrogen-fluorine is used for Mars depart in all cases). For the assumed weight scaling laws the spacecraft gross weights range between 1.04 and 1.43 million pounds for the 1971 and 1975 missions, respectively, for hydrogen-fluorine propulsion. Use of nuclear-hydrogen Earth depart propulsion reduces spacecraft gross weights by 40 percent, to 0.87 million pounds (assuming an I_{sp} of 900 sec).

The effects of variations in Earth hold time, dwell time at Mars, propulsion braking at Mars, and reduction in Earth entry speed to 60,000 fps are shown in Figure 6c for the 1975 mission.

An Earth launch hold of 45 days increased the gross weight from 1.43 to 1.53 million pounds; extending dwell time at Mars from 10 to 30 and 60 days increased spacecraft gross weight from 1.53 to 1.69 and 2.00 million pounds, respectively, assuming a 45-day hold capability at Earth. Propulsion deceleration at Mars increased the gross weight from 1.43 to 3.25 million pounds for the H_2-F_2 system.

Reducing aerodynamic entry speeds at Earth return can be accomplished in several ways, all of which result in increased spacecraft gross weight. Biasing the trajectory optimization to reduce the entry speed from 65,600 to 60,000 fps increases the gross weight from 1.43 to 1.77 million pounds. Use of a retrorocket to reduce speed by the same amount increases gross weight to 1.61 million pounds. Reducing the Earth entry speed to 50,000 fps by means of retro propulsion increases gross weight from 1.43 to 2.05 million pounds.

In special cases it was found that Class I outbound trajectories (greater than 180° transit angle) resulted in slightly lower overall gross weights although Earth departure velocity requirements increased considerably, which are traded off against slightly reduced Mars departure velocities. This effect was noted only for a 1975 mission in which a high performance nuclear-hydrogen Earth depart stage was used in combination with a low performance storable propellant Mars depart stage. Earth entry speeds were reduced to below 5,000 fps, but trip time was extended by more than 100 days.

Trajectory characteristics for the 1975 mission are shown in Figure 7. Transit orbital plane inclination angles are small. During the late 70's the optimum trajectories tend to pass inside the orbit of Venus, and might pose thermal control problems.

VENUS SWINGBY MODES

Venus swingby return modes have proved to be extremely beneficial to the Manned Mars Stopover Missions because of the very large reductions in Earth entry velocities possible with this mode. Essentially the gravitational field of Venus is used to reduce the heliocentric velocity vector of the spacecraft as it passes by Venus, resulting in a more nearly tangential approach to the Earth in contrast to the relatively high radial velocity vector components encountered in the direct return mission modes.

The advantages of the Venus swingby return modes can be demonstrated by a comparison of the direct and Venus swingby return modes for the 1975 mission, which is a typical unfavorable year with an Earth entry velocity of 65,600 fps. The trajectory paths are shown in Figure 8. In the direct return mode from Mars back to Earth the spacecraft passes inside the orbit of Venus to effect its rendezvous with Earth. At Earth, the spacecraft heliocentric flight path angle is 31 degrees with respect to that of the Earth, and results in an Earth entry velocity of 65,600 fps.

By adjusting the return trajectory slightly, the spacecraft can be made to rendezvous with Venus, and if a dark side passage is made at an altitude of about 3300 km, the spacecraft heliocentric velocity will be reduced by 15,000 fps. The resulting Venus to Earth passage approaches Earth in a more nearly tangential path. The heliocentric velocity vector is within 14 degrees that of Earth, and, as a result, the entry velocity at Earth is reduced to 44,000 fps. This result is of extreme importance to the Manned Mars Stopover Mission because entry velocities of 65,000 to 70,000 fps are of marginal feasibility, and may require large retro deceleration.

The question was raised concerning the applicability of the Venus swingby return mode to other opportunities, particularly to those in the unfavorable years. Subsequently two cycle of opportunities have been examined, from 1971 to 1999 (14 opportunities). In many opportunities it was found that the spacecraft did not pass near Venus on the return trip. However, the angular rate of travel of Venus is so rapid compared to that of Mars (by a factor of 4), that the spacecraft can be parked on Mars for relatively short periods of time to await a rendezvous. In addition, the entire mission can be adjusted to effect a

more favorable position for Venus rendezvous. The situation is further enhanced because rendezvous can be effected before or after perihelion passage of the spacecraft. Altogether about 75 percent of the missions can effectively use the Venus swingby return mode to reduce Earth entry velocities.

In the remaining opportunities, Venus swingbys can be éffected by "reversing" the trajectory, that is, by following the long (greater than 180 degree) transfers out to Mars, and returning to Earth by the direct (less than 180 degree) transfers. This mode is normally precluded because the Earth departure propulsion requirements are excessive; however, these can be reduced to near-optimum values, if, subsequent to Earth departure, the whip effect of Venus is used to accelerate the spacecraft towards Mars. Hence, the role of Venus is now to accelerate the spacecraft rather than to decelerate it as in the case of the return swingby. The net result is the same, however, in that the Earth entry velocities are reduced to less than 50,000 fps.

A summary of the Venus swingby mode analyses is given in Figure 9, which compares the Earth entry velocities encountered with direct return mode trajectories with those achieved with the Venus swingby mode. Startling reductions are apparent; even the favorable years are benefitted. A Venus swingby mode can be achieved in all 14 of the oppositions to 1999.

The effects of Venus swingby return modes on total trip time is also shown in Figure 9. In those cases in which Venus rendezvous is made before perihelion passage, trip time is extended to about 500 days. If rendezvous is effected after perihelion passage, little or no extension in trip time results. Spacecraft gross weight is essentially unaffected.

To summarize, certain advantages and disadvantages of the Venus swingby return modes are pointed out in Figure 10. The great reductions in Earth entry velocities made possible by the Venus swingby modes far overshadow other factors involved, and essentially eliminate that key technical problem from the Manned Mars Stopover Missions. As will be discussed later, the use of hydrazine for solar radiation shielding would permit Earth entry retro maneuvers that would essentially reduce the Earth entry velocities (of the Venus swingby mode) to the level of the Apollo entry system. The benefits to be derived are apparent.

AERO ENTRY SYSTEMS

The Manned Mars Mission is characterized by large propellant requirements which lead to large spacecraft weights and correspondingly large Earth launch vehicles. Aerodynamic braking at Mars must be given serious consideration as a means to reduce the propulsion and hence gross weight requirements. In fact successful application of aerodynamic capture at Mars is the most powerful single factor in reducing the size of manned Mars spacecraft.

Earth entry is a second vital phase of the mission, as indicated in the discussion on mission characteristics. The basic mission trade-off relations (which minimize spacecraft gross weight) invariably yield trajectory optimizations that minimize propulsion system performance requirements, at the expense of increased Earth entry velocities. As a result, these velocities tend to range up to 70,000 fps with direct return modes, and, with the exception of the 1971 mission all missions through 1982 have Earth entry velocities in excess of 58,000 fps. Much analytical and experimental work must be performed to determine whether or not Earth entry at these velocities is indeed feasible.

Various aspects of the technical problems are discussed below.

Mars Atmosphere

The model of the Mars atmosphere that was used for flight mechanics and heating calculations in the study was derived from Schilling's "Conjectural Model III Mars Atmosphere." The much-reduced density levels of recent JPL Mars atmosphere models were considered for typical Mars entry conditions, and found to have a minor effect on corridor entry requirements.

The chemical composition of the atmosphere of Mars was assumed to be 1.9 percent CO_2 and 98.1 percent N_2 by volume.

Mars Entry Corridors

To determine entry corridor requirements, trajectories were computed for a range of typical entry velocities, with the results shown in Figure 11. A vehicle can, by proper orientation, fly the entire maneuver with upward lift of downward lift. If the desired exit velocity

is 11,000 fps the corridor depth will be the difference between 388,000 and 303,000 ft, or 30.5 km for a vehicle with L/D = 0.1 and an entry velocity of 27,500 fps. It can be seen that this corridor is decreased if a higher exit velocity is required. However, this effect is not large except for the higher L/D values, where the corridor is already deep. A variation of scale height of the atmosphere gives a less-than-proportional variation in corridor depth. The JPL Mars atmosphere model recently proposed has a substantially reduced scale height, and a far lower surface density. As a result, corridor depths for this model are reduced to 21.4 km, and the distance of closest approach lowered to about 100,000 ft. Navigational requirements are not made critical by this assumed atmospheric model, as will be discussed in a following section. Maximum resultant deceleration levels do not exceed 3.6 g's for a spacecraft lift-to-drag ratio of 0.1.

The higher value of W/C_DA actually used in design (765 lb/sq ft) will not affect the corridor values given above for an L/D = 0.1, although the periapsis altitudes are reduced.

Earth Entry Corridors

For superorbital entry velocities the corridor depth is the difference in altitude between the virtual (vacuum) perigees of the overshoot and undershoot trajectories. The overshoot trajectory is the trajectory using downward lift which, at the entry velocity being considered, has the highest possible virtual perigee while still remaining in the atmosphere. The undershoot trajectory is the trajectory using upward lift which reaches, but does not exceed a specified resultant g limit. Flight in the atmosphere subsequent to these points is not involved in the determination of the corridor although the flight must proceed in such a manner as to prevent further escape from the atmosphere by flying too high or exceeding the g limit by flying too low.

The results of the Earth entry corridor analyses are shown in Figure 12, for a maximum resultant g-limit of 10. Corridor depth as a function of velocity ceases to exist when the maximum deceleration on the overshoot trajectory exceeds 10 g's. For example, at 70,000 fps for L/D = 0.5, a 10 g maximum resultant deceleration is encountered on the overshoot trajectory and therefore no corridor exists for velocities above 70,000 fps.

It is apparent that the corridor depths available become restrictively small at the high entry velocities. The question arises as to corridor depth requirements. This required depth will be made up of several contributions. The first and probably most important corridor requirement is that of guidance accuracy, which in turn is made up of two parts, including possible variations in the location to which the spacecraft can be guided, and secondly, the uncertainty with which the location is known. Another variation contributing to required corridor depth is the possible variation in atmospheric density or its equivalent altitude variation. Results of prior studies indicate that this variation is about +3.7 km. Thus, if the guidance capability is 11 km the required corridor is 18 km from these two contributions.

The available corridor can be deepened by modulating the lift, or, in general, by modulating lift and drag. However, it is assumed that the problems of flight, including maintenance of some known vehicle shape, would not permit the use of controls which would, for example, vary the trim angle of attack. The use of roll to vary the vertical component of lift, which is the mode considered here, does not give g alleviation as does lift variation.

As mentioned previously, a resultant 10 g acceleration limit was observed in determining entry corridor requirements. However, as corridor depths are restrictively small, an appreciable percentage increase can be attained by a relaxation of the g limitation. For the case of a vehicle with $L/D = 1$ and an entry velocity of 70,000 fps, a change from 10 to 16 g's increases the corridor depth by 9 km.

Entry Configurations

As noted previously, a significant problem during Earth entry on the manned Mars mission is guidance into the entry corridor. At a given entry velocity the corridor depth is controlled by the lifting capability of the entry vehicle. Although the optimum vehicle configuration is affected by several less important system requirements, the vehicle lifting capability and the weight needed for entry heating protection are principal configuration determinants. For the purpose of the mission analysis, configurations representative of lift-to-drag ratios of 0.5 and 1.0 have been studied. The combination of lift-to-drag ratio requirements and the necessity of minimizing radiant heating tends to indicate a somewhat pointed vehicle in contrast to the Apollo shape. The shapes of the vehicles used are given in Figure 13. The slender

half cone configuration with a nominal L/D = 1.0 may be capable of a somewhat greater lift-to-drag ratio, however, for the higher entry velocities the ablation phenomenon will cause the exact shape and, therefore, the aerodynamic coefficients to be somewhat uncertain at any particular velocity. For this reason, also, it is assumed that external controls are not used to modulate lift; instead a rolling mode of control is used, similar to Apollo. Jet controls can be used to control attitude prior to entry; during atmospheric flight the angle of attack which gives the nominal lift-to-drag ratio can be attained by the positioning of the center of gravity with provision for some adjustment in the design. The extreme heating problems make control surfaces undesirable.

Mars Entry

From the study of aerodynamic capture trajectories in the Martian atmosphere, it was found that a lift-to-drag ratio of 0.1 was apparently sufficient to attain the required corridor depth. This low value of L/D permitted use of a symmetric vehicle; the weight estimates needed in the mission analysis were made on this basis. The necessary angle of attack can be attained with an offset center of gravity or with some asymmetry of the aft flare. Modulation of lift can be attained by rolling. A relatively pointed front end was used to minimize radiant heating.

Earth Entry System Heat Shield Design

To implement the mission weight trade-off analysis it was necessary to have the weight of the heat shield as a function of entry velocity for aerodynamic braking at Earth entry. The determination of aerodynamic heating for the extreme velocities encountered in Earth entry are beyond the present state-of-the-art, making it necessary to use reasonable assumptions based on known data and analyses to arrive at representative weights.

For velocities approaching 70,000 fps at Earth entry the usual methods of computing convective and radiant heating break down. The energy depletion due to radiation out of the shock layer becomes so large that the effects on both radiant and convective heat transfer are considerable. The properties of the gas in the shock layer, especially temperature, are not uniform, and absorption of energy in the shock layer becomes important. The effects of radiation losses and absorption

on the radiant heat transfer have been analyzed recently by Yoshikawa and Chapman, and by Howe and Viegas. Yoshikawa and Chapman introduced a radiant heat transfer coefficient, λ, which relates the incident radiant heat rate to the total rate of energy flow by the relationship $q_r = \lambda(1/2\ \rho\ V^3)$. For very large vehicles (nose radius ~ 10) and velocities greater than 50,000 fps, total radiant energy losses (including radiation in all directions) in some cases were found to be as large as 60 percent of the total flow energy. Radiant heating rates at the stagnation point were obtained from the analyses of Yoshikawa and Chapman. No allowance was made for non-equilibrium radiation, but it was determined that the values corresponding to the latest data are only a small fraction of the radiation computed by the above method.

It may be inferred from results presented by Myer that the equilibrium radiant heating distribution over the body, normalized relative to the value at the stagnation point, is approximated by the distribution of the square of the normalized pressure. This pressure scaling has been used in determining the radiant heating distribution around the surface of the reentry vehicle.

The stagnation point convective heating rate equation was obtained from Hoshizaki's simplified heating rate relationship for an axisymmetric body. Hoshizaki's equation was developed for velocities up to 50,000 fps. Analyses applicable at higher velocities are not available, and it is judged that Hoshizaki's treatment best lends itself to the rather drastic extrapolation to 70,000 fps.

The convective heating distributions used were those given by Lees. A computational simplification resulted by utilizing an average coefficient of 0.85 to determine the area-integrated heating on the nose region of the vehicle.

In the design of a heat shield for the Earth entry module it was necessary to extrapolate the knowledge of material behavior in lower velocity environments to the conditions of interest for return from Mars. With increasing velocity the relative importance of many factors affecting heat of ablation, such as surface temperature, free stream enthalpy, and radiant heat rates, is changed. Although quantitative data on ablation materials are not available for the very high velocity regimes, approximate design values can be estimated.

As a conservative preliminary design figure, a value of $Q^* = 10,000$ Btu/lb was chosen along with a material density of 120 lb/ft^3. A typical material would be phenolic refrasil. It is recognized that much uncertainty exists regarding this Q^* value, but it represents the best estimate at this time.

Insulation weights were estimated on the basis of an analytical approximation. The total vehicle weights are plotted in Figure 14 as a function of velocity. At the higher velocities the blunt configuration experiences an extremely large amount of radiant heating, and requires greater heat shield weights in this velocity range than does the slender vehicle, even though its ballistic coefficient is always less than that of the slender vehicle.

Mars Entry System Heat Shield Design

A survey of the few available reference works on radiation in a CO_2-N_2 atmosphere revealed that a recent paper by James was most suitable for making quick estimates of radiant heating in the Martian atmosphere. A curve fit to the data presented there, at the CO_2 concentration leading to the maximum radiation (20% by volume), and using the stipulated Martian density profile resulted in the formulation of radiant heating at the stagnation point given by

$$q_{r_o} = 130R \left(\frac{V}{10^4}\right)^{7.4} e^{-\frac{h}{53,000}} \quad (\mathrm{Btu/ft^2\ sec})$$

where

R = nose radius, ft

V = free stream velocity, fps

h = altitude, ft

Pressure scaling, as applied to the Earth entry module, was also used to determine the radiant heating distribution. This pressure scaling is obviously less applicable here than for an air atmosphere, but in view of the lack of other data it does represent the most reasonable assumption that could be made under the circumstances.

The convective heating on the Mars braking vehicle was determined with the STL Aerodynamic Heating Program, which is based on air. However, considering the conclusions of Hoshizaki that the convective heating for CO_2 is close to that for air and considering the fact that the transport properties of nitrogen are close to those for air, the computed results should be sufficiently accurate for this analysis.

Convective and radiant heating rates at the stagnation point were computed for a typical entry. The trajectory corresponds to a lift-to-drag ratio of -0.1 and a ballistic coefficient of 765 lb/ft^2.

The ablation heat shield on the spherical cap and the conical nose section was designed by the same procedure as described for the Earth entry module. Since ablation is primarily due to radiant heating, a Q* of 4,000 Btu/lb has been selected for the computation of the ablated heat shield weight. A typical material is phenolic nylon.

No ablation was predicted for the cylindrical portions and the flare of the vehicle based on the estimated ablation temperature of 4660°R. An insulation thickness of 1-1/2 to 2 inches was calculated to keep the backface temperature below 500°F. The resulting weight was considered to be excessive, leading to the decision to jettison the insulation after the vehicle exits from the Martian atmosphere. The required thicknesses of phenolic nylon to keep the backface temperatures to 500°F at the jettison time are indicated in Figure 15. In the case of the stagnation point and the location on the forecone, the quoted thickness includes the ablated heat shield and 0.6 inch of insulation. A total heat shield weight of 35,020 lb was computed for an entry velocity of 27,500 fps.

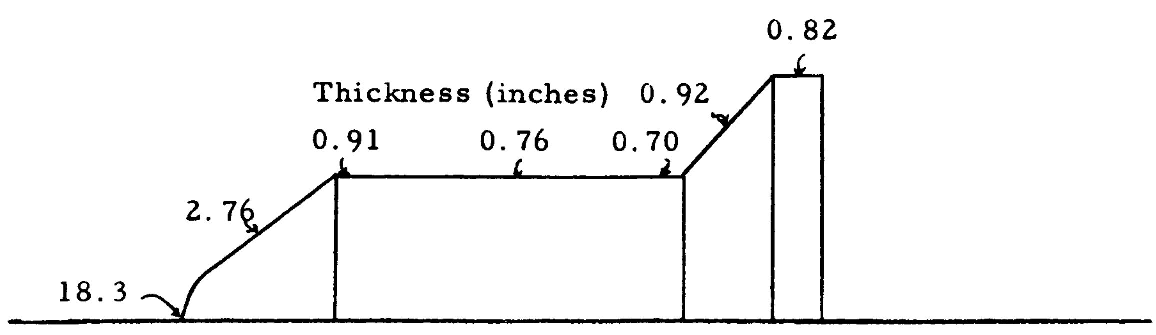


Figure 15. HEAT SHIELD THICKNESS FOR MARS BRAKING VEHICLE (PHENOLIC NYLON)

One operational concept applied to the Manned Mars Missions depends on atmospheric braking at both Mars and Earth. The entry corridor at Mars is currently estimated to be 26 km (27,500 fps) (3 sigma) at an altitude of 90 km. On return to Earth (direct mode), the corridor is about 16 km (70,000 fps) (3 sigma). The constraints on these corridors are due to atmospheric heating, g-loading, atmospheric density variation, etc. Present attempts at meeting these corridor constraints have been limited to the use of onboard navigation systems. Observations of celestial objects are assumed and predictions of perifocal passage at the target planet made. The celestial objects tracked include the sun, Canopus, and the target planets. The angular measurements include those between the celestial objects and those for determination of the angle subtended by one or the other of the planets for purposes of establishing its distance.

In the following sections, the methods of onboard navigation are reviewed, the development of onboard observation error models is described as is the digital computer simulation program. Finally, the results obtained to date are presented.

Review of Interplanetary Navigation

Interplanetary flight requires the early accumulation of data having a high degree of accuracy for prediction of flight path and velocity. Because the requirements on accuracy are great, it is imperative that every source of information be utilized in an optimum manner. During such flights Earth-based tracking should be employed as much as possible, and use made of onboard observation for augmentation. Both automatic and astronaut observations must be considered for the latter.

Onboard navigation requires precision angular measurements between stellar and planetary objects. These measurements are referenced to an inertial measurement unit, a stable platform mounting accelerometers, and gyroscopes, and are processed in a computer employing techniques which enable the determination of the orbit and its predictions at some future time. The onboard observations can be performed automatically or manually. Two types of such observations can be made:

a. An angular measurement between a star and a planet

b. An angular measurement of the angle subtended by a planet.

The types of guidance equipment that can be expected are as follows:

a. Inertial Measurement Unit: A stable platform mounting three accelerometers and, in one configuration, a space sextant. The IMU would be updated by stellar sightings, and would provide the crew with an attitude reference as required.

b. Space Sextant: A two or three telescope sextant capable of subtense and stadiametric measurements.

c. Acquisition Telescope: A wide field of view telescope used for coarse pointing of the sextant.

d. Radio Tracking Subsystems: Subsystems to determine coordinates with respect to the Earth, altitude with respect to Mars and requisite rendezvous and tracking radars operating with the DSIF link as appropriate.

e. Guidance and Navigation Computer: Advanced digital computer using redundancy concepts to achieve reliability.

f. Interface Equipment: Input-output devised for coordinating the operation of the sensor subsystems, the computer and the displays. Interface equipment will also be required for reading in spacecraft status measurements and crew guidance commands. Outputs to the communication system are also desirable.

g. Strapped-down systems may be necessary for use in a possible back-up mode. Such a system would use combinations of body-mounted gyros, accelerometers, horizon sensors, and star, sun and planet trackers.

Onboard Operation Error Model

An important part of a study of interplanetary guidance requiring a high degree of accuracy is the development of a realistic observation error model, which describes statistically both the random and systematic

errors that can be expected with each observation. The observations, of course, are angular measurements of a star or a planet referenced to the IMU or an angular measurement of the angle of subtense of a planet. In an unmanned or automatic system, horizon scanners and star trackers comprise the basic units employed during this operation. The accuracies of these instruments anticipated for the era of 1971 to 1986 are discussed below.

The horizon scanner to be used during this application would operate in the infrared portion of the spectrum, obviating problems associated with partial illumination. A major problem with this instrument is its relative inaccuracy, which is due first to basic instrument uncertainties and second to uncertainties that exist in the planet horizon as seen by the scanner.

One of the advantages of this instrument is its capability of measuring the angle subtense to a planet. If the radius of the planet is known, the distance to the planet can be inferred directly.

The accuracy of altitude information obtainable with this instrument decreases with distance from the planet. With adequate instrument design usable information should be obtainable out to approximately 100 planetary diameters.

Beyond the useful operating distance of a horizon scanner, it becomes necessary to use an instrument similar to a star tracker for planetary observations. This instrument is sensitive to the visible portion of the spectrum. It is anticipated that pointing errors due to partial lighting of the planet can be compensated for in the computer.

The data of Figure 16 were employed to develop the observational error model required by the STL digital simulation, described later.

In addition to planetary measurements, it is necessary to observe stellar objects. For this a star tracker or a sun tracker must be used. When more than one star is to be observed, the star tracker can be programmed from one to another making sequential observations.

Figure 16. Typical 3σ Errors for On-Board Navigation Instruments

Horizon Scanner

Random Error (minutes of arc)		Systematic Error (minutes of arc)	
1. Detector Noise	2.0	1. Detector Circuitry Resolution	1.5
2. Thermal Source Noise	1.5	2. Optical Resolution	1.0
3. Mechanical Noise	2.0	3. Lens Alignment	1.0
4. Horizon Uncertainty	5.0	4. Mechanical Backlash	1.5
		5. Electronics	2.0
		6. Detector Width	3.0
		7. Angular Readout	0.5
TOTAL RSS	6.0	TOTAL RSS	4.5

Star Tracker

Random Error (minutes of arc)		Systematic Error (minutes of arc)	
TOTAL RSS	0.5	1. Optical Resolution	0.1
		2. Detector Circuit Error	0.2
		3. Angular Readout	0.2
		TOTAL RSS	0.3

Sun Seeker

Random Error (minutes of arc)		Systematic Error (minutes of arc)	
TOTAL RSS	0.5	TOTAL RSS	0.3

Tracking Accuracy Prediction Program (TAPP)

TAPP is a high-speed IBM 7090 program designed for the statistical analysis of observations for purposes of orbit determination and prediction. Its major application has been evaluation of Earth-based radio tracking. A capability exists for operating on spacecraft-based celestial observations which was utilized during this study.

The trajectory computation is based on a three-dimensional, multicentered, patched conic model, which shows good agreement with results obtained with exact integrating programs. Operationally the desired inertial and final orbit conditions are inputted and the intervening conics are searched, providing a nominal trajectory and the means for direct computation of all transitional partials.

Also a part of the program input are the observational quantities:

Type of observation
The sigma value of random observational uncertainty
The sigma value of systematic observational uncertainty
Observational sample rate
Times of initial and final observations
The weighting function

Multiple observables can be selected. Random and systematic observational errors on angular measurements were specified. Solving for systematic errors during the course of a run decreased their influence to some extent.

The output of the program consisted of a 6 x 6 covariance matrix positional and velocity uncertainties at the point of closest approach to the planet.

Summary of Results

Simulated flights from Earth to Mars and return have been completed employing the assumed onboard observation models. A 1978 mission was chosen for analysis because of its relatively long duration and high entry velocities.

On both legs of the flight, it was assumed that launch into the interplanetary transfer orbit was made from a circular parking orbit

about the launch planet. The criteria for evaluating the effectiveness of onboard tracking was the crossrange uncertainty at perifocal passage at the target planet. The results of the outbound leg from the Earth to Mars are given in Figure 17. It is seen that shortly after injection into orbit the uncertainty is of the order of 440 km and that improvement is very slow until close approach to Mars. At perifocal passage the uncertainty of 0.91 km is equivalent to an entry corridor of 5.4 km (3 sigma). This is within the expected required corridor based on aerodynamic entry system performance.

Figure 17 also gives the crossrange uncertainties as a function of time during the return flight from Mars to Earth. Because of the lower level of a prior knowledge leaving Mars, the initial uncertainty is greater than for the outbound leg. Again improvement is slow until near the target planet. At perifocal passage the uncertainty is 3.42 km which is equivalent to an entry corridor uncertainty of 20.5 km (3 sigma). This corridor is too wide to meet the aerodynamic and loading constraints anticipated for high speed reentry (70,000 fps). A maximum entry velocity of about 56,000 fps, with atmospheric uncertainties, and 63,000 fps without, can be accommodated. Further investigation was made of the Earth return navigation problem with a view to reducing perifocal passage uncertainties to a level compatible with entry corridor constraints. In addition some improvement was sought in obtaining an accurate prediction in advance of perifocal passage. The benefits gained from early reduction of uncertainty are in reduction of velocity correction requirements. Augmentation of onboard navigation by Earth-based radio guidance was explored. The results are shown in Figure 17.

The improved-accuracy optical onboard observational model was achieved by replacing the sun sensor with a second star sensor, which is basically more accurate. The assumed accuracies are believed achievable with state-of-the-art techniques. The substantial improvement in tracking uncertainty is readily apparent, and is more than adequate to meet the requirements for Earth entry at velocities approaching 70,000 fps.

The use of Earth-based radio tracking (for Earth entry) yields very low tracking uncertainties, assuming availability of the 210-ft DSIF receivers, currently planned for operation before 1970. In addition, the radio tracking technique produces accurate tracking information several days prior to Earth entry, thereby reducing

correction velocity requirements. The radio tracking system will not obviate the need for an onboard optical system, which is required for Mars approach and entry maneuvers, but its use should be given serious consideration for Earth entry, where the need is critical. Corridor depths are given in Figure 18.

Midcourse and Terminal Velocity Correction Requirements

Midcourse and terminal velocity correction requirements have been estimated based on the tracking uncertainty characteristics shown in Figure 17. Conservatively, these requirements are 345 fps for the Earth-to-Mars leg, and 300 fps for the Mars-to-Earth leg (assuming use of the star/sun sensor onboard system). Use of the star/star high-accuracy optical sensor system would reduce the velocity correction requirements slightly.

VEHICLE DESIGN

Vehicle layout drawings were generated to serve as a basis for the weight scaling relations shown in the mission analysis nomographs (Fig. 2), and to gain an insight into the problems of solar radiation protection, artificial gravity, entry vehicle design and crew habitation requirements. The design effort completed to date has been very helpful in determining key problem areas and major trade-offs, and in outlining possible approaches to the resolution of these problems.

Mission Modes

A representative mission mode was selected to serve as a basis for preparing preliminary vehicle designs, and for generating vehicle weight scaling laws. The following mission characteristics were assumed:

Propulsion	Hydrogen-Fluorine
Crew	6
Deceleration at Mars	Aerodynamic
Deceleration at Earth	Aero and Retro/Aero
Mars Landing	MEM (2 man crew)

It is essential to emphasize the prime importance of utilizing atmospheric braking at the target planets, in contract to retro braking, because of the profound effect on spacecraft gross weight, as discussed previously. Spacecraft gross weights in Earth orbit are reduced by a factor of five if aero rather than propulsive braking is used at Mars. Further reductions are achieved if similar techniques are used at Earth entry, because Earth entry velocities tend to range up to 30,000 fps beyond capture velocity. Preliminary study results indicate that aerodynamic deceleration at Mars is feasible with the entry velocities encountered (25,000 to 30,000 fps). Severe radiative heating, due to the presence of CN in the shock, is encountered during Mars entry, and dictates the use of pointed rather than blunt forebody shapes, but overall heat shielding weights are not prohibitive. Accordingly, the aerodynamic capture mode has been assumed for all system design analyses.

System Design

The spacecraft design developed during the study is shown in Figure 19. The vehicle is pointed in shape to reduce the radiant heating experienced during Mars entry. The Earth departure tank is attached aft of the main spacecraft by means of a flared skirt, which serves to decelerate and stabilize the vehicle during Mars entry. The Mars departure tanks are installed in the flare. Hydrogen-fluorine propellants are used in both Earth departure and Mars departure propulsion systems because of the high performance, and high propellant bulk density (compared to that of hydrogen-oxygen propellants).

The main spacecraft consists of a central mission module of 260-inch diameter, which contains the basic crew quarters. The Earth entry module and Mars excursion module are housed in the main spacecraft, and contribute to the functional space requirements. The command module is equipped with guidance and navigation gear, communication links withEarth, and with adequate life support to perform Earth entry and satisfy abort requirements.

Preliminary layouts of the mission module demonstrate the approximate space requirements of the crew and supporting equipments (space requirements must be established on the basis of functional needs and layouts and not on the basis of volume only). For comparison purposes, the spacecraft provides about 650 cubic feet of habitable volume per man. A partially closed ecological system was assumed

(water and atmosphere are processed and reused, but food is used open loop). Results of this and previous studies emphasize the need to integrate cabin environmental control with the electric power and spacecraft thermal control systems.

As noted in previous sections, Earth entry velocities can exceed 65,000 fps, which tends to reduce Earth entry corridors and requires higher L/D capsules. For this reason, it was necessary to consider capsules having L/D's up to 1.0. The L/D = 1.0 capsule is shown in Figure 19. The high L/D capsules pose a problem in cg control because the available volume for crew occupancy is distributed relatively far aft, which tends to move the cg aft also. Heavy equipment must be moved forward where possible to offset the crew weights. Earth entry capsules having L/D capabilities of 0.5 were also considered (the Frontispiece shows an Apollo-type capsule that has been modified to accommodate a six-man crew. This type of module is adequate for a 1971 mission, but is not feasible for other direct return missions requiring higher Earth entry velocities. The module is entirely feasible for Venus swingby return missions, which characteristically have Earth entry velocity comparable to those of Apollo. The Apollo-type entry capsule installed aboard the spacecraft is shown in the Frontispiece.

The Mars Excursion Module (MEM) is designed to land two men on the surface and support their activities for five to six days. Aerodynamic entry and deceleration in the Mars atmosphere was assumed for the MEM. Terminal touchdown was accomplished with a parachute plus retrorocket; parachute settling speeds (assuming the Schilling Model No. 3 atmosphere), are about 40 percent higher than those in an Earth atmosphere, but landing system weight is only slightly greater than that of an Earth system if an optimum distribution of parachute and retrorocket weights are employed. An atmosphere of much lower density (reduction of 5:1) will result in an overall MEM weight increase of about 10 percent. A hydrogen-oxygen propulsion stage returns the MEM to orbit. The external shape is designed to reduce ascent drag and gravity losses, which can be severe (4000 ft/sec). A sketch of the Mars excursion module is given in Figure 19.

Thermal control of the Mars departure tanks can be accomplished with passive insulation techniques with reasonable weight penalties. Refrigerator systems are lower in weight but less reliable. Thermal control of the Mars excursion module while on the Mars surface poses a difficult problem because vacuum insulation techniques lose their effectiveness.

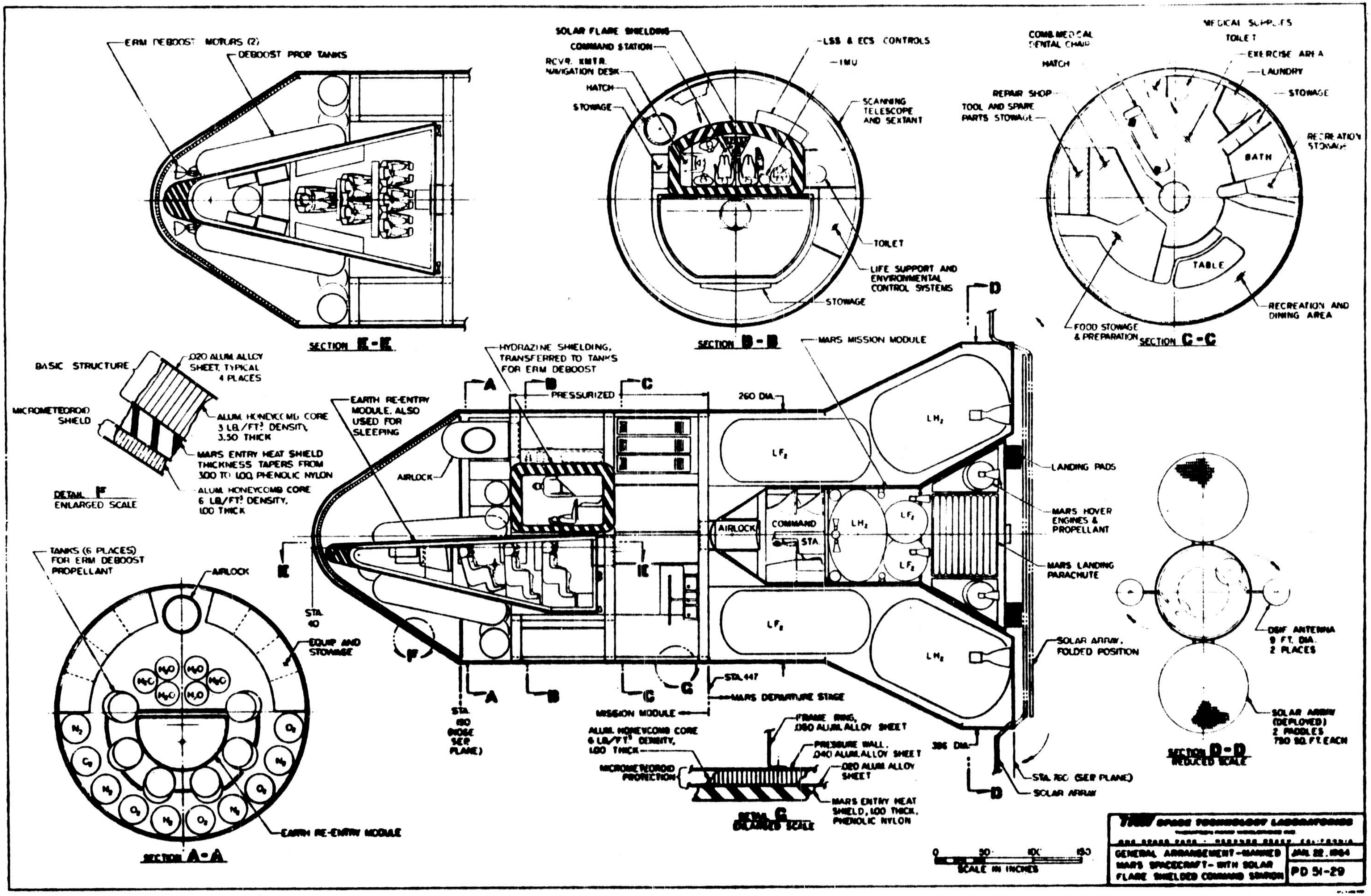


Figure 19. Manned Mars Spacecraft - General Arrangement

Electric power requirements are about five kw, which is in the range suitable for solar dynamic (or solar static) power sources. Solar cell arrays are competitive in terms of weight, and devoid of many operational problems associated with nuclear systems. A fuel cell may be used as a power source during solar occult in the Mars orbit (the water output would contribute to the life support system). The spacecraft design shown in Figure 19 utilizes a solar-static power source, which is folded into the flare during launch and Mars entry.

Solar Radiation Protection

Analyses indicate that about 32 gm/sq cm of solar radiation shielding are required to protect the crew over the year-long mission to Mars. This shielding can be provided in a variety of ways, but all lead to shielding weights of 7 to 9 tons and the problem is to make the best use of this mass. Several approaches to the problem are shown in Figure 20, including use of LSS water, individual water suits, Earth entry retro propellant (N_2H_4), and configurations making maximum use of structure, thermal shielding, and electronic subsystems for protection. Although small differences in total weight are noted, shielding composed of Earth entry retro propellant has certain advantages in that Earth entry velocities can be reduced by about 6,000 fps. A chart showing aerodynamic Earth entry velocities after retro propulsion utilizing the solar radiation shielding N_2H_4 is given in Figure 21. These velocities are approximately Apollo entry system velocities with few exceptions over a 30-year period of opportunities.

Earth Launch and Assembly

The Saturn V can be employed as the Earth launch system by rendezvousing four to six spacecraft elements in the parking orbit. The unfueled spacecraft can be placed in orbit in two shots, and loaded with propellants by subsequent tanker shots. Alternatively, fully-loaded propulsion elements can be brought up and joined with the main spacecraft, thus avoiding the necessity of transferring propellants in orbit. These two concepts are illustrated in Figure 22. Construction in orbit is not required, or desirable: rendezvous and docking procedures only need be involved. The nuclear-hydrogen configurations pose some difficulty because of the large bulk of the low-density propellant.

Relatively low thrust engines can be used with small gravity losses in this application because departure from both planets is made from parking orbits. Multiple J-2's and RL110A's are well-suited for Earth departure and Mars departure, respectively. Conversion to hydrogen-fluorine propellants may be desirable.

Artificial Gravity Provisions

A brief analysis was made of an artificial-gravity version of the manned Mars spacecraft. In this version spent propellant tanks are suspended from the main spacecraft by means of a single cable for use as counterweights. A gravity level of 1/6 g was provided. This configuration was verified for dynamic and long term stability on the Langley nine-degree-of-freedom computer program developed for space station analyses for Langley. The results indicated that no stability problems will be encountered with the STL configuration if suitable damping is provided at the cable attach points on the propellant tanks. Attitude control jets must be provided on both masses. A sketch of the system is given in Figure 23.

SUBSYSTEMS

Life Support and Environmental Control

Independent life support subsystems were considered for each of the three Mars vehicle modules, the Mars mission module, MMM, the Mars excursion module, MEM, and the Earth entry module, EEM. The EEM subsystem was designed to meet the needs of the total 3- to 10-man crew for a period up to 10 days. The MMM subsystem provided for the total crew requirements over the cruise phases of the mission (400 days) while the MEM subsystem supports only a part of the total crew during a Mars surface excursion of up to 40 days.

It seems clear from these requirements that the life support subsystems for the MEM and EEM will utilize expendable stores. Because of the long duration of the cruise phases of the mission, however, the MMM subsystem may well regenerate both oxygen and potable water from metabolic waste.

Waste reclamation processes have the effect of producing process loops within the life support subsystem and hence tend to make all the system processes interdependent. This point is amply illustrated by Figure 24 in which process interrelationships are presented for an integrated MMM system.

ATMOSPHERIC COMPOSITION CONTROL PROCESSES

The integrated life support subsystem presented in Figure 24 can be used as a basis for discussing atmospheric composition control processes which will be considered during subsystem integration. General features of this system are the following:

a. Provision for an N-man crew with a mission duration of τ days. The daily material balance for the system is indicated at pertinent points throughout the diagram.

b. A two component atmosphere, 52% O_2, 44% N_2 with a total pressure of 7 psia.

c. Control of CO_2 by conversion to H_2O and methane.

d. Control of cabin relative humidity to approximately 60% at a temperature of 70°F. The water removed from the atmosphere is purified and reused.

e. Water reclamation from utility water and urine by chemical treatment and vapor compression distillation.

f. Generation of the major part of the required oxygen by electrolysis of water.

g. Control of atmosphere contaminants of higher molecular weight by adsorption on activated carbon.

h. Control of lower weight atmospheric contaminants by catalyzed burning.

i. Stored oxygen and nitrogen to make up cabin leakage losses as well as to provide ten complete changes of the vehicles atmosphere.

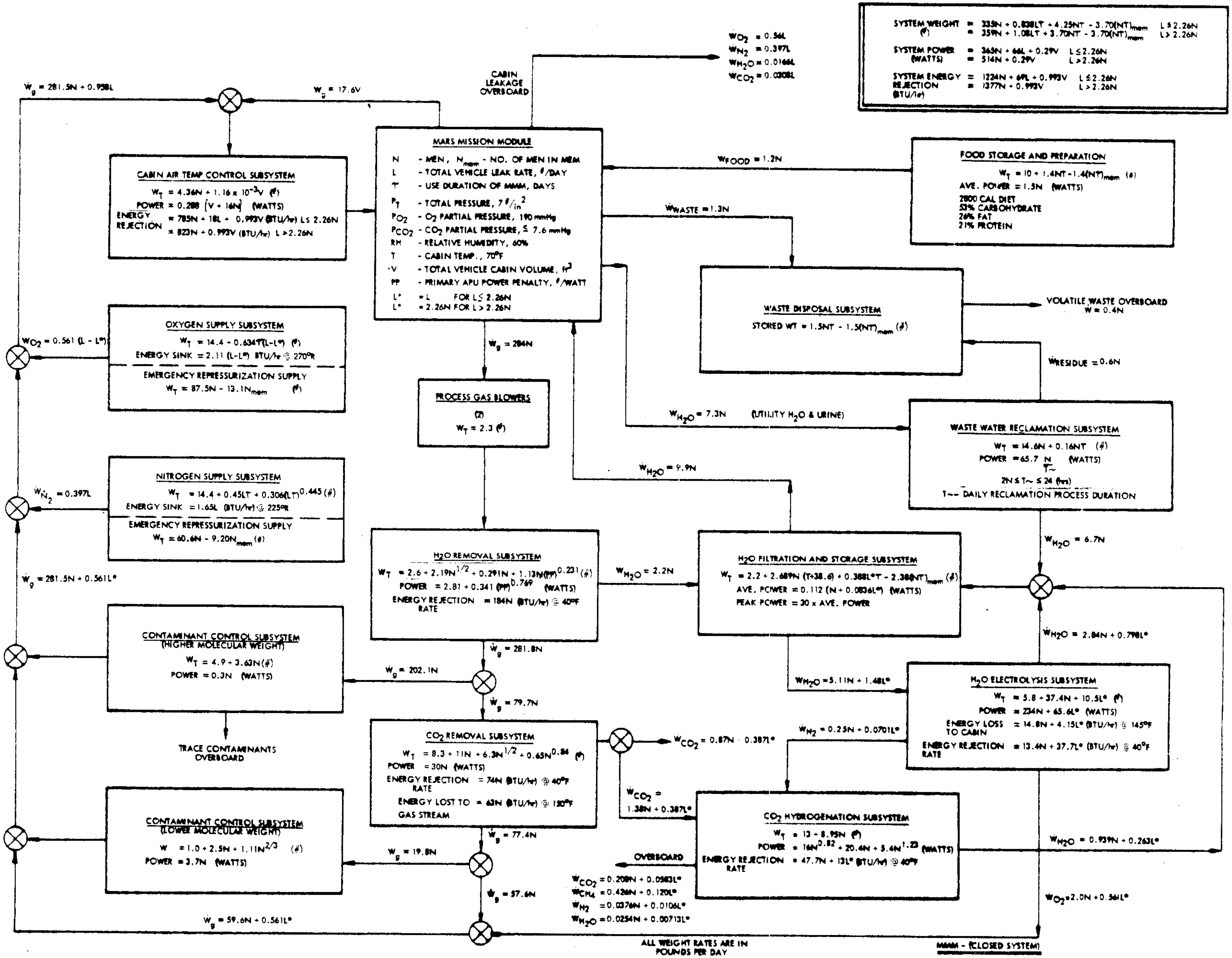


Figure 24. LSS Schematic

Communications

For information transmission from the MMM to Earth during trans-Mars, trans-Earth, and near-Mars phases of the mission, the following performance criteria were assumed:

a. The radio terminals on Earth are DSIF-stations with 210-ft antennas. The receivers operate at a DSIF channel in the frequency band 2295 ±mc. The receiver system noise temperature is 100°K, which includes all sources of internal and external noise; the receiver IF bandwidth is 3 mc.

b. The communication range extends to about 1.0 AU.

c. During near-Mars phases of the mission, the communication range does not exceed 0.5 AU.

d. The spacecraft radio terminal must be able to transmit at either of two RF power levels.

e. When transmitting at low-power, the spacecraft equipment must be capable of continuously supporting angular tracking and range-rate (two-way doppler shift) measurements. The maximum duty-cycle for low-power transmission is 100 percent.

f. When transmitting at high-power the spacecraft equipment must be capable of transmitting either real-time television simultaneously with low-rate telemetry, or one speech channel and medium-rate telemetry, or high-rate telemetry only. During this transmission, the capability of performing angular tracking and two-way doppler measurements must be maintained. The duty cycle of high-power transmission is 10 percent during coast, and 80 percent about Mars.

g. The television channel must be capable of transmitting one 250-line frame per second during near-Mars phases of the mission (0.5 AU) and every 5 seconds at ranges up to 1.0 AU.

The design concept derived includes a transmitter output power of 500 watts in the high-power mode of transmission and 10 watts in the low-power mode. The output power of the high-power transmitter can be raised to 1000 watts for short periods. The carrier is phase-modulated either directly by digital data or by two sinusoidal subcarriers.

The directional antenna at the MMM has a diameter of 12 feet. Two such antennas provide adequate coverage for all relevant spacecraft orientations. In mode (a), the video signal from the television camera directly frequency-modulates a television subcarrier while the digital telemetry bi-phase modulates the telemetry subcarrier. Both subcarriers phase-modulate the RF carrier. At the DSIF receiver coherent demodulation is used for the carrier and for both subcarriers. The carrier modulation indices for television and telemetry are chosen such that the telemetry transmission causes a 20 percent reduction in the television transmission rate. A television video bandwidth of 9.0 kc is obtained. This permits readout of a 250 line television frame in 3.4 seconds. Camera tubes for this relatively slow readout rate require an erasure phase of about 1.5 seconds after each readout, which increases the total frame period to 5 seconds. The analysis further shows that telemetry at a rate of 2000 bits/second can be transmitted simultaneously with the television. This is based on a bit-error probability of 10^{-3}.

In mode (b), the analog speech signal directly frequency modulates the speech subcarrier, while the digital telemetry bi-phase modulates the telemetry subcarrier. Both subcarriers then phase modulate the RF carrier. At the DSIF receiver coherent demodulation is used for the carrier and both subcarriers.

The telemetry rate is 20,000 bits/second with a bit-error probability of 10^{-3}.

In mode (c) only digital telemetry is transmitted. The digital data directly phase-shift-keys the carrier with a deviation of about ±70 degrees. This phase-deviation is slightly less than in the bi-phase modulation (±90 degrees) and, therefore, results in a ten percent reduction in the data rate. This reduction in communication efficiency is the penalty for maintaining sufficient power in the carrier to assure phase-lock in the carrier tracking loop at the DSIF receivers on Earth.

At the DSIF receiver the digital data are recovered by coherent demodulation of the carrier. This method, i.e., pseudo bi-phase modulation with coherent detection yields better communication efficiency than all other known practical methods. A bit-rate of 90,000 bits/second with a bit-error probability of 10^{-3} is feasible.

For television, the transmission rate near Mars results in a frame-rate of 1 frame per second. The capability to transmit at double power, i.e., at 1000 watts, during shorter lengths of time, permits further increase to 2 frames/second. For critical mission phases, such as docking of the MEM with MMM, 8 frames per second is feasible if the picture resolution can be degraded by changing from 250-line frames to 125-line frames.

Thus, a variety of frame rates, from 1 frame every 5 seconds (500 watts, 1 AU, 250 lines) to 8 frames per second (1000 watts, 0.5 AU, 125 lines) can be made available.

Power Subsystem

The power system for the MMM must provide power in the kilowatt range over long periods of time at a low specific weight and with good reliability. This represents one of the important subsystem design problems because of its effect on the overall system. For example, systems which utilize solar energy as the basic energy source may impose difficult orientation requirements on the total system while systems which utilize nuclear energy impose radiation shielding requirements. The tradeoff between dynamic and static energy conversion is equally important because of reliability and maintenance considerations.

POWER REQUIREMENTS

Careful analysis and control of subsystem power requirements is of paramount importance in establishing the design criteria for space power systems.

The major continuous power consumer is the life support system, which requires 2.6 kw for a 6-man crew, assuming a closed system (water and atmosphere regenerated). The other major power consumer is the communications subsystem, which requires over 2 kw for television transmission from Mars to Earth (30 percent duty cycle). This estimate is believed to be realistic for foreseeable communication techniques. Total power required is not excessive: 5 kw continuous.

APPLICABLE POWER SYSTEMS

The systems which must be analyzed in detail for applicability to the MMM include both static and dynamic systems utilizing either nuclear or solar energy. Previous STL studies have shown that a dynamic system shows the most promise for this application, but this depends very much on the power level required and on predictions of hardware development over the next 10 to 15 years. In fact, the difficulties which have shown up in the development of dynamic systems have led to intensified interest in thermionic converters utilizing a radioisotope as the heat source. Within 10 years or so this may be possible using an alpha emitter. (The AEC is predicting that by 1970 about 120 thermal kw per year of curium will be available.)

Nuclear power plants utilizing dynamic energy conversion are currently being developed for electrical power levels ranging from 3 to 60 kwe; SNAP-2 and SNAP-8 are the prime examples of such power supplies. At present, no dynamic conversion power plant is under development using a radioisotope heat source, although several direct-conversion, radioisotope-powered generators are being developed for electrical power levels below 100 watts. However, generators of up to 1-kw rating are in preliminary design.

The use of a solar-dynamic power system for the MMM involves many of the same considerations as the nuclear-dynamic system, particularly in regards to the conversion machinery. Its use will result in a lighter system because the radiation shielding problem is eliminated, but it has its own problems in that the collectors must always be oriented toward the sun. This may be particularly difficult for an artificial gravity type configuration. Also, allowances must be made for the changes in the incident energy, e.g., from 130 watts per square foot at Earth to approximately 56 watts per square foot at Mars as well as additional heat input for a mission whose perihelion is less than 1 AU. The possibility of an eclipsing orbit at Mars also complicates this approach.

To compensate for both near-Earth and near-Mars eclipse effects, lithium hydride can be used as a heat source. This material will be heated and melted while solar illuminated. The heat that is stored can then be used for eclipse operation.

Assuming an efficiency of 28 percent, two solar dynamic units will provide the required power output. A weight of 430 pounds and a collector diameter of 29 feet for each of the two units is indicated. To these weights must be added allowances for batteries, cabling, conversion equipment, and emergency power units. These are estimated to weigh 750 pounds, giving a total power system weight of 1180 pounds.

SOLAR PHOTOVOLTAIC POWER

Power systems utilizing conventional solar cells must be considered for this mission because of their known availability and demonstrated reliability. Fuel cell batteries would be included in the system for energy storage during eclipse. These are possibilities for marked improvements in solar cells (for example, thin film techniques) by the time a manned Mars mission takes place, but these developments are difficult to predict. The weight of the solar photovoltaic power system is competitive with that of previously discussed systems if the weight of the fuel cell energy storage unit is charged to the LSS water management unit.

SURFACE EXPLORATION OF MARS

A review of past and current studies on the Manned Mars Mission reveals that the operations and functions of the manned Mars landing expedition have received little attention, whereas such considerations should accompany the overall mission studies. The objectives to be accomplished and the associated systems required for the exploration of the surface of Mars have a predominant effect on overall planning, from the standpoint of weight in Earth orbit, spacecraft design (for injection and recovery of the lander module), crew size, stay time, and orbital operations in the vicinity of Mars.

Appraisal of the various techniques for the scientific exploration of the Mars surface must be studied. Man has certain unique capabilities which permit him to extend and exploit precursor missions such as Mariner and Voyager. These capabilities relate to man's ability to perform more accurate and more perceptive reconnaissance than would be possible from unmanned observatories. From a low altitude orbit

about Mars prior to landing, man can make more detailed visual reconnaissance of the surface and apply well based judgment to the selection of landing sites. In addition, man's capability to perform reconnaissance on the surface after landing cannot be duplicated by unmanned probes.

In general, man will play a much more dominant role in the exploration of Mars than in the exploration of the Moon because the planet is more complex and more remote, and is less amenable to comprehensive exploration by unmanned systems. Because of the total cost of the expedition, and because of the infrequent opportunities, man must be fully equipped and prepared to exploit his unique capabilities.

The purpose of manned operations is to acquire useful information about Mars, to enlarge the domain of discovery as far as practical, and to improve the intelligence available to subsequent expeditions. To accomplish this purpose in the most efficacious manner, manned landing parties must be equipped with instruments to perform scientific measurements while on the surface to augment and supplement visual reconnaissance. Results of preliminary studies consistently indicate that instrument weights and supporting power, communications, and telemetry are not large--of the order of a few hundred pounds. In spite of the low weights these observatories or instrument payloads are extremely versatile and efficient in gathering large quantities of information. For example, the advanced OSO can gather and transmit 60 billion bits of information within the first six months of orbital operation. This suggests that landing parties can be furnished with well equipped instrument payloads without large expenditures of weight; but the early visualization of such equipment and modes of operation for the lander party needs to be accomplished.

A possibly more significant consideration from the standpoint of overall system weight is the necessity to provide landing parties with the means for surveying possible sites for the placement of automatic observatory stations, and for means of transporting the men and equipment to these sites. Furnishing the landing parties with ground or airborne transport vehicle must be considered. Given the capability to perform reconnaissance, and the capability to situate remote observatories at desired sites, man should then manage these test stations initially to attain maximum efficiency of operation. For example, the stations can be transported and installed at the most favorable sites and after initial readings, adjustments can be made to favor the most important instruments.

Collecting and preparing samples for return will comprise an important function of the expedition. The crew should have access to a laboratory at the main base for performing essential experiments and observations (analysis of surface material, gathering and analyzing biological specimens, etc.), to reduce the bulk and weight of material to be transported back to Earth. Because of the desire to maintain the planet as an uncontaminated test site area, there are many restrictions imposed on unmanned missions such as Voyager, which tend to limit their ability to perform many facets of the scientific mission. Hence, it remains for the early manned landing parties to fully exploit the scientific exploration of Mars. A possible surface operation scheme is shown in Figure 25.

CREW UTILIZATION

A brief human factors study was made to estimate crew requirements for the mission and to determine the long range research necessary for accomplishment of the mission in the 1971 to 1986 time period.

To arrive at crew size requirements, a task analysis of the mission was conducted. The crew's role in the mission will be to increase reliability by acting as a control element on a normal or backup basis, and to perform in-flight maintenance. The crew will also increase the probability of mission success by utilization of judgement and decision-making ability to manage the mission and be selective in the collection of scientific data. The task analysis indicated that a crew of eight men is desirable and seven is a minimum. Substantial improvements in subsystem reliability and maintainability could allow a reduction in crew requirements.

The major research task is to determine the effects of extended periods of sub-gravity or zero gravity on crew performance and physiology. An extensive research program utilizing orbital research laboratories is required. Considerable research is also required on the effects of multiple stresses on crew performance and physiology. Confinement and boredom could be a problem but the effects can be minimized by crew selection and motivation, proper design of crew tasks, and onboard training program. Research is required in the areas of human reliability, optimization of trouble shooting and

maintenance procedures, prolonged confinement, manual control, work-rest cycles, controls and displays, and task definition and design but on a less intensive basis than is required for sub-gravity and multiple stresses.

CONCLUSIONS

Based on the mission analysis study, the following conclusions can be stated:

System Trade-offs

1. Unfavorable years increase spacecraft weight 35%
2. Launch holds increase weight 7%
3. Launch hold plus 30-day stopover increases weight 18%
4. Nuclear Earth depart reduces weight 40%
5. Retro deceleration at Mars increases weight 500%

Reduced Earth Entry Velocities

1. Retro braking to 60k increases weight 13%
2. Off optimum to 60k increases weight 24%
3. Retro braking to 50k increases weight 43%

Venus Swingby Modes

1. Reduces Earth entry velocities to under 50k
2. Available in all opportunities

Navigation

1. Earth corridors for entry velocities of 70k can be provided with DSIF tracking on onboard star/star sensors

2. Mars corridors for nominal range of entry velocities can be provided with onboard optical system.

APPENDIX A

ANALYTICAL MISSION OPTIMIZATION TECHNIQUES

Spacecraft traveling to and from Mars on stopover missions have a wide choice of paths and mission modes available to them. Each combination of calendar date of departure and trip duration (inbound and outbound) requires a different transfer trajectory, which has unique departure and arrival velocities. These combinations of velocities determine the spacecraft aerodynamic and propulsion system performance requirements. To select an optimum mission profile, which usually implies a minimum spacecraft gross weight, it is necessary to trade off the effects of changes in mission characteristic velocities, or trajectories, with resultant changes in spacecraft weight.

Previous studies of manned interplanetary missions have approached the optimization problem by repeated calculations of the overall mission performance equations, using suitable spacecraft weight scaling laws.

The optimization (to find minimum spacecraft weight) was performed by first constraining the overall trip duration to a selected value, and searching for the best path within this constraint, first by searching along lines of constant leg duration to find the best Earth launch date for that given leg duration; and second, searching along the curve of spacecraft weight versus leg duration to find the overall minimum weight. The process was then repeated for other values of overall trip duration until the best overall trip duration was determined. This approach to the optimization process is rather lengthy because of the large number of repeated calculations of the overall mission performance equations. The number of calculations becomes very large if complexities such as launch hold allowances are introduced.

A basically different approach to the optimization of manned interplanetary missions is presented in this note. An analytical technique has been developed which selects the launch dates and transfer trajectories that minimize spacecraft gross weight. Use is made of differential equations, which contain vehicle weight-velocity exchange ratios and derivatives of mission characteristic velocities with respect to launch date and trip duration. The process does not require total trip duration constraints (although these can be introduced

if desired) and has been generalized to include launch hold allowances and several aerodynamic-propulsion mode combinations. Arbitrary dwell time at the target planet is permitted.

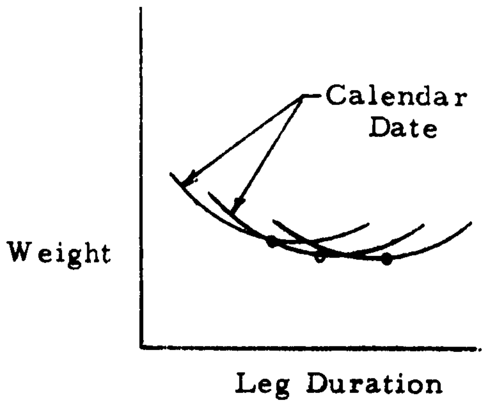
The overall optimization process is divided into two phases: first, the solution for best outbound (inbound) leg durations for selected calendar dates, and second, the solution for the best calendar date. These phases are described in the following sections.

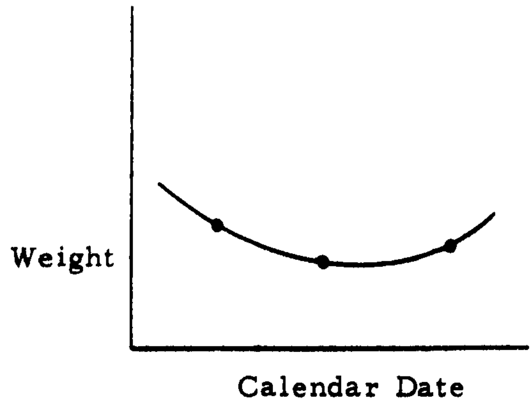
The technique has been developed for a variety of mission modes including aerodynamic and/or propulsive capture at the target planet, jettison phases, total trip duration constraints, and the effects of variable thrust-to-weight ratio and midcourse correction. Provisions for launch holds require interior search routines within the general equations to assure adequate propellant tank volumes for all contingencies. The suboptimization procedure (best leg durations for given calendar dates) is used to find the least penalty for each launch within the hold period.

Suboptimization for Best Leg Durations

It has been found advantageous to graph the mission characteristic velocities as shown in Figure 3, with mission velocity requirements plotted versus calendar date at Mars for various leg durations. This conveniently separates the mission into the outbound and inbound phases, which can be optimized independently and subsequently pieced together to find the overall optimum combinations. This is possible when no constraint on total trip duration has been introduced. Another advantage of this method of presentation is readily apparent: near-optimum leg durations can be determined by inspection since they tend to occur near or at the lower envelope of the velocity curves (particularly if aerodynamic capture is used at the target planet).

The suboptimization procedure determines the best outbound (or inbound) leg durations for given calendar dates of arrival (or departure) at the target planet. The process is shown schematically below.





Consider, for example, an outbound leg time optimization for a given calendar date. The significant characteristic mission velocities are the Earth departure velocity and the planet arrival velocity, both of which depends upon leg time. The optimization is initiated by graphing the characteristic mission velocities versus leg time for several calendar dates of arrival (departure) at Mars. The optimum leg time can be found by the following relations. Assuming that the weight of the spacecraft is a function of Earth departure ΔV and Mars arrival V, the total derivative of spacecraft weight with respect to leg time, T is

$$\frac{dW}{dT} = \frac{\partial W}{\partial \Delta V}_{V} \frac{d\Delta V}{dT} + \frac{\partial W}{\partial V}_{\Delta V} \frac{dV}{dT} \tag{1}$$

For minimum weight $dW/dT = 0$, and

$$\frac{\partial W}{\partial \Delta V}_{V} \frac{d\Delta V}{dT} = - \frac{\partial W}{\partial V}_{\Delta V} \frac{dV}{dT} \tag{2}$$

The total derivatives of equation 2 can be obtained directly from the slopes of the curves of velocity versus leg time. The coefficients of these derivatives are exchange ratios based on the weight scaling laws used in the mission analyses computations. These coefficients generally are functions of the characteristic mission velocities, and of the weight of the spacecraft. However, for most optimizations these coefficients vary slowly in the neighborhood of the optimum. The coefficients for the derivatives can be computed at a near optimum point and held constant for the optimization process with negligible error.

Weight scaling laws based on STL spacecraft designs were used to derive suitable analytical expressions for the coefficients or exchange ratios. The appropriate weight scaling law for the Earth departure propulsion phase is:

$$\frac{W_o}{W_p} = \frac{r(1-\sigma)}{1-r\sigma} \tag{3}$$

where

$r = \exp\ (\Delta V/I_{sp}g)$

σ = ratio of structural weight to stage weight

I_{sp} = specific impulse

W_p = payload weight (gross weight less stage weight)

W_o = gross weight (stage weight plus payload weight)

If the structural factor is held constant, the above expression can be differentiated and rearranged to give the required exchange ratio as

$$\frac{\partial W_o/W_p}{\partial \Delta V} = \frac{r(1-\sigma)}{(1-r\sigma)^2}\ \frac{1}{I_{sp}\ g} \tag{4}$$

Similarly, the weight scaling law for the Mars arrival aerodynamic deceleration phase is:

$$\frac{W_o}{W_p} = C_2\ \left(\frac{v}{1000} - C_3\right)^m \tag{5}$$

The exchange ratio is:

$$\frac{\partial W_o/W_p}{\partial V} = \frac{MC_2}{1000}\ \left(\frac{V}{1000}\ -C_3\right)^{m-1}$$

Equations 4 and 6 can be substituted into an equation of the form of equation 1 for evaluating the optimum. First it is necessary to expand equation 1 into a form suitable for the two-phase operation. Assume the following subscript definitions:

1 = condition at Earth orbit before injection

2 = condition after departure from Earth orbit (Earth departure stage jettisoned)

3 = condition after deceleration at the target planet (entry system stage jettisoned)

It is noted that the spacecraft weight after capture, W_3, is a constant throughout the optimization procedure. Further, it is convenient to deal in terms of weight ratios, in particular, W_1/W_3. Equation 1 becomes

$$\frac{d\, W_1/W_3}{dT} = \frac{\partial(W_1/W_3)}{\partial \Delta V}\frac{d\Delta V}{dT} + \frac{\partial(W_1/W_3)}{\partial V}\frac{dV}{dT}$$

$$= \frac{\partial(W_1/W_2 \quad W_2/W_3)}{\partial \Delta V}\frac{d\Delta V}{dT} + \frac{\partial(W_1/W_2 \quad W_2/W_3)}{\partial V}\frac{dV}{dT}$$

In the first term on the right, W_2/W_3 remains constant during the Earth entry phase, and if it is assumed that the partial derivative of W_1/W_2 with V is approximately zero (since ΔV is constant), equation 7 becomes:

$$\frac{d\ W_1/W_3}{dT} = \left(\frac{W_2}{W_3}\right) \frac{\partial W_1/W_2}{\partial \Delta V} \frac{d\Delta V}{dT} + \left(\frac{W_1}{W_2}\right) \frac{\partial W_2/W_3}{\partial V} \frac{dV}{dT} \tag{7}$$

It now remains to insert the proper expressions for the coefficients and weight ratios into equation 7 to obtain the final relations. The expressions for phase (1-2) are based on the propulsion weight scaling laws (eq. 3 and 4), and the expressions for phase (2-3) are based on the aerodynamic deceleration weight scaling laws (eq. 5 and 6). Combining these equations and setting the derivative of weight ratio with time equal to zero gives:

$$\frac{d\Delta V_{12}}{dT} = -\frac{(1-r\sigma)^2}{1-r_o\sigma_o} \left[\frac{r_o(1-\sigma_o)}{r(1-\sigma)}\right] \left[\frac{m\left(\frac{V}{1000} - C_3\right)^{m-1}}{\left(\frac{V_o}{1000} - C_3\right)^m}\right] \frac{I_{sp_{12}}\, g}{1000} \frac{dV}{dT} \tag{8}$$

The subscript zero indicates nominal values of terms in the exchange ratios for the non-varying phase. If it is assumed that the coefficients of the varying phase remain constant as well, equation 1 reduces to

$$\frac{d\Delta V_{12}}{dT} = -\left(\frac{mg\ I_{sp_{12}}}{1000}\right) \left[\frac{1-r_o\sigma_o}{\left(\frac{V_o}{1000} - C_3\right)}\right] \frac{dV}{dT} \tag{9}$$

Similar expressions can be derived if propulsive deceleration is assumed.

A midcourse maneuver between conditions (2) and (3) does not alter equations 8 and 9 if the midcourse maneuver exchange ratio remains constant during the optimization process.

The inbound trip usually involves jettisoning the mission module and life support system prior to entry into the Earth's atmosphere. Assuming atmospheric braking, equations 8 and 9 can be modified to accommodate the jettisoning phase. Using the subscript MM to denote the mission module with life support system, the appropriate expressions are given below, assuming constant coefficients for each phase.

The slopes of the velocity - time curves are very nearly straight line functions of leg time over the range of interest, which indicates that the velocity - time relations can be approximated with second-order polynomials, differentiated analytically and inserted into equation 9 so that a closed form solution can be obtained for the optimum. Graphical solutions are thus avoided. Equation 9 takes the following form: Let

$$\frac{d\Delta V_{12}}{dT} = A + BT, \quad \frac{dV}{dT} = C + DT$$

then

$$A + BT = - \frac{mgI_{sp_{12}}}{1000} \left[\frac{1 - r_o \sigma_o}{\left(\frac{V_o}{1000} - C_3\right)} \right] (C + DT) = -\alpha(C + DT) \tag{10}$$

$$T_{OPT} = T^* = - \frac{A + \alpha C}{B + \alpha D} \tag{11}$$

α's are given for various mission modes in Table 1.

Table 1. α - FUNCTIONS FOR SUBOPTIMIZATIONS

Leg		Mode	α
1.	Outbound	Aerodynamic deceleration.	$\left(\frac{mg\, I_{sp_{12}}}{1000}\right)\left[\frac{1 - r_{12_o}\sigma_{12_o}}{\frac{V_{23_o}}{1000} - C_3}\right]$
2.	Outbound	Propulsion deceleration.	$\left(\frac{I_{sp_{12}}}{I_{sp_{23}}}\right)\left[\frac{1 - V_{12_o}\sigma_{12_o}}{1 - V_{23_o}\sigma_{23_o}}\right]$
3.	Inbound	Aerodynamic deceleration. Jettison mission module before entry.	$\left(\frac{mg\, I_{sp_{45}}}{V_{67} - C_5}\right)\left[\frac{1 - r_{45_o}\sigma_{45_o}}{\frac{1 + W_{MM}/W_7}{C_4(V_{67_o} - C_5)^m} + 1}\right]$
4.	Inbound	Propulsion braking to V_{67A}. Aerodvnamic braking below V_{67A}. Jettison mission module before retro.	

$$\left[\frac{1}{\frac{1 - V_{67_o}\sigma_{67_o}}{V_{67_o}(1 - \sigma_{67_o})}\;\frac{W_{MM}/W_7}{1 + \frac{C4}{W_7}(V_{67A} - C_5)^m} + 1}\right]\left(\frac{1 - r_{45_o}\sigma_{45_o}}{1 - r_{67_o}\sigma_{67_o}}\right)\left(\frac{I_{sp_{45}}}{I_{sp_{67}}}\right)$$

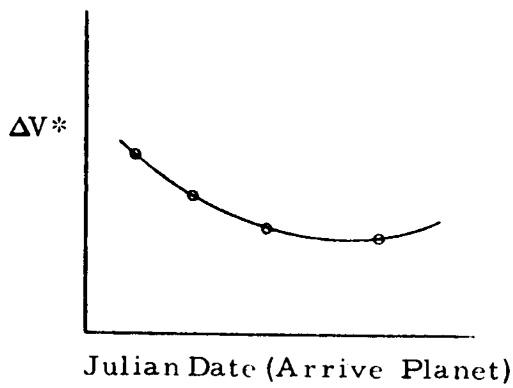
Subscript	Condition
1	Earth orbit before departure
2	After injection toward Mars. Earth departure tanks jettisoned.
3	After capture at Mars. Mars deceleration stage jettisoned.
4	After capture at Mars. Mars landers jettisoned.
5	After injection toward Earth. Mars departure tanks jettisoned.
6	Earth approach. Mission module jettisoned.
7	After Earth entry. Earth entry stage jettisoned.

General Optimization

In order to extend this technique to find the overall best Julian data, a relation between characteristic mission velocities and Julian date is required. The characteristic mission velocities corresponding to the optimum leg times of equation 11 are:

$$\Delta V^* = \Delta V_o + A\,(T^* - T_o) + \frac{B}{2}\,(T^{*2} - T_o^{\,2}) \tag{12}$$

where the subscript o refers to nominal conditions. It is now possible to construct single curves of ΔV^* versus Julian date, along which leg time varies in an optimum manner. Similar single curves exist for each phase of the mission. It remains to find the best overall Julian date using these suboptimum characteristic velocities. This is accomplished by solving the following expression:



$$\frac{dW_1/W_7}{dT_{AM}} = 0$$

where W_1/W_7 is the spacecraft weight ratio taken between space departure and Payload returned to Earth (AM refers to condition at Mars). The total derivative can be written as:

$$\frac{d\,W_1/W_7}{d\,T_{AP}} = \frac{1}{(W_1/W_2)_o}\,\frac{\partial W_1/W_2}{\partial \Delta V_{12}}\,\frac{d\Delta V_{12}^*}{d\,T_{AM}} + \frac{1}{(W_2/W_3)_o}\,\frac{\partial W_2/W_3}{\partial \Delta V_{23}}\,\frac{d\Delta V_2^*}{d\,T_{AM}}$$

$$+ \frac{1}{(W_3/W_5)_o}\,\frac{\partial W_o/W_s}{\partial \Delta V_{45}}\,\frac{d\Delta V^*_{45}}{d\,T_{AM}} + \frac{1}{(W_3/W_7)_o}\,\frac{\partial W_3/W_7}{\partial V_{67}}\,\frac{dV^*_{67}}{d\,T_{AM}} \tag{13}$$

It is assumed that the slopes of the velocity-Julian date curves can be approximated over the range of interest by straight-line functions, as before. It is necessary to distinguish between an outbound and inbound leg in equation 12 because of the dwell time at Mars. If Mars arrival date is used as the independent variable, the Julian dates for the inbound leg will be equal to the arrival dates plus the dwell time. Hence for the Mars departure phase (4-5),

$$\frac{d\Delta V_{45}^*}{d\,T_{AM}} = A_{45} + B_{45}\,(T_{AM} + T_o) \qquad (14)$$

The above relations, and the appropriate vehicle exchange ratios (see equations 3-6), are inserted into equation 13 to obtain a solution for the best Julian date. Setting

$$\frac{dW_1/W_7}{d\,T_{AM}} = 0, \text{ and solving for } T_{AM}^* \text{ gives:}$$

$$T_{AM}^* = -\frac{\beta_{12}\,A_{12}+\beta_{23}\,A_{23}+\beta_{45}(A_{45}+B_{45}\,T_P)+\beta_{67}(A_{67}+B_{67}\,T_o)}{\beta_{12}\,B_{12}+\beta_{23}\,B_{23}+\beta_{45}\,B_{45}+\beta_{67}\,B_{67}} \qquad (15)$$

Expressions for β's are given in Table 2.

Launch Holds at Earth Departure

Provisions for the launch holds at Earth departure necessitate a variation in the final optimization procedure because one specific vehicle design no longer follows a unique path to and from the target planet. A vehicle may be launched on a nominal departure date, T_{LE_N}, or on any day during a succeeding hold period, T_H. The vehicle must be able to perform the mission over the range of launch dates. The problem, then, is to find the best path for each launch day within the hold period, and to design the vehicle to perform the mission with the least penalty in launch weight compared to the no-hold launch weight.

Table 2

β - FUNCTIONS FOR FINAL OPTIMIZATIONS

1. Aerodynamic Braking at Mars and Earth

$$B_{12} \qquad \frac{1}{I_{sp_{12}}\, g\, (1 - r_{12_o}\, \sigma_{12_o})}$$

$$B_{23} \qquad \frac{m}{1000\left(\frac{V_{23_o}}{1000} - C_3\right)}$$

$$B_{45} \qquad \frac{1}{I_{sp_{45}}\, g\, (1 - r_{45_o}\, \sigma_{45_o})} \left[\frac{1}{1 + \frac{1 - r_{45_o}\, \sigma_{45_o}}{r_{45_o}(1 - \sigma_{45_o})} \quad \frac{W_{LDG}/W_7}{1 + \frac{W_{mm}}{W_7} + \frac{C_4}{W_7}\,(V_{67_o} - C_5)^m}}\right]$$

$$B_{67} \qquad \frac{m\, \frac{C_4}{W_7}\,(V_{67_o} - C_5)^{m-1}}{\left[1 + \frac{W_{mm}}{W_7} + \frac{C_4}{W_7}(V_{67_o} - C_5)^m\right] + \left[\frac{1 - V_{45_o}\, \sigma_{45_o}}{V_{45_o}(1 - \sigma_{45_o})}\right] \frac{W_{LDG}}{W_7}}$$

2. Propulsion Braking at Mars, Aerodynamic Braking at Earth

β_{12} = Same as in 1.

$$\beta_{23} = \frac{1}{I_{sp_{23}}\, g\, (1 - V_{23_o}\, \sigma_{23_o})}$$

β_{45} = Same as in 1.

β_{67} = Same as in 1.

3. Aerodynamic Braking at Mars, Aero/Prop Braking at Earth

β_{12} = Same as in 1.

β_{23} = Same as in 2.

$$\beta_{45} = \frac{1}{I_{sp_{45}}\, g\, (1 - V_{45_o}\, \sigma_{45_o})} \quad \frac{1}{1 + \frac{1 - V_{45_o}\, \sigma_{45_o}}{V_{45_o}(1 - \sigma_{45_o})} \quad \frac{W_{LDG}/W_7}{\left[1 + \frac{C_4}{W_7}\,(V_{67_A} - C_5)^m\right]\left[1 + \frac{V_{67_o}(1 - \sigma_{67_o})}{1 - V_{67_o}\sigma_{67_o}}\right] + \frac{W_{MM}}{W_7}}}$$

$$\beta_{67} = \frac{\frac{r_{67_o}\,(1 - \sigma_{67_o})}{g\, I_{sp_{67}}\,(1 - r_{67_o}\, \sigma_{67_o})}}{1 + \frac{V_{67_o}(1 - \sigma_{67_o})}{1 - V_{67_o}\, \sigma_{67_o}} + \frac{W_{MM}/W_7}{1 + \frac{C_4}{W_7}\,(V_{67_A} - C_5)^m} + \frac{1 - r_{45_o}\, \sigma_{45_o}}{r_{45_o}(1 - \sigma_{45_o})} \quad \frac{W_{LDG}/W_7}{1 + \frac{C_4}{W_7}\,(V_{67_A} - C_5)^m}}$$

4. Propulsion Braking at Mars, Aero/Prop Braking at Earth

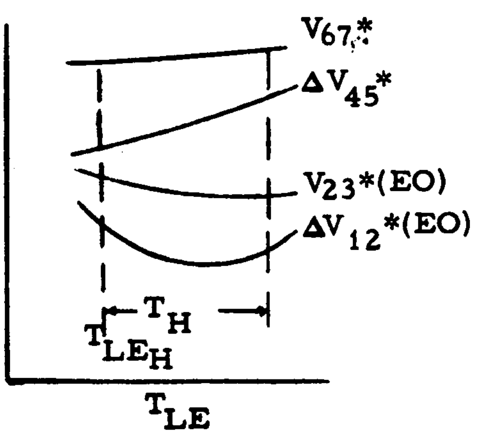
β_{12} = Same as in 1.

β_{23} = Same as in 2.

β_{45} = Same as in 3.

β_{67} = Same as in 3.

The optimum overall trip path (outbound plus inbound) for a given Earth launch date cannot be found by the preceding methods, which employ the Mars arrival date as the primary independent variable. If Earth launch date is used as the primary independent variable, as is required for launch hold analyses, the best combination of inbound and outbound leg paths must be determined for each launch date (the outbound and inbound legs can no longer be treated independently). For example, assume a given Earth launch date: to find the maximum payload for this launch date, a range of outbound leg times must be searched; for each outbound leg time, which determines a Mars arrival date, the best inbound leg time can be computed directly from the suboptimization routine described previously. The overall payload ratio is curve-fitted and the maximum is found by locating the zero slope joint. The corresponding optimum characteristic velocities are now curve fitted over a range of Earth departure dates and the analysis of launch holds initiated. The equations for the payload ratios are given in Table 3.



It is possible to construct curves of mission characteristic velocities versus Earth departure date, as shown in the accompanying figure and curve-fitted as noted previously. The starred values indicate that leg time varies in an optimum manner along the curve. The subscript EO indicates that the velocities vary in an optimum manner for the case wherein Earth departure date is used as the primary independent variable. A nominal Earth departure date and a hold time, T_H, are indicated. The problem is to determine the optimum nominal Earth launch date.

For each nominal Earth departure date the vehicle must be cabable of performing the mission on any day during the hold period, following the best paths as described above. The propulsion stages, for example, will be designed by some worst case during the hold period, and will be off-loaded as required during the more

Table 3

SUMMARY OF EQUATIONS FOR ITERATIVE OPTIMIZATION

(Zero Launch Hold)

1. Aerodynamic Braking at Mars and Earth

$$\frac{W_1}{W_7} = \left\{ \left[1 + \frac{C_4}{W_7} (V_{67}^* - C_5)^m + \frac{W_{MM}}{W_7} \right] \left[\frac{r_{45}(1-\sigma_{45_o})}{1-r_{45}\sigma_{45_o}} \right] + \frac{W_{LDG}}{W_7} \right\} \left[C_2 \left(\frac{V_{23}^*}{1000} - C_3 \right)^m \right] \left[\frac{r_{12}(1-\sigma_{12_o})}{1-r_{12}\sigma_{12_o}} \right]$$

2. Propulsion Braking at Mars, Aerodynamic Braking at Earth

$$\frac{W_1}{W_7} = \left\{ \left[1 + \frac{C_4}{W_7} (V_{67}^* - C_5)^m + \frac{W_{MM}}{W_7} \right] \left[\frac{r_{45}(1-\sigma_{45_o})}{1-r_{45}\sigma_{45_o}} \right] + \frac{W_{LDG}}{W_7} \right\} \left[\frac{r_{23}(1-\sigma_{23_o})}{1-r_{23}\sigma_{23_o}} \right] \left[\frac{r_{12}(1-\sigma_{12_o})}{1-r_{12}\sigma_{12_o}} \right]$$

3. Aerodynamic Braking at Mars, Aero/Prop Braking at Earth

$$\frac{W_1}{W_7} = \left\{ \left\{ \left[1 + \frac{C_4}{W_7} (V_{67_A} - C_5)^m \right] \left[\frac{r_{67}(1-\sigma_{67_o})}{1-r_{67}\sigma_{67_o}} \right] + \frac{W_{MM}}{W_7} \right\} \left[\frac{r_{45}(1-\sigma_{45_o})}{1-r_{45}\sigma_{45_o}} \right] + \frac{W_{LDG}}{W_7} \right\} \left[C_2 \left(\frac{V_{23}^*}{1000} - C_3 \right)^m \right] \left[\frac{r_{12}(1-\sigma_{12_o})}{1-r_{12}\sigma_{12_o}} \right]$$

4. Propulsion Braking at Mars, Aero/Prop Braking at Earth

$$\frac{W_1}{W_7} = \left\{ \left\{ \left[1 + \frac{C_4}{W_7} (V_{67_A} - C_5)^m \right] \left[\frac{r_{67}(1-\sigma_{67_o})}{1-r_{67}\sigma_{67_o}} \right] + \frac{W_{MM}}{W_7} \right\} \left[\frac{r_{45}(1-\sigma_{45_o})}{1-r_{45}\sigma_{45_o}} \right] + \frac{W_{LDG}}{W_7} \right\} \left[\frac{r_{23}(1-\sigma_{23_o})}{1-r_{23}\sigma_{23_o}} \right] \left[\frac{r_{12}(1-\sigma_{12_o})}{1-r_{12}\sigma_{12_o}} \right]$$

favorable portions of the hold period. To illustrate, consider an aerodynamic Earth entry and propulsive braking mode at Mars. The Earth entry system (with fixed payload) will generally experience a maximum Earth entry speed over the range of Earth departure dates within the launch hold period. A search of V_{67*} over the range of Earth departure dates will reveal a maximum V_{67*}, which can be termed V_{67*_m} The Earth entry system weight for this condition is

$$\frac{W_6}{W_7} = 1 + \frac{C_4}{W_7} \left(V_{67_m^*} - C_5 \right)^m \tag{16}$$

$$T_{LE_N} \leq T_{LE} \leq (T_{LE_N} + T_H)$$

$$\frac{W_5}{W_7} = \frac{W_6}{W_7} + \frac{W_{MM}}{W_7} = 1 + \frac{W_{MM}}{W_7} + \frac{C_4}{W_7} \left(V_{67_m^*} - C_5 \right)^m \tag{17}$$

The Mars departure stage must be capable of accelerating the payload, Earth entry system and mission module (the sum of these weights, W_5, remains constant for a given nominal Earth departure weight) to a maximum velocity $\Delta V_{45_{m*}}$. This condition sizes the Mars departure stage jettison weight. The appropriate expression for the stage jettison weight is:

$$\frac{V_{45}}{W_7} = \left[\frac{\sigma_{45_o}(V_{45_m} - 1)}{1 - r_{45_m} \sigma_{45_o}} \right] \left[1 + \frac{W_{MM}}{W_7} + \frac{C_4}{W_7} \left(V_{67_m^*} \; C_5 \right)^m \right] \tag{18}$$

where r_{45_m} is the stage mass ratio corresponding to $V_{45_m}{}^*$.

A search is now conducted to find the maximum Mars aerodynamic entry system weight (which remains constant for a given nominal Earth departure date). The Mars entry system weight is

$$\left\{\left(\frac{1-\sigma_{45_o}}{1-r_{45_M}\sigma_{45_o}}\right)\left[1+\frac{W_{MM}}{W_7}+\frac{C_4}{W_7}\left(V_{67_m^*}-C_5\right)^m\right] r_{45}+\frac{W_{LDG}}{W_7}\right\}\left[C_2\left(\frac{V_{23}{}^*}{1000}-C_3\right)^m-1\right] \tag{19}$$

Equation 19 must be searched to find

$$\frac{W_2-W_3}{W_7}=\left(\frac{W_2-W_3}{W_7}\right)_m \tag{20}$$

(r_{45} and $V_{23}{}^*$ vary with Earth departure date). The entry system weight remains constant for a given nominal Earth departure date.

Similarly, the gross spacecraft weight in Earth orbit is now searched to find its maximum, using the fixed vehicle structural and entry system weights as determined by equations 16 through 19, keeping in mind that the Mars departure stage will be off-loaded except for one launch date.

$$\frac{W_1}{W_2}=\left\{\left(\frac{W_2-W_3}{W_7}\right)_m+\frac{W_{LDG}}{W_7}+\left(\frac{1-\sigma_{45_o}}{1-r_{45_m}\sigma_{45_o}}\right)\left[1+\frac{W_{MM}}{W_7}+\frac{C_4}{W_7}\left(V_{67_m^*}-C_5\right)^m\right] r_{45}\right\}\left[\frac{r_{12}(1-\sigma_{12_o})}{1-r_{12}\sigma_{12_o}}\right] \tag{21}$$

Equation 21 is searched to find

$$\frac{W_1}{W_7} = \left(\frac{W_1}{W_7}\right)_m \tag{22}$$

(r_{45} and r_{12} vary with Earth departure date). The process is repeated for a range of nominal Earth departure dates, and can be searched for a minimum. Similar expressions can be derived for other mission modes.

Special Case With Zero or Fixed Launch Hold

Considerable simplifications can be made in the above equations for the special cases with zero or fixed launch holds (the latter case arises because the parking orbit precession of a few days each separated by periods of a month or longer, when the parking orbit has precessed through 360 degrees). The resulting equations provide an alternate approach to the one given by equation 15. A summary of the appropriate equations is given in Table 3. It is noted that no interior search routines are required in the case of zero hold (a fixed hold requires the comparison of end point values). W_1/W_7 is repeatedly calculated for a range of nominal Earth departure dates (using the optimum paths given by the "starred" velocities). The resulting curve of gross weight ratios versus nominal Earth departure date is curve-fitted to find a minimum.

Special Case With No Propellant Off-Loading

The method of equations 16 - 22 requires, in the general case, the off-loading of propellants to meet the mission propulsion requirements for any given day within the hold period; on certain days the Earth departure stage would be fully loaded, and the Mars departure stage off-loaded, and conversely for other days during the hold period. The implication on the Earth orbital launch operations are unfavorable, because propellants must be stored in orbit and transferred into or out of the spacecraft on a daily basis as required. Furthermore, propellants must be placed into orbit to meet the maximum requirements of all propulsion stages simultaneously, which may be a greater propellant load than that required for any given day. Hence, the gross weights given by the performance equations are understated to a slight degree.

An alternate mode of operation for the launch hold situation, and a significantly easier one to perform in actual practice, is one in which the vehicle is assembled in orbit fully loaded, and remains fully loaded throughout the hold period. The vehicle is capable of performing the mission under all hold situations, and in general will be heavier than a vehicle designed for the off-load mode. However, since the hold penalty is not large, the no off-load spacecraft is penalized but slightly.

The equations for the no off-load mode are the same as those presented in Table 3. The velocity terms entering into these equations (V_{67*}, r_{45}, V_{23*}, and r_{12}) now take the maximum values for these terms over the span of Earth departure days, T_H.

Effect of Thrust-to-Weight Ratio

The suboptimization procedures developed previously in equations 1 through 11 assume impulsive phases, i.e., gravity losses are zero or remain constant in the values of ΔV. It is possible to account for these losses rather easily in the suboptimization equations (the final optimization equations are unaltered since they operate on inputs from the suboptimization routines; hence, it is necessary to revise the suboptimization equations only). Assuming that the ΔV in equation 11 is the ideal value, i.e., without gravity losses, the actual velocity increment is found as

$$\Delta V_a = C_F \ \Delta V$$

where C_F is a correction term for gravity losses dependent upon ΔV, I_{sp}, thrust-to-weight ratio, and departure altitude. For purposes of the present optimization, C_F can be assumed invariant during any given optimization calculation (since I_{sp}, F/W and departure altitude are fixed, and ΔV varies but slightly from a selected nominal). Hence

$$T^* = \frac{C_F A + \alpha C}{C_F B + \alpha D}$$

If propulsion deceleration is used at Mars, equation 11 becomes

$$T^* = \frac{C_{F_{12}} A + C_{F_{23}} \alpha C}{C_{F_{12}} B + C_{F_{23}} \alpha D}$$

Equation 11a can be used for inbound leg optimization.

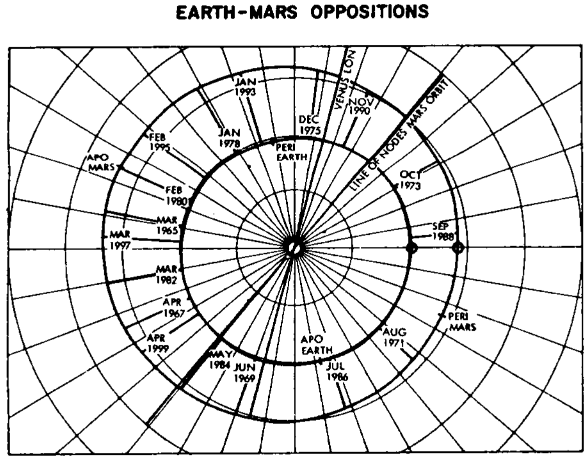


Figure 1. Mars-Earth Oppositions

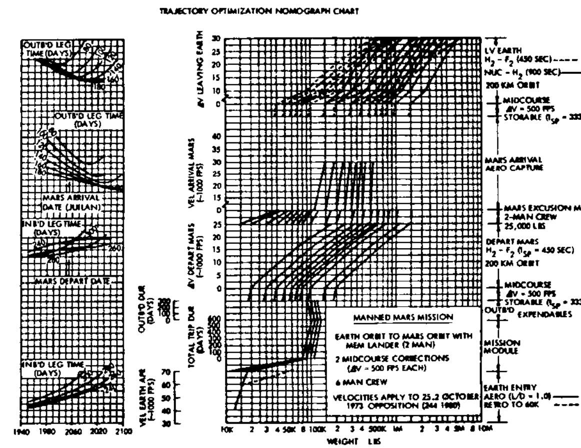


Figure 2. Nomograph for Mission Optimization

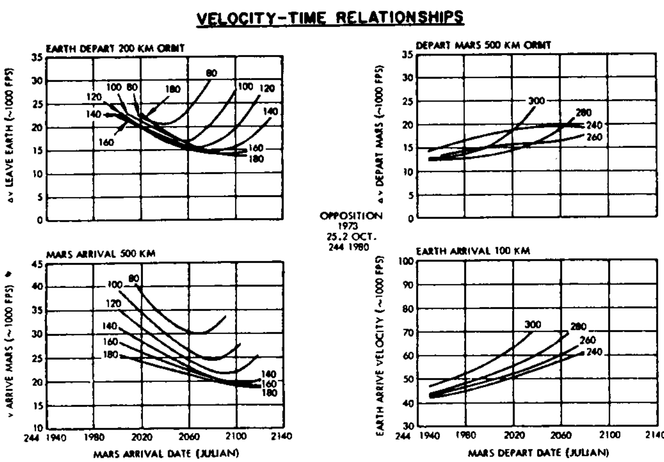


Figure 3. Velocity - Time Relationships

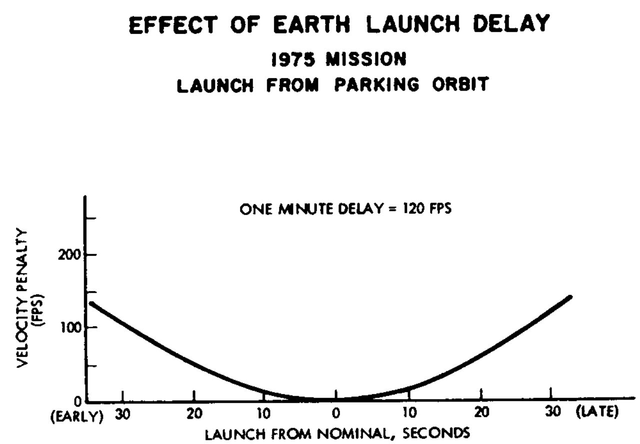


Figure 4. Earth Launch Delay

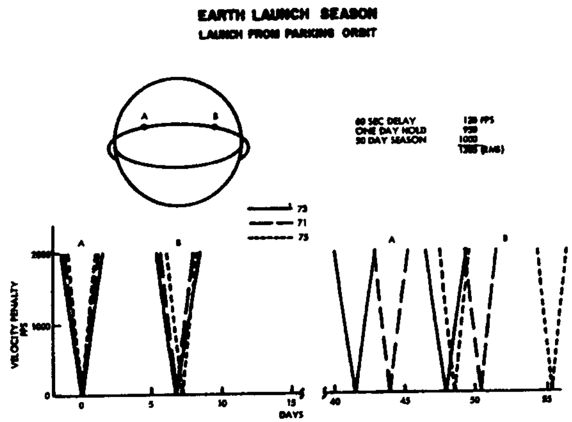


Figure 5. Earth Launch Season

MISSION CHARACTERISTICS
10 DAY STOPOVER
NO EARTH HOLD

OPPOSITION	DATE LEAVE EARTH	DURATION (DAYS) OUTBOUND	INBOUND	TOTAL
10 AUG 1971 (2441174)	1080	140	272	422
25 OCT 1973 (1980)	1880	160	266	436
16 DEC 1975 (2762)	2665	170	250	430
22 JAN 1978 (3531)	3436	179	241	430
25 FEB 1980 (4295)	4192	185	235	430
31 MAR 1982 (5060)	4960	177	232	419
11 MAY 1984 (5831)	5737	168	255	432

Figure 6a. Mission Characteristics - Launch Dates

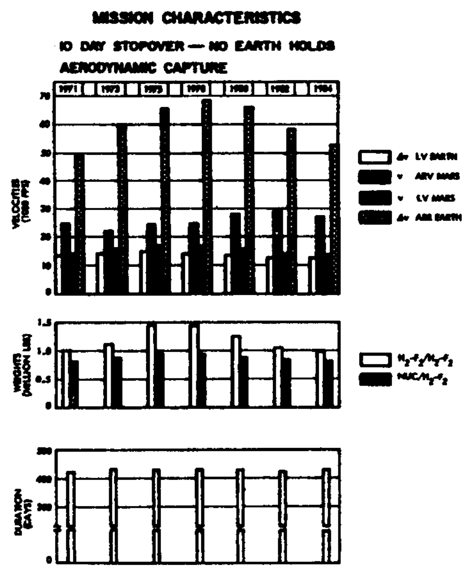


Figure 6b. Mission Characteristics - Velocities and Weights

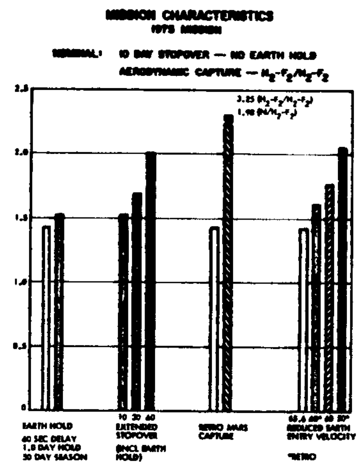


Figure 6c. Mission Characteristics - Performance Requirements

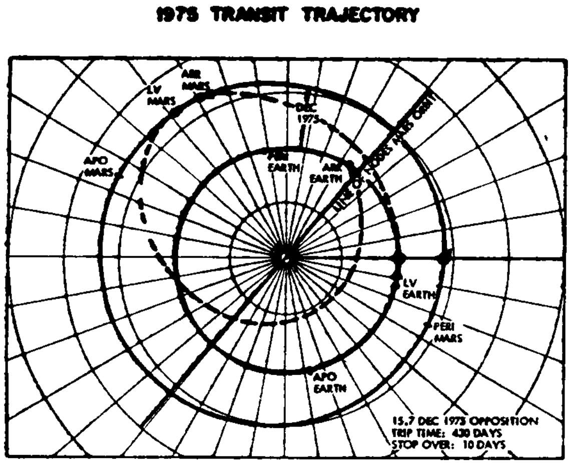


Figure 7. 1975 Trajectory

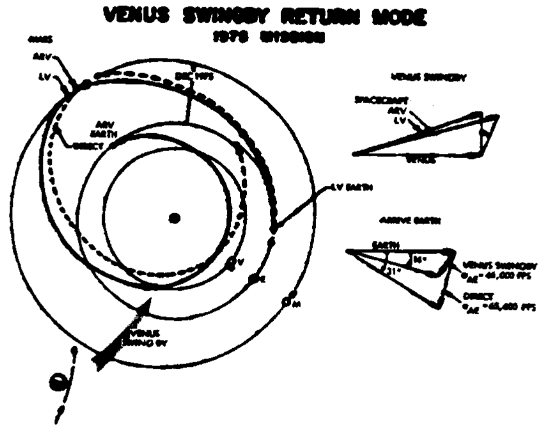


Figure 8. Venus Swingby Return Mission Mode

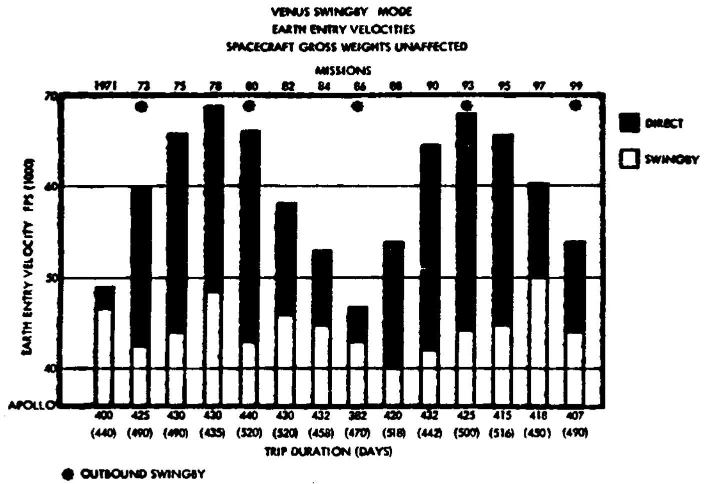


Figure 9. Venus Swingby Return Mode - Effect on Earth Entry Velocities

VENUS SWINGBY RETURN MODE

ADVANTAGES

1. REDUCES HIGH EARTH ENTRY VELOCITIES
2. PERMITS VENUS INSPECTION
3. BREAKS MONOTONY OF RETURN TRIP
4. GENERAL AVAILABILITY WITH STOPOVER MISSIONS

DISADVANTAGES

1. TRIP TIMES INCREASED BY 20% IN SOME YEARS
2. MISSION PROFILE MORE COMPLEX

Figure 10. Advantages - Disadvantages of Venus Swingby Mode

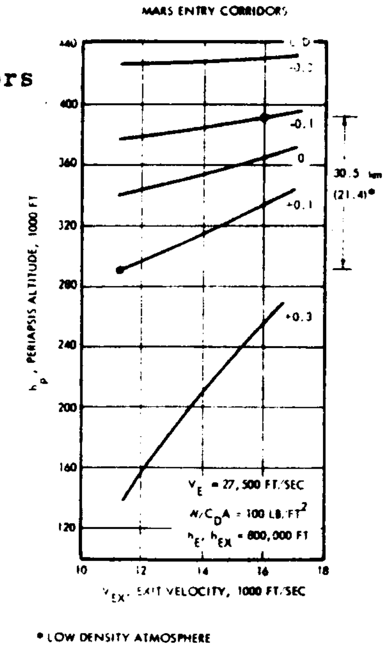


Figure 11. Mars Entry Corridors

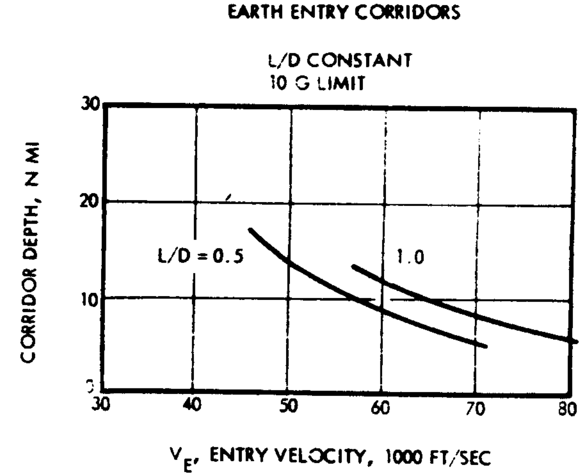


Figure 12. Earth Entry Corridors

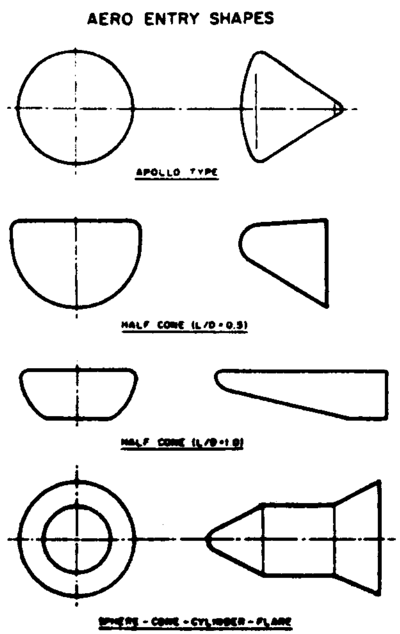


Figure 13. Aero Entry System Shapes

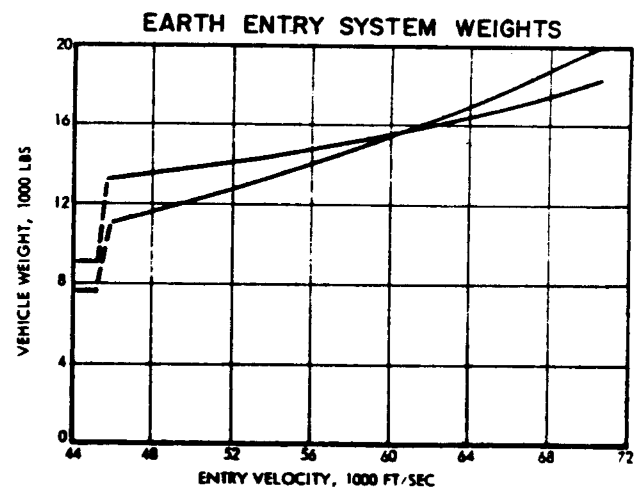


Figure 14. Earth Entry System Weights

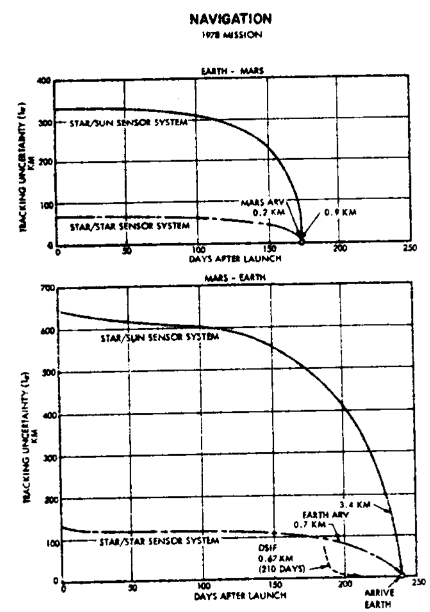


Figure 17. Navigation - Tracking Errors

NAVIGATION
1978 MISSION

	TRACKING UNCERTAINTY (KM)	CORRIDOR (KM)
EARTH-MARS		
STAR/SUN SENSORS	0.9 (1σ)	5.4 (3σ)
STAR/STAR SENSORS	0.2	1.1
MARS-EARTH		
STAR/SUN SENSORS	3.4 (1σ)	20.4 (3σ)
STAR/STAR SENSORS	0.7	4.2
DSIF (AT 210 DAYS)	0.67	4.0

CORRECTION VELOCITIES

EARTH - MARS	345 FPS (3σ)
MARS - EARTH	300
	645

BASED ON: DSIF NEAR EARTH
STAR/SUN SENSOR SYSTEM NEAR MARS.
(CORRECTION REDUCES TO 600 FPS WITH STAR/STAR SENSOR SYSTEM.)

Figure 18. Navigation - Corridors

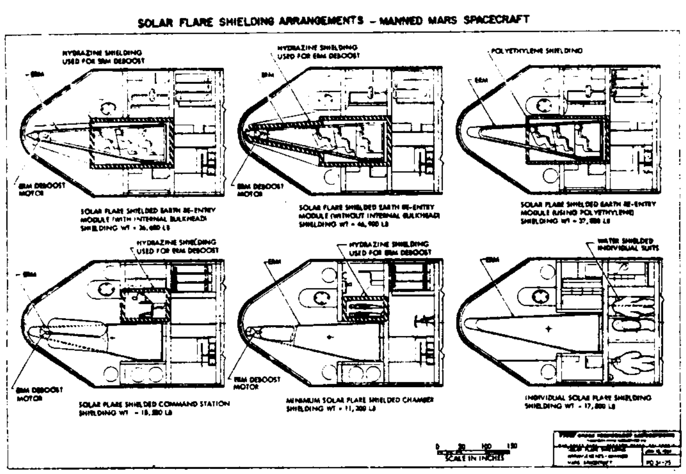


Figure 20. Solar Radiation Shielding Comparison

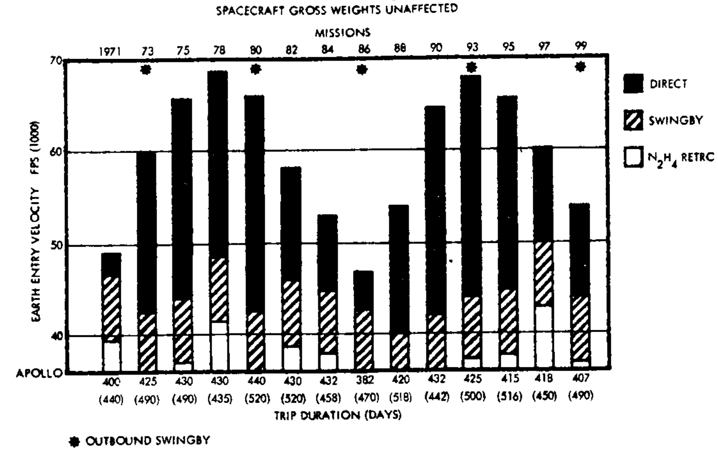


Figure 21. Venus Swingby Return Modes - Use of N_2H_4 Earth Retro

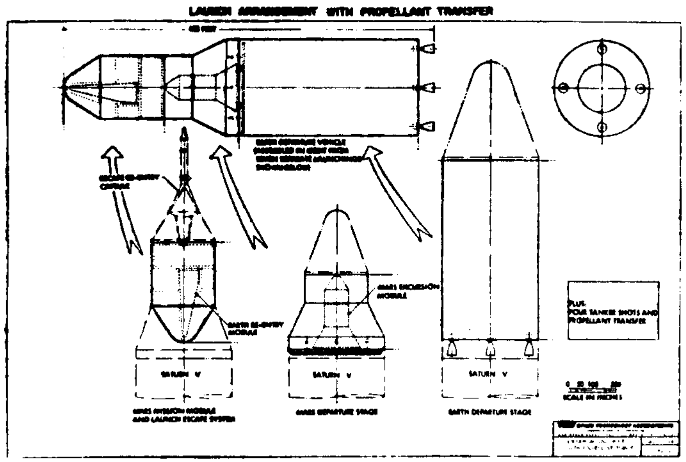


Figure 22a. Earth Launch and Assembly - Tanker Mode

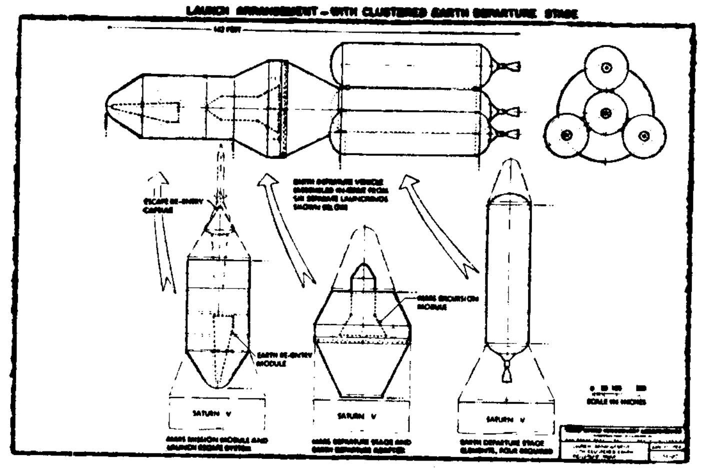


Figure 22b. Earth Launch and Assembly - Module Mode

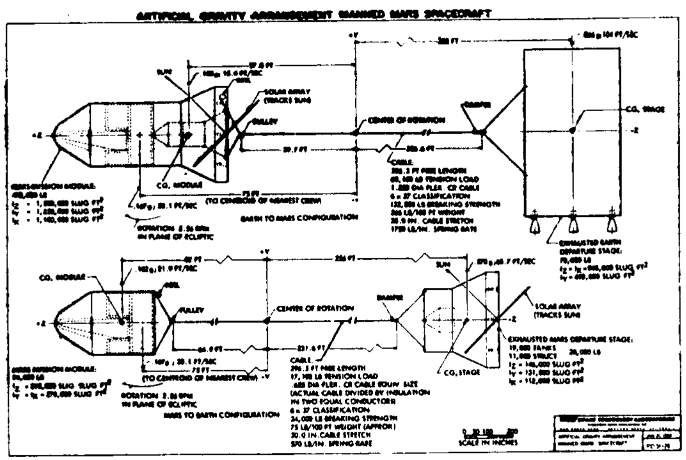


Figure 23. Artificial Gravity Provisions

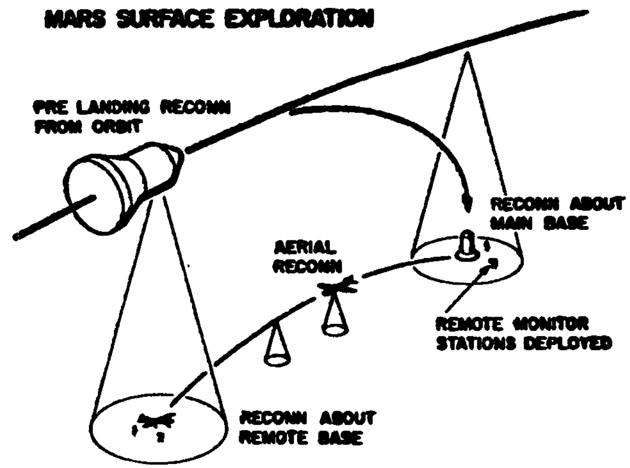


Figure 25. Surface Exploration

26984

PART 6

MANNED MARS LANDING AND RETURN MISSION STUDY

by

A. L. Jones
North American Aviation
Contract No. NAS2-1408

MANNED MARS LANDING AND RETURN MISSION STUDY

by

A. L. Jones, North American Aviation
Contract No. NAS 2-1408

This study of manned Mars landing and return missions is being conducted by Advanced Systems of S&ID for the Ames Research Center. The Study is approximately 80 percent complete at this time, however, many characteristics of the mission and the spacecraft modules can be described at this time. The final conclusions must await completion of the study.

The interaction between the mission requirements and spacecraft design are complex and require detailed analysis to evaluate concepts for complete systems and to establish the characteristics of the most promising modes. Before a commitment is made for a manned landing on the planet Mars, analytic proof must be given as to the feasibility of such a mission. In order to acquire the necessary quantitative data one must initially evaluate systems concepts; concepts involving not only the mission modes, but hardware concepts as well. Within the constraints of the evaluation some concepts may appear more promising than others thereby leading to a more detailed analysis in order to expose technical problem areas which might significantly enhance overall mission feasibility. These, therefore, are the broad objectives of the Manned Mars Landing and Return Mission Study.

The guidelines which both direct and constrain our analyses are listed in Figure 1 and the logic of how these guidelines act as constraints to the evaluation are shown in Figure 2. Basically, the evaluation can be divided into the areas of mission characteristics and spacecraft modules. The time/energy characteristics serve to identify the various mission modes and directly influence the design of the spacecraft modules, both manned and propulsive. It is obvious from the diagram that the interrelationships of the mission characteristics with the spacecraft prohibit independent suboptimizations. For example, the duration of flight from Earth to Mars not only specifies an explicit ΔV

requirement (for a given launch date) but is also a controlling factor in the determination of booster efficiencies because of propellant storage time.

In any investigation of this type, it is natural to start with the "end" payload and work backwards, since both the Earth entry module and the crew quarters (mission module) are returned to the vicinity of the Earth. The discussion today will be directed to several of the significant factors influencing the design of the manned modules and propulsion modules. The effect of a mission time and energy requirements on these modules will be presented to show the effect of mission duration, stay time, crew size and propellant combination of the size of the complete spacecraft system. Throughout this study, all of these modules were investigated and defined parametrically as a function of the study constraints. Such parametric data are required to accurately evaluate system concepts. The time available here today will not permit detailed discussion of the extensive parametric data produced in this study. However, I would like to highlight a few of the areas which may be of interest.

The manned modules must provide environmental protection, life support and system command and control. These functions are required for the crew no matter what mission mode or concept is used. In this study a separate manned module was used for Earth entry and landing in all mission concepts. Modules for interplanetary flight and Mars landing were analyzed and parametric data developed for various crew sizes and mission times.

One of the initial investigations required in the study was to evaluate the radiation protection requirements to establish manned module design concepts. Shown in the upper part of Figure 3 are the recommended radiation exposure limits to different areas of the body. These are somewhat arbitrary radiation exposure limits over a long duration since the term maximum permissible dose frequently receives the unfortunate interpretation that there is some magic level of radiation exposure which is harmful or harmless. One could as well have a "maximum permissible exposure to gunfire" or an "allowable limit of tissue damage due to gunshot wounds." Clinical radiologists as well as radiation researchers sometimes receive higher levels in the normal course of their occupation. In an emergency situation, such as might be experienced by a physician during a decontamination procedure, a single exposure of 25 rem up to 300 rem in a month is frequently listed

"permissible exposure." Diagnostic and therapeutic radiation exposure levels are governed by the relation of seriousness of illness to danger or radiation. During fluroscopic examination of the abdomen patients may receive 50 to 100 rem to the abdomen in a single exposure while therapy for some cancers produces as much as 100 rem in a single whole body exposure, which may be repeated daily.

Based upon these data, recommendations were made for emergency limits that could be allowed in order to establish design requirements. The permissible average levels presented in the lower portion of Figure 3 are for exposures where the dose is received in small increments that can be incurred with no serious permanent effects.

The two major sources of radiation are from galactic cosmic particles and from solar flare protons. Inside the spacecraft the radiation is comprised mainly of primary particles and secondary protons and neutrons. The galactic secondary dose rate is plotted, in Figure 4, for a typical mission module "configuration." The point to be made here is that the crew will be exposed to a low dose rate from this background radiation which is fairly level over a range of end thickness and which imposes no design problems for these interplanetary spacecraft. The manned mission module design concepts that we now have are considerably less than 10 grams per square centimeter outside the storm shelter, which further reduces this background dose. The solar proton dose rate can establish radiation shielding requirements for the storm shelter. Figure 5 is a plot of the dose rate encountered for the Bailey model proton event on a spherical aluminum shell of varying thickness. This is a large flare (larger than any observed to date) of approximately 8×10^9 protons of greater than 20 MEV. If one were to design to limit the dose rate to 1 to 2 rads per hour, then the "storm cellar" should have an equivalent shield thickness of about 10 to 15 grams per square centimeter. These values are very reasonable for design of Mars spacecraft.

A preliminary study of the expected Mars mission doses was completed in which the effective residual dose was determined. This approach considers the human body's ability to repair itself from certain injuries. Such repair is inaccurately known but in these preliminary analyses approximately 90 percent of the effects sustained were assumed to not be residual. With this assumption the effective residual dose (ERD) for acute whole body doses of $\angle 100$ rem can be expressed as:

$$ERD = 0.1D + 0.9D^{-0.231t}$$

where D is the acute dose received and t is the time in days since the acute dose. This equation formula states the assumption that 10 percent of the biological effect is irreparable, and the other 90 percent is repaired, half of it within 30 days. Figure 6 illustrates the results obtained from a 480-day mission (220 days outbound, a 30-day stay on Mars, and 230 days return). A random group of seven flares ranging from 70 rem to 5 rem shielded doses were assumed and the doses due to galactic cosmic rays and the trapped radiation belts were accounted for. The peak ERD is 110 rem or less at all times during the trip. Radiation shielding protection requirements for this example mission were approximately 15 grams per square centimeter. The ERD for this example was less than 50 rem upon return to Earth. Since the support of life and control of cabin environment is necessary for the success of manned Mars missions, the study of the ecological control system (life support and environmental control) is essential in overall mission and vehicle study programs. Ecological systems with various degrees-of-closure were analyzed to determine their weight, volume, and power requirements. The effects of cabin leakage, power penalty weight, and additional heat load on the ecological system were investigated. These investigations included considerations of 3- to 10-man crew size, of 10- to 600-day missions, of 0 to 20 lb/day cabin atmosphere leakage rate, and of seven ecological systems representative of various steps in degree of closure. A closed system is defined here as a system which supplies all of the life support requirements without makeup. The total life support requirement used in this study was 17.36 pounds per man-day which included eight pounds of wash water per man-day.

The weight, volume, and power requirements for ecological systems depend on the subsystem processes selected, mission duration, crew size, and supply requirements. In order to estimate the supply requirements and performance requirements for the ecological subsystems, a balance of the atmospheric gases, water, food, and makeup supply is necessary. All of the seven final systems analyzed required 100 percent makeup supply of food. Systems for reconverting waste products into food were not considered to be at a stage of development which would permit a sufficiently accurate analysis for purposes of comparison.

The water balance for these systems is shown in Figure 7. The basepoint system (least amount of closure) reclaimed wash water, and water vapor from perspiration and respiration. This basic system (A_A) supplied 14.2 pounds of water per man-day for personal hygiene, food, and drink. Since this system uses CO_2 removal by adsorption, a

makeup from storage of 2.33 pounds of water per man-day is required. This would amount to about 7000 pounds for 6 men on a 500 day mission. When a CO_2 conversion loop is added in the atmospheric system to maintain an O_2 balance, the water makeup requirements increase. The methanation process can be added for partial water recovery. This illustrates the importance of integrating the various ecological subsystems.

Systems such as C and D which include subsystems for urine and then fecal water reclamation show a significant decrease in water makeup requirements. Systems E and F achieve a water balance by not supplying water for O_2 makeup by electrolysis.

The oxygen makeup requirements for these same seven systems are shown in Figure 8. The basic system supplied the metabolic requirements of 1.8 pounds per man per day from stored supplies. Addition of the electrodialysis method of CO_2 removal results in a small reduction in the O_2 makeup requirements. However, by addition as subsystem for CO_2 reduction by methanation and electrolysis of water, the O_2 makeup can be reduced essentially to zero, as shown by systems B, C, and D. These systems present an oxygen balance but require water makeup. System F which uses a subsystem for direct conversion of CO_2 to carbon and oxygen does not require water makeup. This subsystem, however, is expected to have a long development time.

After detailed investigations of the ecological systems, each was analytically expressed in terms of crew size, mission duration, leakage rates and power penalties. A cross-plot of these data for the particular mission parameters shown appears in Figure 9 as system weight versus percent (degree) closure. This is shown to illustrate that weight alone may not be the deciding factor in choosing an ecological system. Systems D and F, for example, have subsystems or components with predicted development times which preclude their selection as a representative system for only the last part of the time periods of this study. Although this development problem was known earlier, it is necessary to conduct thorough analyses on these systems in order to determine what advances in technology are required and what advantages these advances offer. Similar studies were conducted for the electrical power requirements, the communications system and for guidance and control of the spacecraft in terms of weight, power, and volume requirements to define the structural requirements and design criteria of a representative mission module.

The Earth entry module is characterized by a totally different set of design constraints, the most significant of which are shown in Figure 10. This plot of corridor depth versus entry vehicle L/D with entry velocity as a parameter shows little is gained using vehicles with L/D of > 1.0. The shaded area is representative of the limitations imposed on a return vehicle by the inaccuracies of not only the guidance systems but also by variations in the Earth's upper atmosphere. Irrespective of the problems encountered in aerothermal analyses and heat protection at the higher velocities, one may deduce that guidance limitations may establish Earth entry velocities which may be a mission condition which specifies the entry module design parameters. A small change in guidance capability can influence greatly the optimum combinations of mission profile and spacecraft design. A later chart will illustrate the effect on mission profile and energy requirements of restricting the Earth entry velocity to a specific value.

Preliminary designs were conducted for Apollo, M-1, M-2, and lenticular shaped Earth entry modules, each for crew sizes of 3, 6, and 10 men. The design analysis of heat shield and structure for each of these configurations allows us to express this payload parameter quantitatively for mission analysis, as shown in Figure 11 (6-man crew curves omitted for clarity). The designs reflected minimum vehicle envelopes to enclose the crew and their associated equipment which provides life support, command, and control capability for two days space flight before entry. Reference data on aerodynamic heating was developed for each configuration in which the radiative heating contribution was determined by approximate methods. The heating was separated into three categories: convection, equilibrium radiation and non-equilibrium radiation. The separation of these heating mechanisms at high speeds leads to conservative results as compared to solution when iterations are included.

The packaging efficiency of the various spacecraft shapes results in a fairly large difference in weight between shapes for small crew sizes. This difference decreases as the crew size increases. The increase in weight for higher velocities is quite significant for the blunter shapes due to the increasing importance of radiation heating on the total vehicle.

In order to determine the characteristics of the most promising Mars mission, the payload modules must be defined in a parametric form such as these data for the Earth entry modules.

In a similar manner, analyses were conducted and design layouts prepared for Mars landing spacecraft. Because of the low velocity of entry into Mars from near-Mars orbits (about 12,000 ft/sec), the possibility existed that ablation heat protection is not required, but rather, that radiation cooling employing a thin external metallic skin and an efficient internal insulation can be employed for heat protection. Because of this possibility, initial emphasis was placed on determination of radiation equilibrium temperatures that might be expected for orbital entry of Mars landing vehicles. Three basic investigations were conducted: (1) a determination of the flight path angle and velocity combinations that result in a given maximum surface radiation equilibrium temperature during initial entry with full positive L/D for a typical Apollo vehicle, (2) a determination of the maximum radiation equilibrium surface temperature that occurs during sub-circular equilibrium glide as a function of the fundamental vehicle parameters, and (3) a determination of the insulation requirements to maintain a given structural temperature for M-2 and Apollo shaped vehicles. From Figure 12, if descent from a circular orbit at 1000 nmi with a range angle of 100 degrees occurs, the maximum radiation equilibrium temperature during entry is about 2200 degrees R for the initial conditions at an entry altitude of a million feet of $\gamma_i = 13.5$ degrees and $u_i = 12,000$ ft/sec. These studies indicated that the use of radiation cooling with an external metallic skin is feasible for Mars orbital entry.

Having briefly discussed some aspects of manned module design parameters, we will direct our attention to the propulsion modules which, as will be shown later, are the most sensitive elements of the spacecraft. It became obvious early in the study that the necessity for developing realistic propulsion module weights would indeed be necessary to provide valid results. Referring back to the Evaluation Logic Model of Figure 2, one can see that the propulsion modules are a function of the energy requirements, duration of propellant storage, the payloads and the choice of propellant combinations. Additionally, all engine parameters, feed systems, tankage and structural configurations, loads as a function of thrust-to-weight, stage L/D's, penalties for thermal insulation and micrometeoroid protection, etc., must be accounted for. In order to accomplish this the S&ID IBM 7094 Stage Mass Fraction Program was employed to compute results such as shown in Figure 13. Equations were developed for structural elements, insulation, and stage subsystems for propulsion module designs for both interplanetary (long storage time) and Earth orbit escape (short storage time) application. The stages for different propellant combinations were computer designed as a function of payload, and ΔV. The

engine systems were computed for the thrust required for the specific stage, however, all loads were computed based on an Earth orbit escape burnout thrust-to-weight ratio of 1.0. Figure 13 shows only one of a large number of data plots required to effectively determine the total spacecraft weight in Earth orbit as a function of mission time. These data are for interplanetary stages and include provisions for the margins as indicated.

Finally, these values are entered into a Stage Optimization Program from which stage efficiency curves (gross-to-payload ratios) are generated such as shown on the left side of Figure 14. Note that the efficiency of OF_2/MMH is highest (lowest W_G/W_{PL} ratio) for the smaller payload weights at lower ΔV. A vertical "slice" taken at a payload of 10^5 pounds is plotted on the right hand side illustrating the sensitivities of the three propellant combinations to storage time in space. Once a payload is established for a given ΔV, the Stage Optimization Program is rerun to yield the number of stages with the design features of each stage. In essence, the mission/spacecraft parameters are analytically controlled to develop quantitatively substantiated design criteria.

Figure 15 illustrates the variation in total minimum energy requirements as a function of the years of Mars opposition for retro-braking missions of varying duration. For the cases shown, the variation is greater for the missions of shortest duration. There is also a cyclical effect for the Earth entry velocities associated with these minimum energy dates. The difference between high and low energy years can be minimized by proper selection of mission duration and stay time. Since any small increase in energy requirements reflects a very large increase in spacecraft weight, it is easy to see why the "difficult years" are so named.

Means of reducing these mission energy requirements were investigated in the study, however, not all of these analyses are complete at this time. Lunar encounter proved to provide only a maximum of 200 feet per second reduction with a considerable increase in mission complexity. Venus encounter is presently under study and some preliminary results are shown in Figure 16. This compares a 1970 launch that swings by Venus on the trans-Mars leg, with a favorable (May 1971) launch for a direct flight to Mars. The ΔV's are subscripted in the usual convention: ΔV_1, Earth-Mars transfer; ΔV_2, Mars retrobraking; and ΔV_3, Mars-Earth transfer. The mission

with encounter is of longer duration and results in favorable Earth entry velocities and Mars departure ΔV's. The spacecraft weights are shown for the retrobraking mission only. Although the module weights data for aerobraking missions are not yet available, one can surmize that a reversal in the weight trends can occur by the summation of ΔV_1 and ΔV_3 and recognizing the lower Earth entry velocities for the Venus encounter mission. With aerobraking at Earth and Mars the total velocity requirement for an August 1970 launch with Venus encounter is 21,500 feet per second. This can be compared to 28,300 ft. per second for the favorable direct mission of May 1971. In taking a more detailed look at the energy requirements as a function of mission duration, the results from a particular opposition period can be illustrated as shown in Figure 17. The plot to the left shows the minimum total energy requirements for aerobraking and retrobraking missions of 10, 30, and 60 days Mars stay times with corresponding Earth entry velocities shown to the right. If a Mars stay time of 30 days is selected, these minimum energies will be encountered for aerobraking missions at a total mission duration of 420 days and for retrobraking missions at 480 days. If a maximum Earth entry velocity of 65,000 ft/sec is a constraint to these missions, the dashed line illustrates how the energy curves are modified (for the 30-day cases, only) . . . the shaded area representing the penalties. It can be seen that the aerobraking mode is much less sensitive to the imposition of this mission constraint.

Figure 17, however, is applicable to missions launched on a specific day. What is the penalty, therefore, if there is a launch delay, e.g., what ΔV contingency must be designed for in the spacecraft? Figure 18 illustrates the variation in ΔV_1, ΔV_2, ΔV_3, and V_e as a function of launch date as well as the summation of launch date as well as the summation of ΔV's. In "opening" the available launch date, one must pay a penalty in each of the energy requirements as well as in V_e. For example, a spacecraft designed for a launch window of 20 days to either side of the minimum (2442670) must be sized for a ΔV_1 and ΔV_2 at 2442650 and for a ΔV_3 and V_e at 2442690. There are two other effects not accounted for in this chart; the so-called "push button" penalty (almost of negligible significances) and the effect of nodal regression of the parking orbital planes at both Earth and Mars. These are not additive quantities, however, but must be accounted for more so from the standpoint of Earth launch operations than of spacecraft design.

Similar conditions surrounding a 1975 aerobraking mission are shown in Figure 19 with ΔV_2 replaced by a dashed line of Mars entry velocity. Note that the mission duration is more optimally chosen for the aerobraking mission (Fig. 17), that the Earth entry velocities are lower than those for the retrobraking mission and that the Mars entry velocities are not prohibitive.

The computation of weights as a function of Earth launch window illustrates a (Fig. 20) point made earlier -- a small increase in ΔV results in a very large increase in spacecraft weight. Although the comparison of aerobraking with retrobraking spacecraft weights is not made for the identical propellant combinations, it is obvious that aerobraking missions would be preferred from a weights criterion. In general, the penalty for large operational launch windows will be severe with the aerobraking mission a little less sensitive to this problem.

The problem of spacecraft/mission evaluation has just begun with the selection of candidate missions and the computation of spacecraft weights. At this stage of investigation of a manned landing on Mars, it behooves one to ask the questions: What are sensitive parameters of the mission? Where can we tolerate approximations and where must we direct detailed investigations? Figure 21 is a tabulation of gross parameter sensitivities for a 1975 retrobraking spacecraft utilizing LF_2/LH_2 propulsion with mission characteristics shown at the bottom of the chart. The column to the right of the chart reflects the percentage change of the spacecraft weight in Earth orbit for a percentage change shown in the parameters listed on the left. A brief review of these numbers clearly illustrates the high degree of sensitivity of all propulsion module parameters, e.g., ΔV, I_{sp}, and the stage mass fraction, $\sqrt{B}$. The manned module weights, however, are relatively insensitive -- a 100 percent change in MEM weight, for example, would alter the total spacecraft weight by only 6 percent.

The effect of propellant choice on spacecraft weight is shown at the bottom of the chart in Figure 21 and to the left of Figure 22. Weight alone, however, may not be the deciding factor, as illustrated by the right hand side of the figure. The comparisons were not drawn either to throw out LO_2/LH_2 nor to promote LF_2/LH_2, but rather to illustrate the operational factor of Earth launch which must be considered in order to develop realistic design criteria and delineate areas needing further investigation. For example, the LO_2/LH_2 case was computed on

the basis of relatively current technology. By considering high chamber pressure LO_2/LH_2 engines, we can achieve a vacuum I_{sp} equal to that of LF_2/LH_2 and with the added benefit of a higher propellant bulk density. This will have the dual effect of significantly reducing both spacecraft weight and size.

A comparison with a representative aerobraking configuration is also shown. It is becoming apparent that aerobraking cases are more desirable from the standpoint of spacecraft weights and spacecraft sizes, Earth entry velocities and general sensitivities to Earth launch windows. More comprehensive analyses are currently underway to better define and to compare these with other mission modes.

FIGURE 1

MANNED MARS LANDING & RETURN MISSION STUDY

OBJECTIVES

- DETERMINE FEASIBILITY
- EVALUATE SYSTEM CONCEPTS
- ESTABLISH PROMISING CONCEPT CHARACTERISTICS
- EXPOSE TECHNICAL PROBLEM AREAS

GUIDELINES

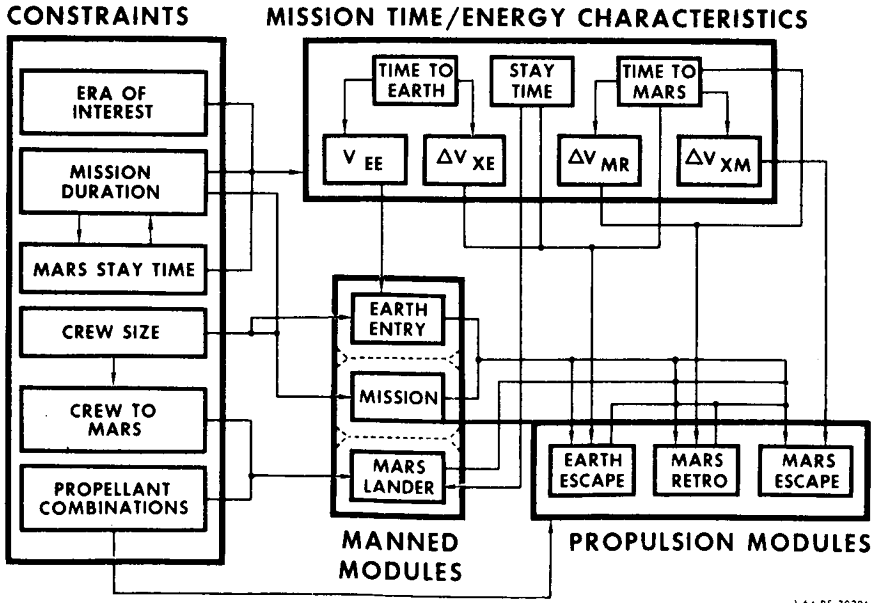
- 12 TO 18 MONTHS TOTAL MISSION TIME
- 7 TO 60 DAYS ON OR AROUND MARS
- 3 TO 10 MAN CREW
- CHEMICAL PROPULSION EARLY 70'S
- NUCLEAR PROPULSION LATE 70'S & 80'S

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

12-63 PS 4A706

FIGURE 2

EVALUATION LOGIC MODEL



NORTH AMERICAN AVIATION INC
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70286

FIGURE 3

RADIATION EXPOSURE LIMITS

AREA	DEPTH G/CM^2	MAXIMUM PERMISSIBLE AVG YEARLY DOSE RAD
SKIN OF WHOLE BODY	0.007	250
EYES	0.3	25
BLOOD FORMING ORGANS	5.0	50

WHOLE BODY ACUTE EXPOSURE

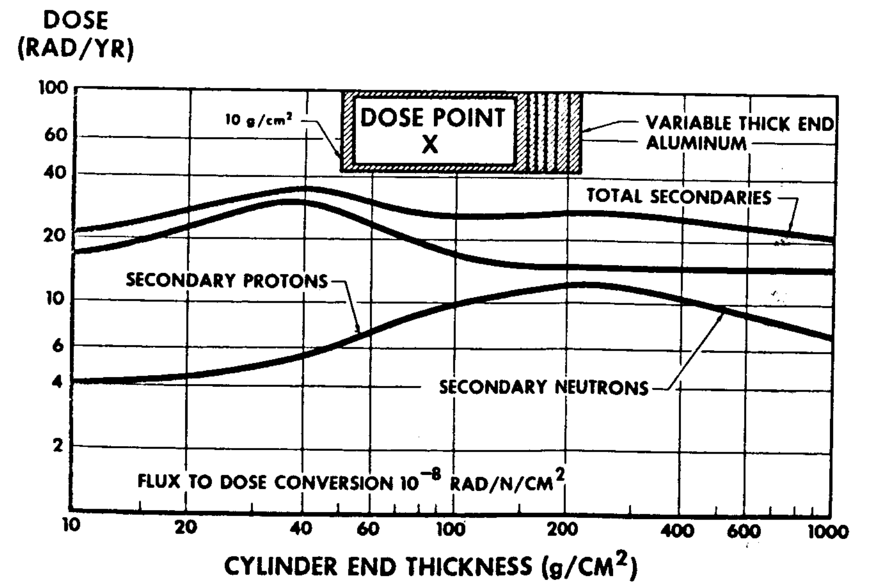
RATE	TOTAL	REMARKS
25 REM	250/YEAR	RECOMMENDED LIMIT
25 REM	300/MO 1000/YEAR	EMERGENCY LIMIT

NORTH AMERICAN AVIATION, INC.

9-63 PS 59724 A

FIGURE 4

GALACTIC SECONDARY PARTICLE DOSE



NORTH AMERICAN AVIATION, INC.

PS 70287A

FIGURE 5

SOLAR PROTON DOSE RATE

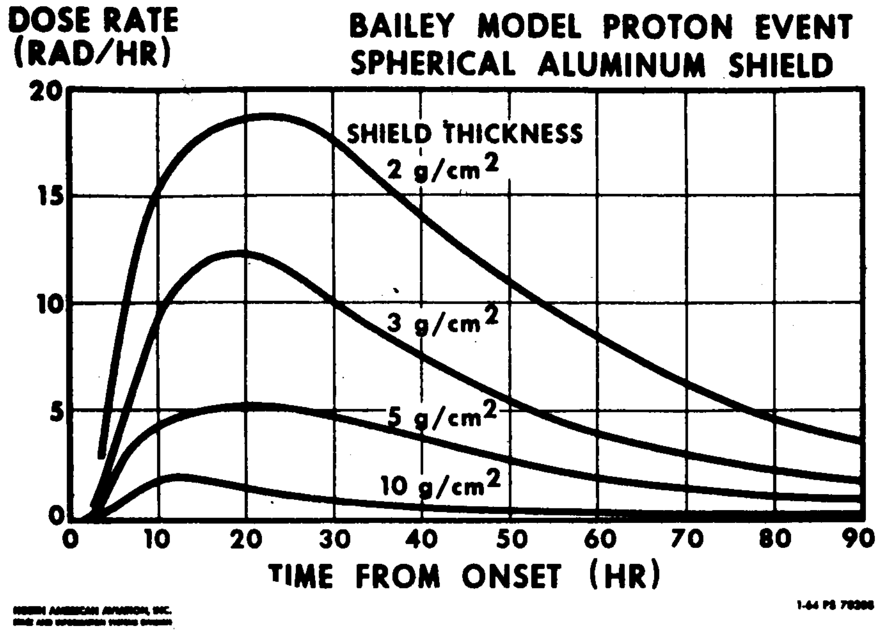


FIGURE 6

MISSION INTEGRATED DOSE

15 g/cm² AL SHIELD

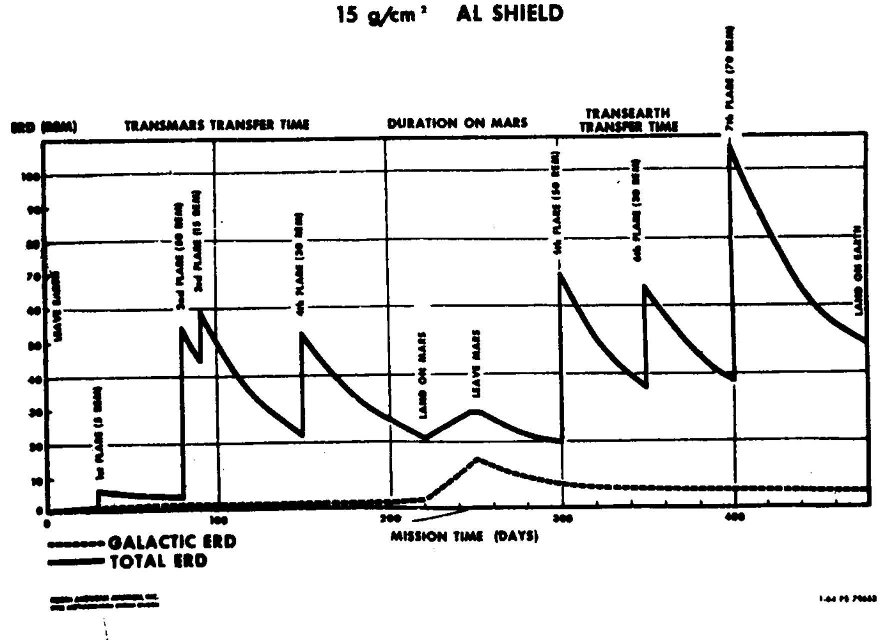


FIGURE 7

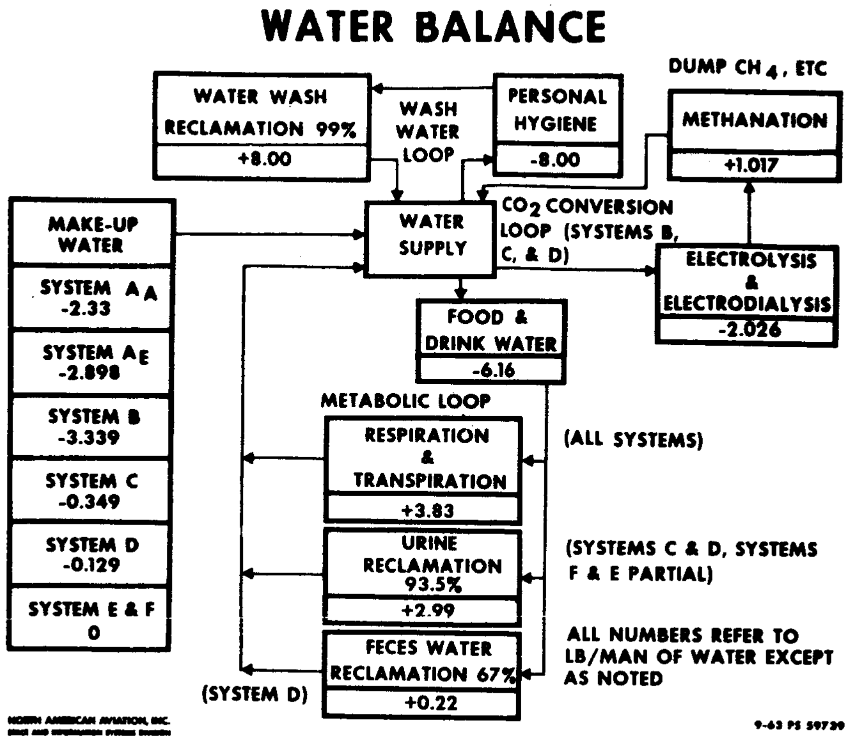


FIGURE 8

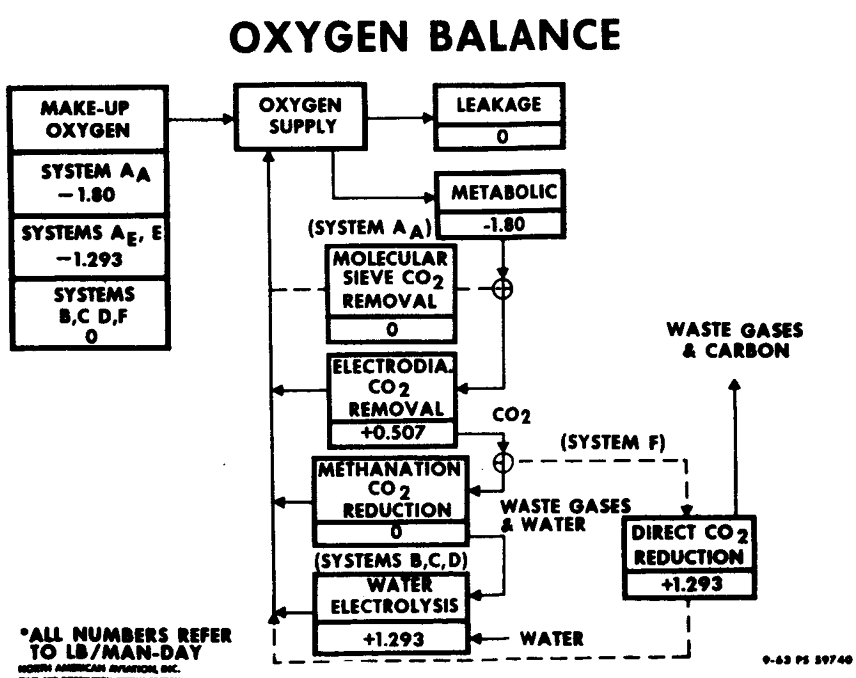


FIGURE 9

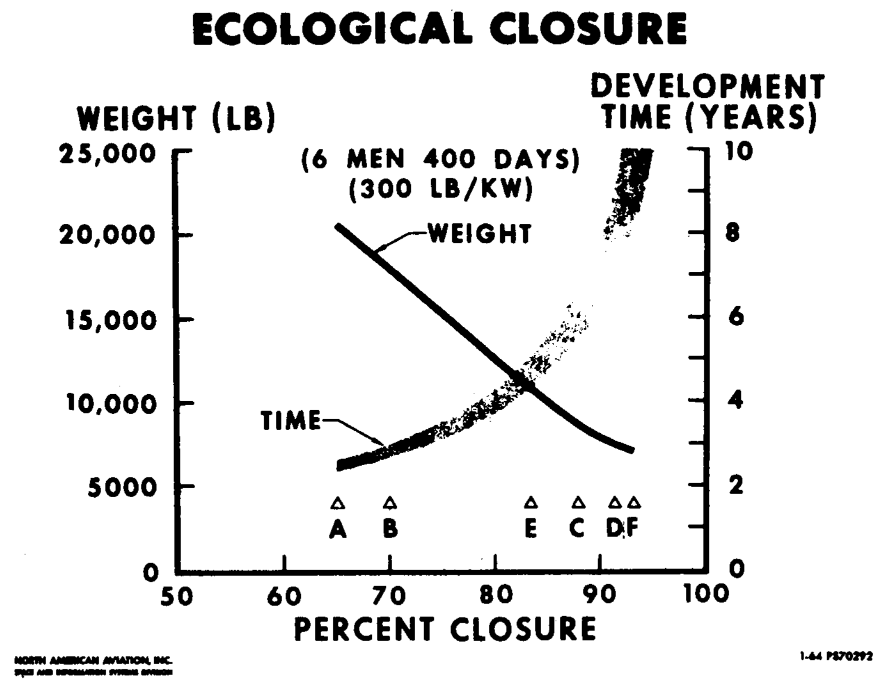


FIGURE 10

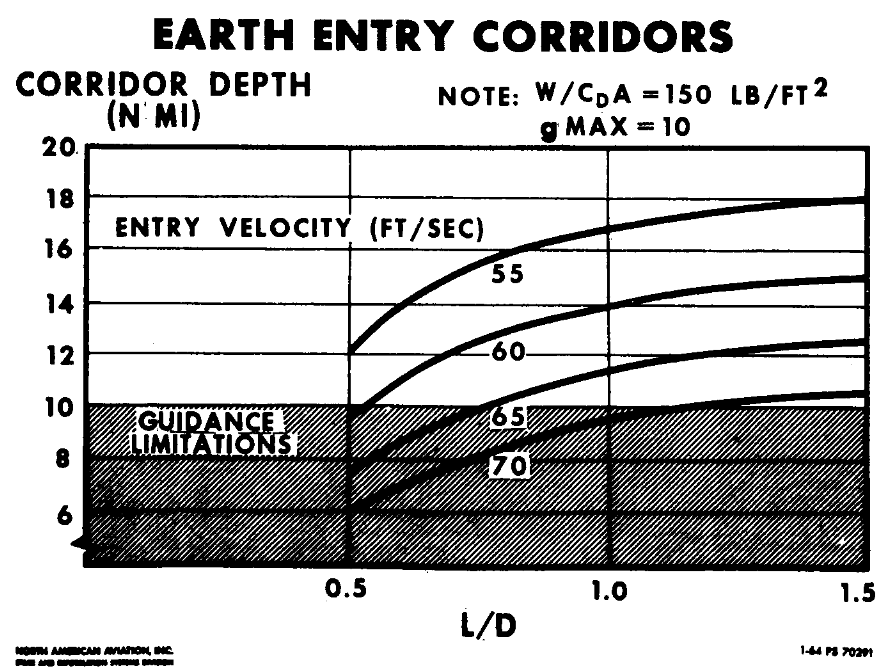


FIGURE 11

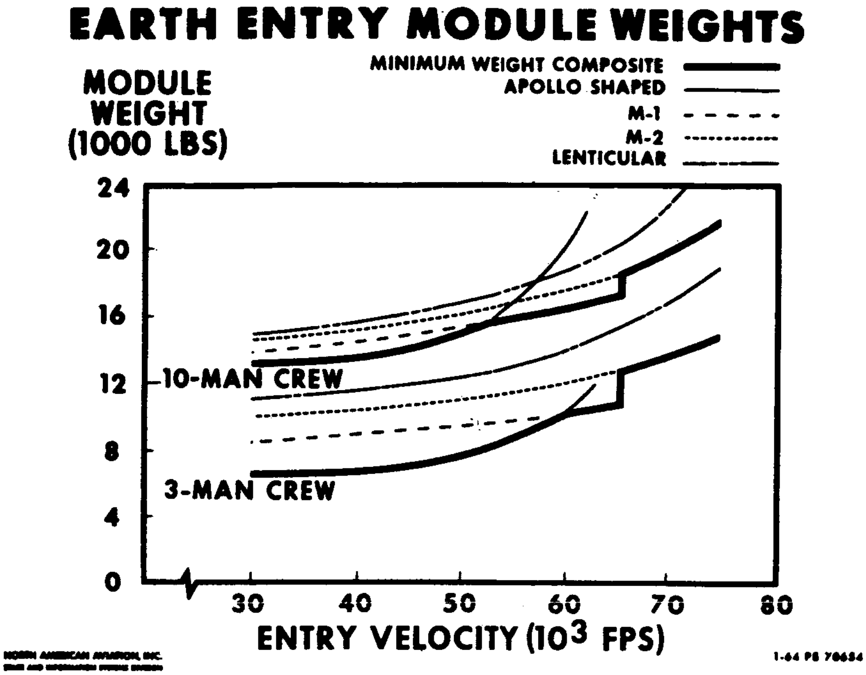


FIGURE 12

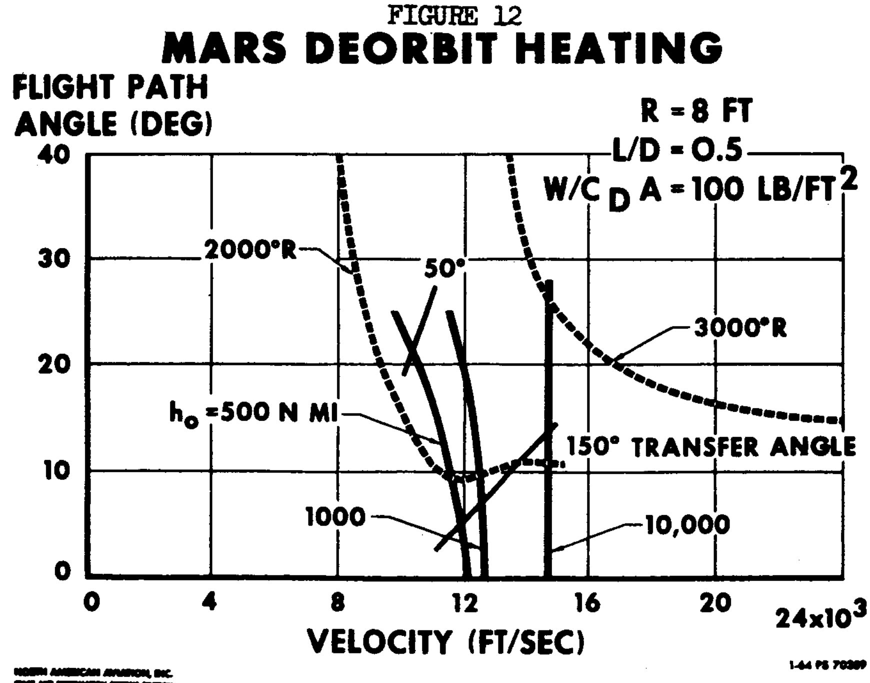


FIGURE 13

STAGE MASS FRACTIONS

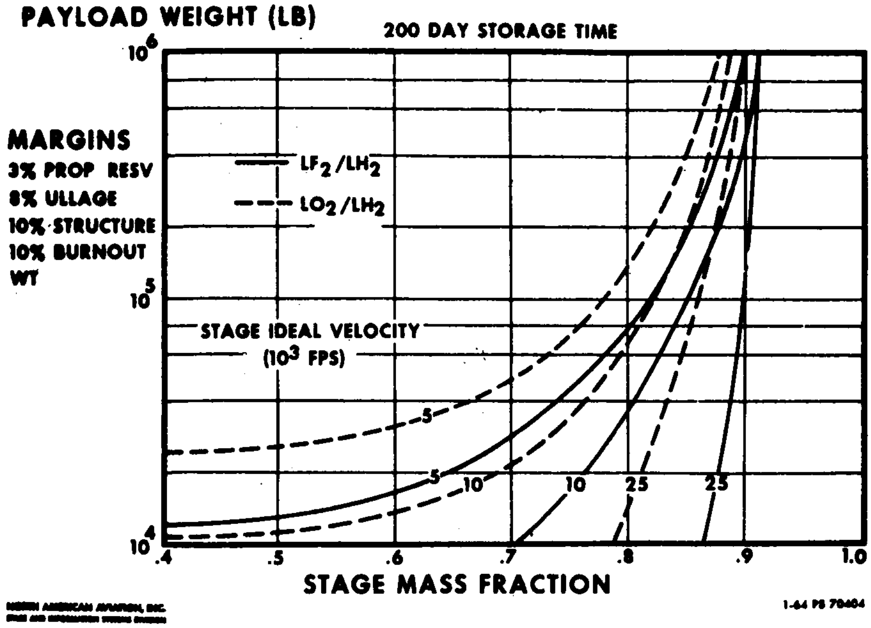


FIGURE 14

STAGE EFFICIENCY COMPARISON

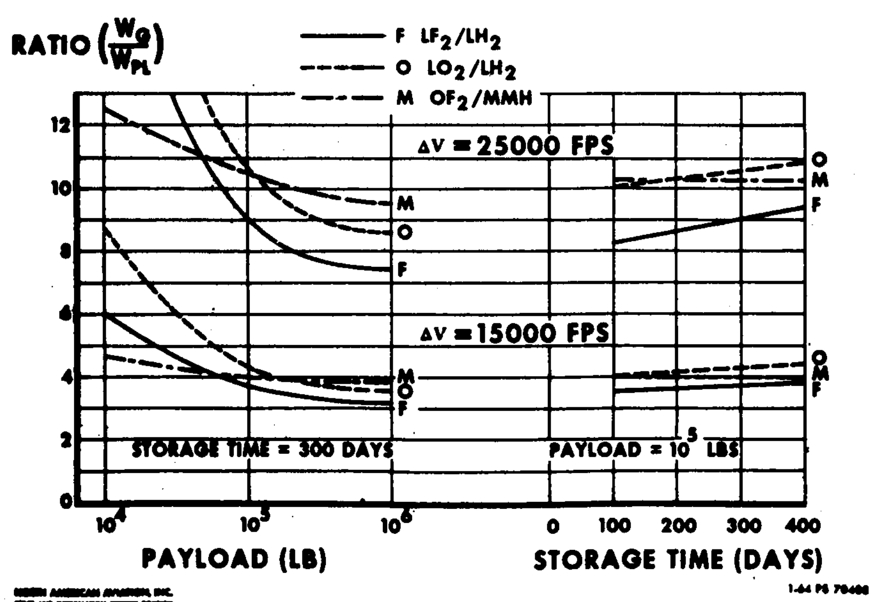


FIGURE 15

MINIMUM ENERGY MISSIONS

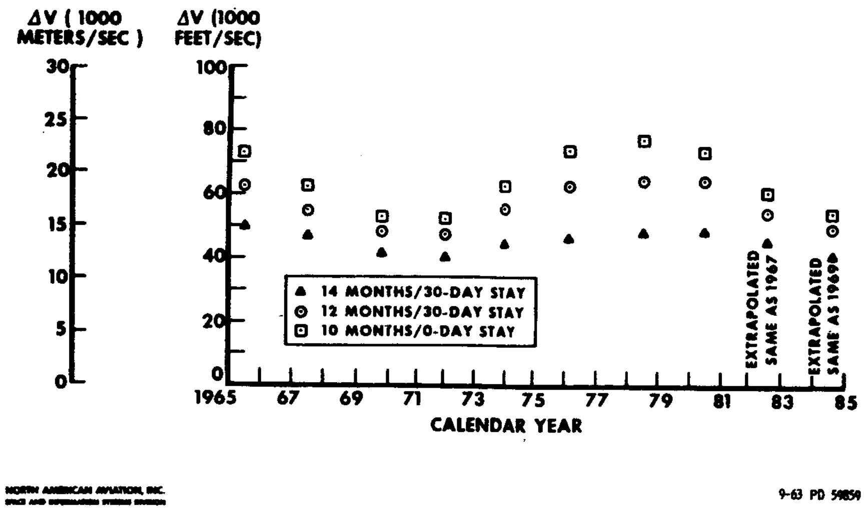


FIGURE 16

VENUS ENCOUNTER

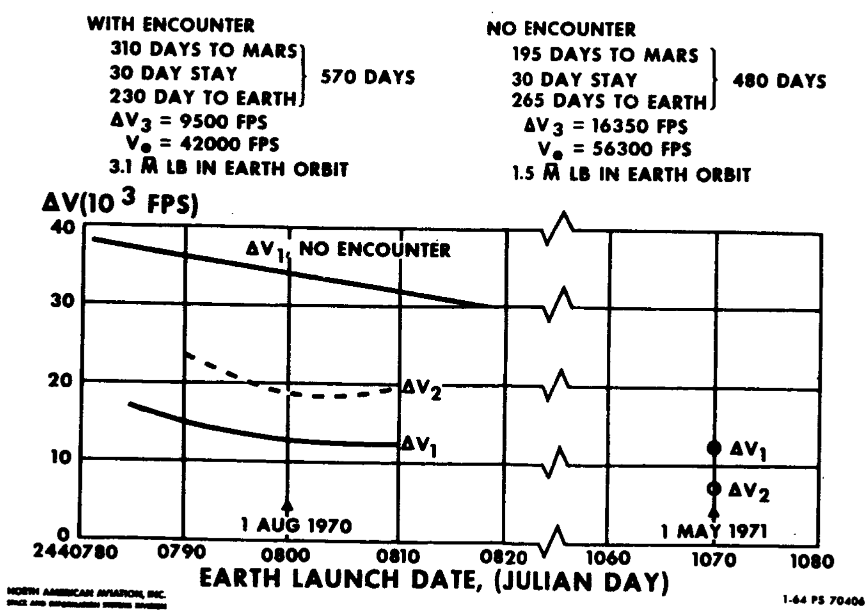


FIGURE 17

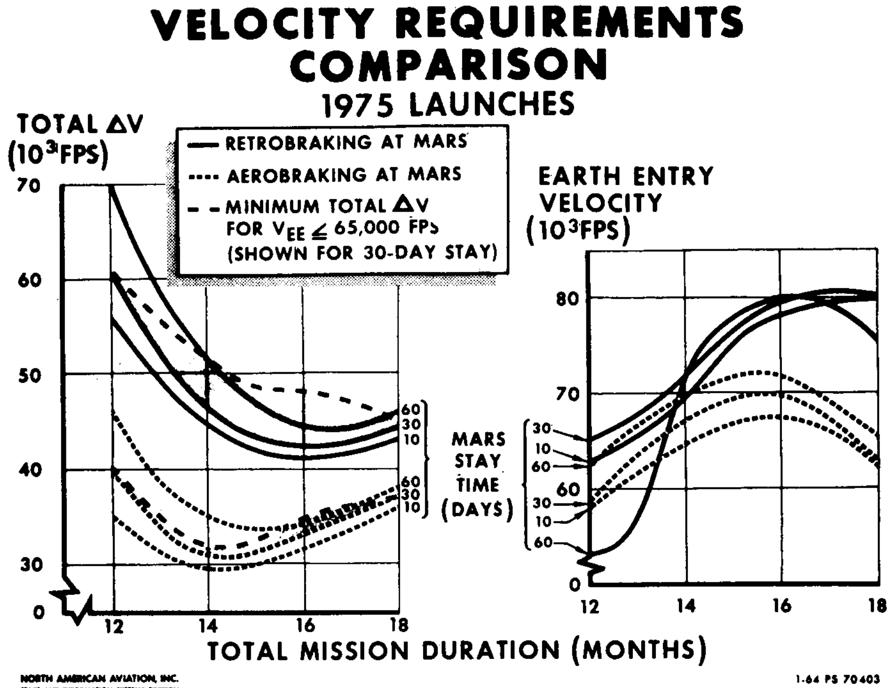


FIGURE 18

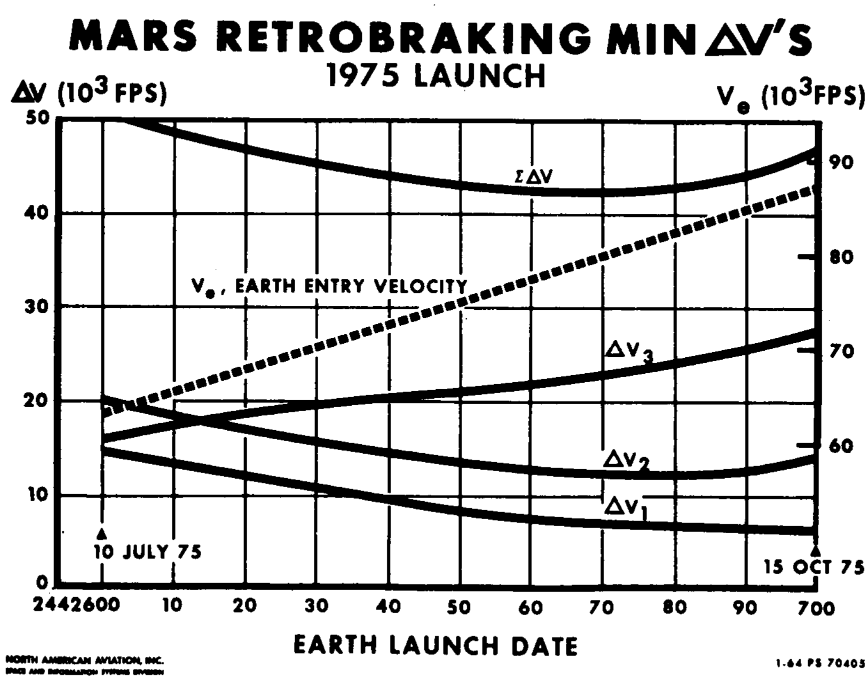


FIGURE 19

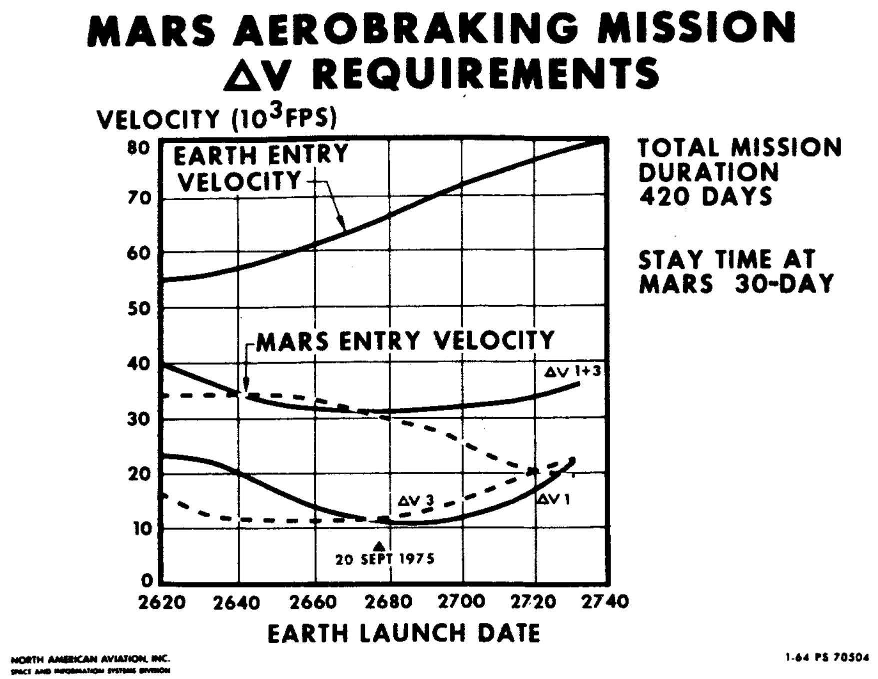


FIGURE 20

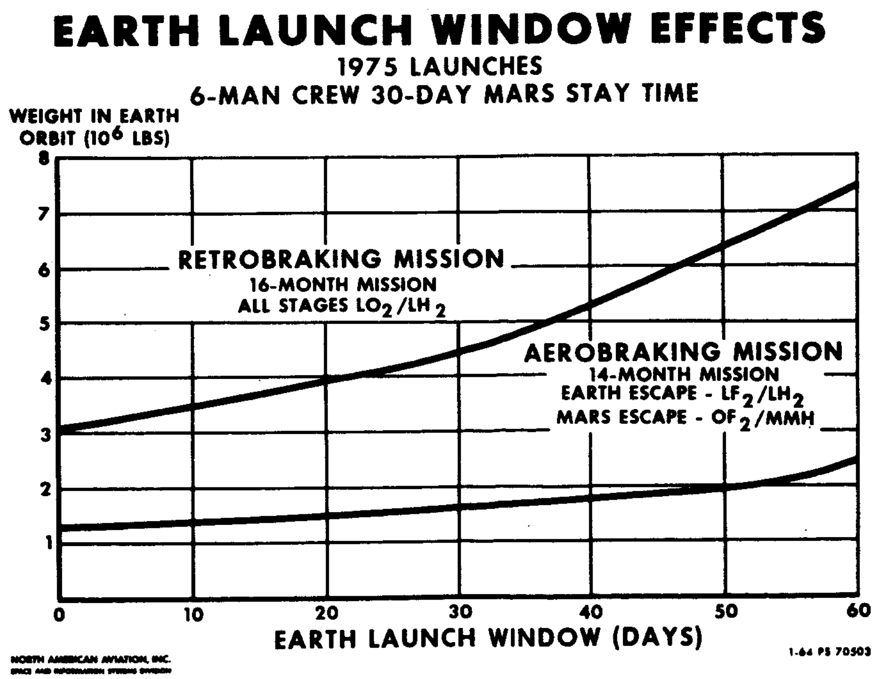


FIGURE 21

SYSTEM SENSITIVITY CHARACTERISTICS - 1975 MISSION*

PARAMETER	% CHANGE	% WT_{EO}
TRANS-MARS INJECTION ΔV	10	11.7
MARS RETROBRAKIING ΔV	10	8.4
TRANS-EARTH INJECTION ΔV	10	22.4
ERM WEIGHT	10	3.3
MM WEIGHT	10	6.0
MEM WEIGHT	10	0.6
TRANS-MARS I_{SP}	1	-1.14
MARS RETROBRAKING I_{SP}	1	-0.82
TRANS-EARTH I_{SP}	1	-1.08
TRANS-MARS ν_B	1	-1.95
MARS RETROBRAKING ν_B	1	-1.22
TRANS-EARTH ν_B	1	-1.86
ALL OF_2/MMH PROPULSION	-	10.93
ALL LO_2/LH_2 PROPULSION	-	7.54
ALL LF_2/LH_2 PROPULSION	-	- 19.85

*16-MONTH MISSION, 30-DAY MARS STAY TIME, 30-DAY EARTH LAUNCH WINDOW, OF_2/MMH INTERPLANETARY PROPULSION WITH LF_2/LH_2 TRANS-MARS PROPULSION

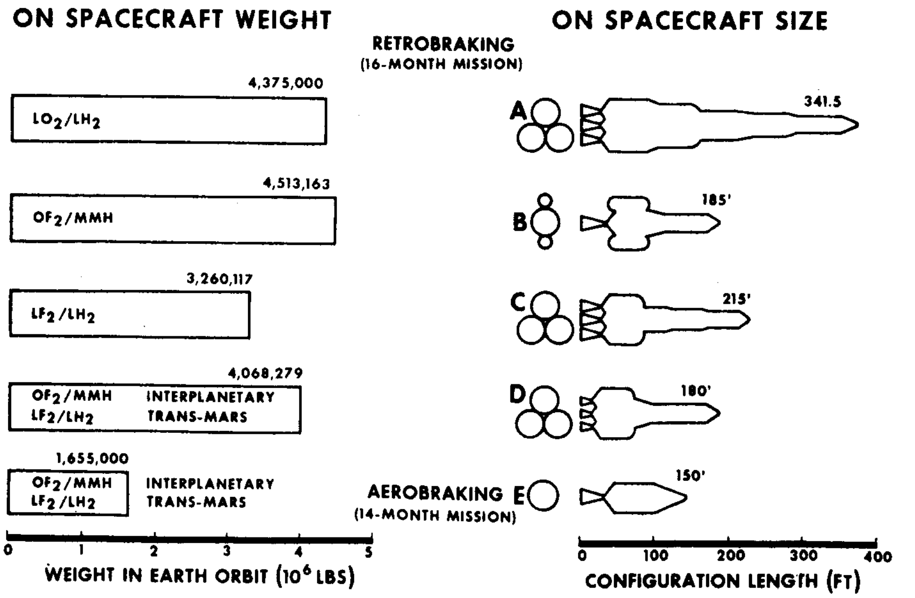
NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70527

FIGURE 22

EFFECT OF PROPELLANT CHOICE

1975 LAUNCH, 6-MAN CREW, 30-DAY MARS STAY TIME



NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70526

26985

PART 7

MANEUVERABLE DESCENT SYSTEMS FOR MARS LANDING

by

R. N. Worth
Northrop Corporation
Contract No. NAS2-1411

MANEUVERABLE DESCENT SYSTEMS FOR MARS LANDING

Prior to the first generations of manned planetary expeditions, barely a minimum amount of landing data will have been gathered by unmanned probes. A single point or small areas will have been searched and the data extrapolated to determine general landing conditions. Certainly no prepared landing sites will exist, and the success of these missions will depend entirely upon the variety of on-board systems which can be provided. One such system, which is being investigated under contract to the NASA-Ames Research Center, is a maneuverable descent system for Mars and Earth landings. Under an 11-month contract, Northrop Ventura is exploring the possibility of using a rocket propulsion system to augment the inherent horizontal translational capability of a gliding parachute system. Study was begun in July 1963 and is currently at the half-way point. Results presented here are indicative of trends, but should be considered tentative in the quantitative sense and subject to change as the investigation proceeds.

During this study four major portions of the landing system (the parachute system, propulsion system, control system, and impact system) are being investigated to uncover common parameters. The parameter interplay and the trade-offs possible to provide optimization of the overall system are being determined. From the start of the study, vehicle configuration concepts have been evolved for each of the weights considered. These are not detailed except to provide the range of parameter values necessary to calculate landing systems weight allocations, entry trajectories, landing trajectories, and system dynamic characteristics. A sequence of events (Figure 1) has been evolved to describe the landing operation from the time of parachute deployment and provide a flexible framework within which the study would be conducted. The figures quoted here are compatible with the Schilling mean atmospheric model.

Parachute deployment is to be initiated at an altitude of 92,000 ft. A 3-minute period is then available to establish a circling mode of descent, to deploy the viewing navigation, and impact system devices, to select the landing area desired, and to achieve the correct heading at maximum glide angle. This starts the actual landing operation at an altitude of 70,000 ft. Wind effects can be minimized since the speed of a parachute system, designed for example to a 100-ft/sec terminal sea level velocity, will be capable of operating at a horizontal velocity of 150 ft/sec at this altitude.

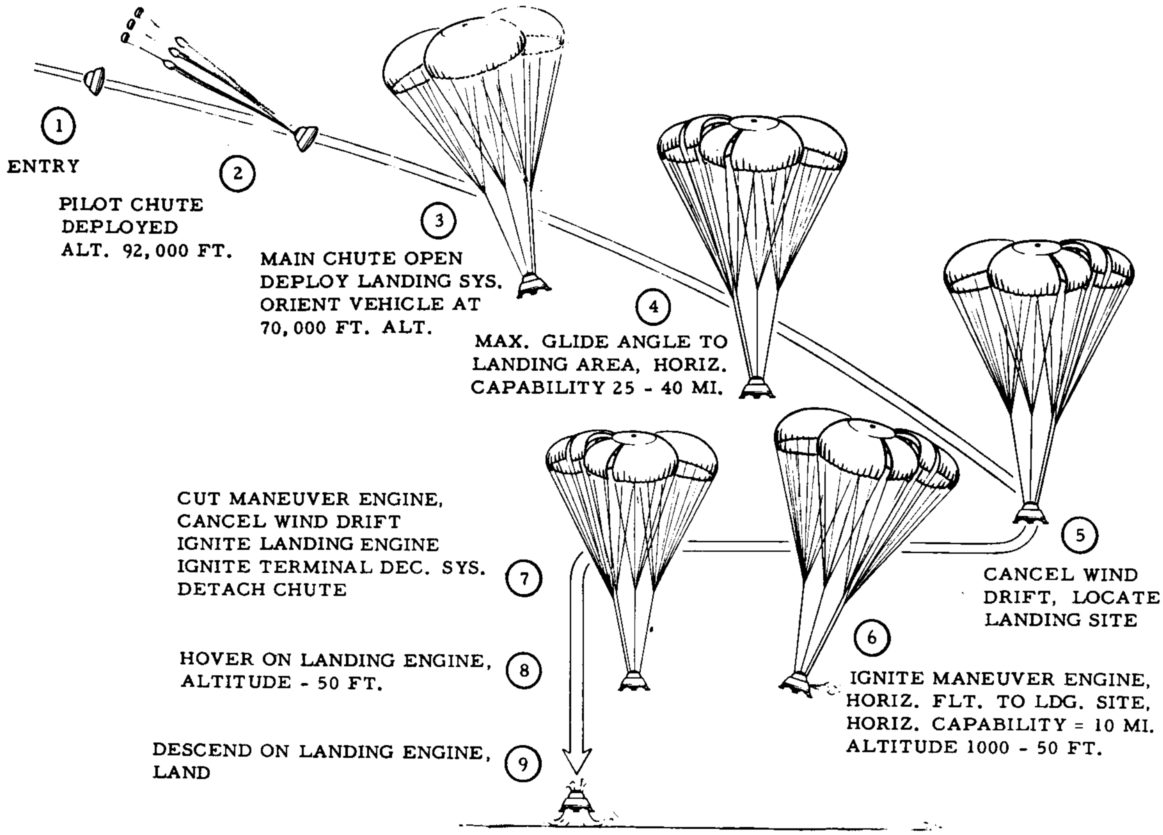


Figure 1. Landing Sequence for an Entry Vehicle for Mars

The landing trajectory is considered to have five separate phases. First is the gliding phase which involves the parachute system alone. During this phase the vehicle descends from 70,000 ft to an altitude of approximately 1000 ft. Depending on the parachute design a horizontal translation of up to 40 miles can be achieved during this operation. Upon reaching the selected landing area, a circling mode of operation may again be established. By the time the 1000 ft altitude is reached the actual landing site is selected, or at least the direction of final search is established, and the second phase of operation is started. This is the rocket augmented glide phase combining the rocket and parachute systems for terminal site selection and obstacle avoidance. This phase, through the use of a horizontally oriented rocket engine, can achieve up to 10 miles of horizontal flight. Upon reaching the selected

landing site, the maneuver engine is cut and the vehicle descends to an altitude of 100 ft. At this time the landing engine is ignited and the terminal deceleration phase is initiated. This brings the vehicle to rest 25 to 50 ft above the ground in a hovering mode on the thrust of the landing engine. The vehicle is then eased down to impact on a landing leg system at a vertical velocity of 10 ft/sec and a horizontal velocity of less than 5 ft/sec. For the purposes of this study the launch engine is considered to be used as the landing engine since off loading is small. The only weight chargeable to the landing system will be the weight of propellant consumed, the increase in tankage, and any increase in engine weight due to using the launch engine for this operation.

With the parachute gliding capability available, the main chute system should be deployed at as high an altitude as possible. To determine the effect of the various Mars atmosphere models on the parachute deployment altitude, entry trajectories have been calculated from high and low orbits for all the atmospheres (Figure 2). Feasible deployment altitudes for both main and drogue parachute systems

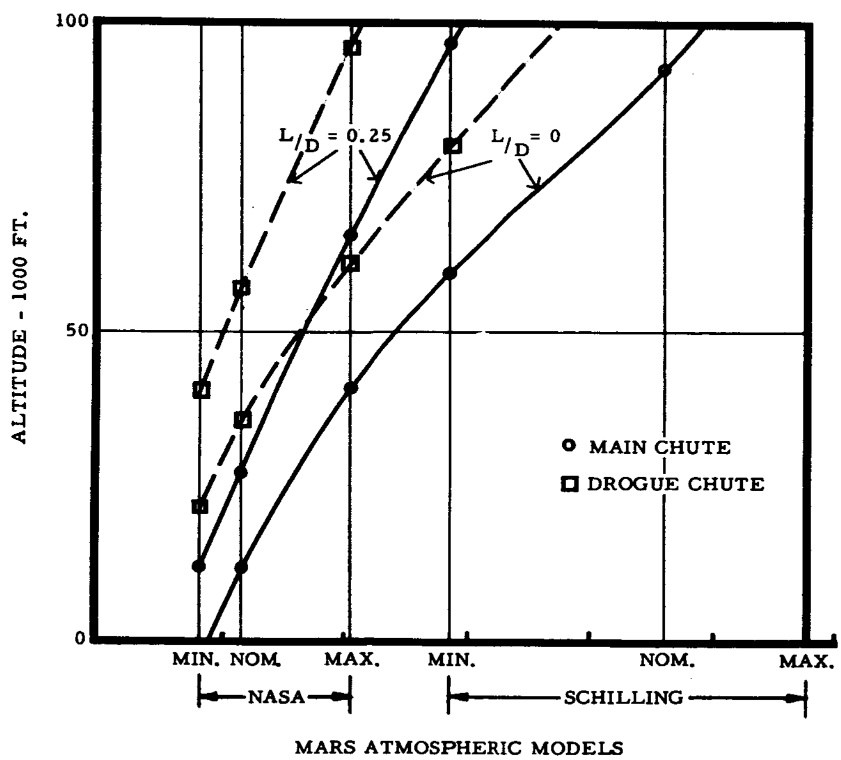


Figure 2. Parachute Deployment Altitudes (Main chute has Mach 1.2 as limiting deployment velocity; drogue chute deployment occurs at Mach 2).

have been plotted for the different model atmospheres. They are based on an entry vehicle ballistic coefficient of fifty, and for vehicle lift-to-drag ratios of zero and one quarter. Schilling models allow direct deployment of the main chute at altitudes above 60,000 ft; however, reasonable deployment altitudes for the NASA models will probably require the addition of a drogue system. The use of the vehicle lifting capabilities may also be required, especially for the minimum model. Calculations made for ballistic coefficients up to 150 and show the desirability of keeping this configuration coefficient below a value of one hundred. Essentially, through the judicious use of the system design, parachute landing systems may be efficiently used over the entire range of atmosphere models.

PARACHUTE SYSTEM

The steerable chute concepts serving as the basis for the parachute system investigation have been derived from the aerosail configuration work and wind tunnel tests conducted over the past few years. The aerosail chute configuration (Figure 3) consists of a single or cluster of generally nonporous canopies. A double-flap system is located in the aft portion of the canopy with control lines running from the vehicle to provide pilot control of the

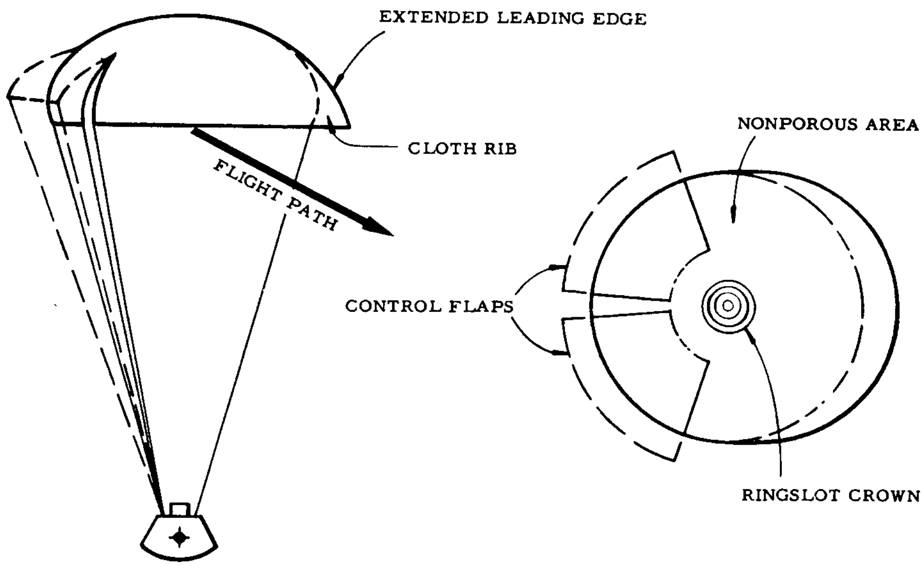


Figure 3. Aerosail Parachute Configuration

flap position. In addition to the flap a leading edge extension with cloth ribs is incorporated into the canopy design. The chute system is deployed with the leading edge extension retracted and the flaps locked into a neutral position to provide a symmetrical configuration until full inflation is accomplished. Position of the flaps may be controlled in unison to allow variation in the flight path angle from vertical up to the maximum capability of the chute. Flap position may also be controlled differentially to accomplish turning maneuvers. The chute will stay essentially horizontal, with the vehicle hanging directly beneath, over the entire range of glide angle change.

Wind tunnel tests of 12-ft-diameter models have been conducted in the Ames Research Center 40 ft by 80 ft tunnel under a NASA-MSC contract.

Test runs included a single canopy configuration at a maximum flap setting. In Figure 4, the canopy itself is stable in the glide configuration and has attained a high L/D glide position. A maximum L/D of 1.8, based on force data recorded during this test, is indicative of a horizontal travel capability 1.8 times the vertical travel.

Results of the aerosail tests also indicate a properly grouped cluster can achieve a higher L/D value than the single chute. This is due primarily to the higher effective aerodynamic aspect ratio and lower profile drag of the cluster. These configurations are no longer primarily drag devices but are actual flying airfoil shapes.

Other tests have been conducted in the FDFR vertical wind tunnel under an Air Force contract which included modified models of the type demonstrated at Ames. Single canopy simulation has been made of a cluster of three parachutes during one of the test runs (Figure 5). The cluster was in a gliding attitude representing a high L/D. This configuration utilizes the shaped leading edge of the aerosail model with a leading chute and two pushing chutes incorporating the flaps. The two aft chutes have been trimmed to mate with the leading chute, giving the single canopy effect. The portion of the canopy seen separating the chutes was found to be required to maintain air pressure within the leading chute. This prevents the collapse of the leading edge under the pressures involved with higher lift-to-drag ratios. The control lines to the two flaps are separate systems to allow differential flap control. Models 12 ft in diameter, of this and other configurations are currently being fabricated for tunnel tests at Ames. Drop tests are planned for next spring with chute sizes up to 40 ft in diameter.

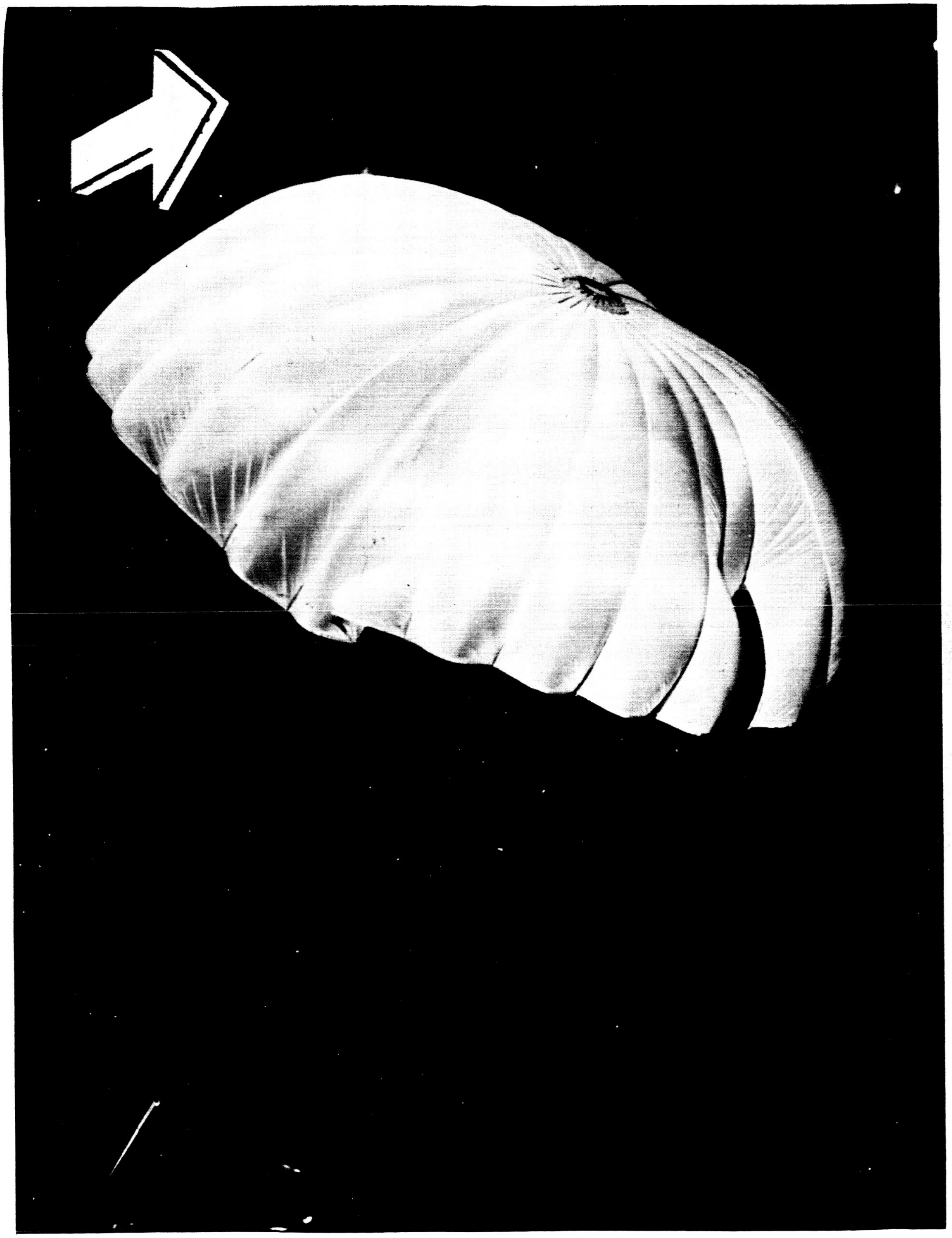


Figure 4. A 12-ft Gliding Model Parachute Under Test (Pylon attach point is at upper left; air flow is horizontal as shown by the arrow.)

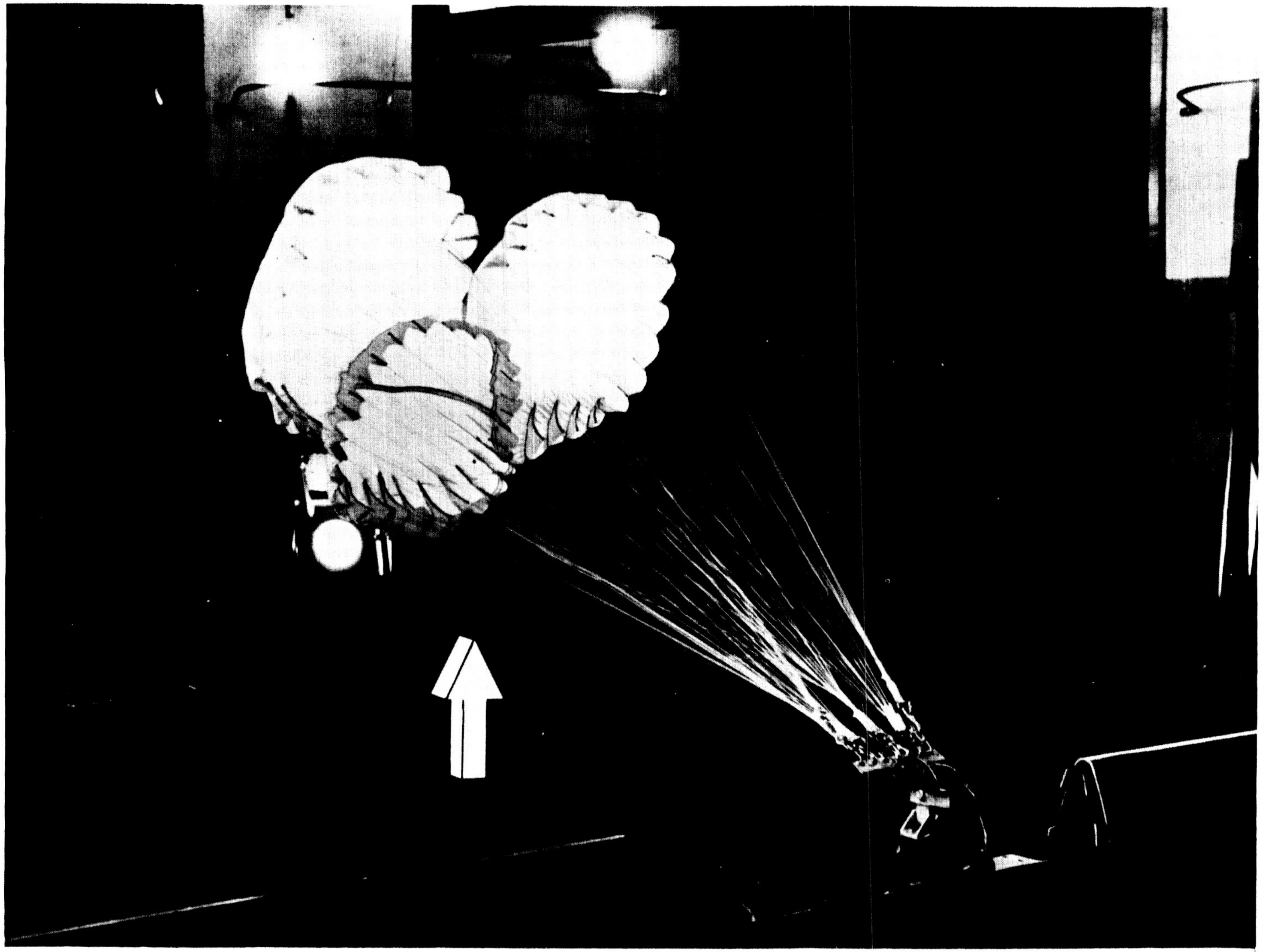


Figure 5. A 3-ft Cluster Test (Arrow indicates air flow.)

Estimates have been made as to what can be achieved from gliding chute development over the next 10-year period. Gliding parachute development in the past has been extrapolated to estimate the aerodynamic glide parameters that can be achieved in the 1970 era (Figure 6). The solid lift-to-drag ratio curves are derived from the Ames test results for single parachutes and for three parachute clusters along with the accompanying drag coefficient (dot-dash line) curves. These are presented as what is currently in the development stage. Maximum lift-to-drag ratios of 1.8 can be achieved with a single canopy while the self-inflating cluster can provide an L/D of better than 2.0. Operational systems of this capability could be available within the next few years.

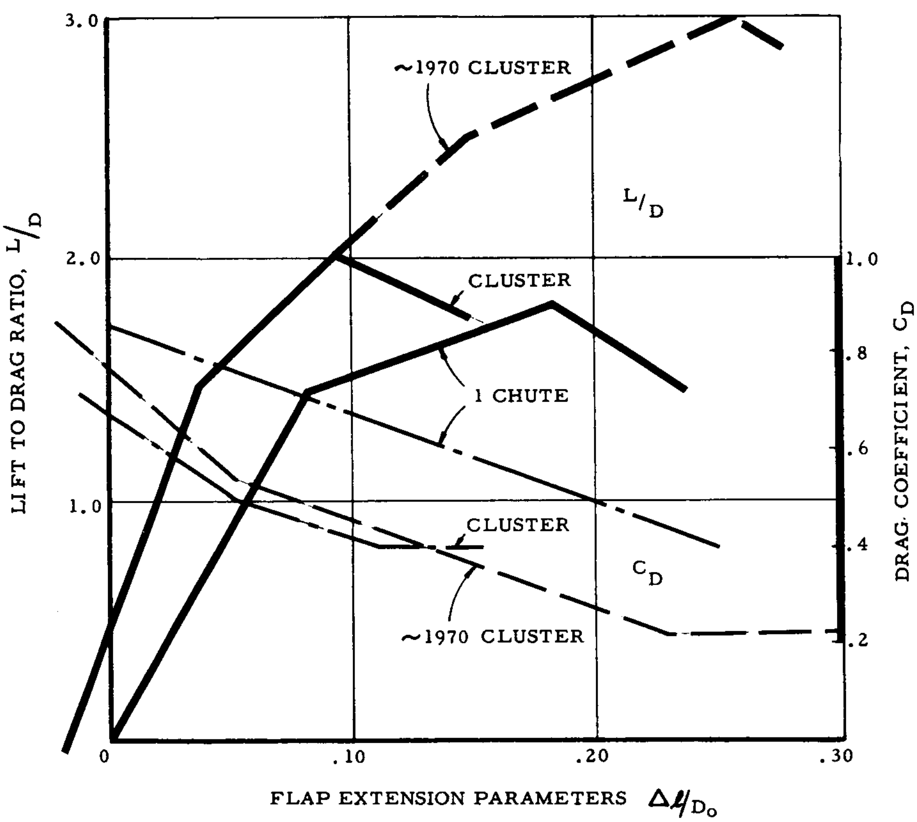


Figure 6. Estimated Lift-to-Drag ratios (left ordinate) and Drag Coefficients (right ordinate) vs Flap Extension Parameters from the Current Ames Tests and Projected into the 1970 Era.

Further development of the aerosail concept, coupled with pressurized leading edges and stiffening members to maintain aerodynamic shape at the higher L/D's, should yield lift-to-drag ratios better than 3.0 as shown by the heavy dashed line for the 1970 era. The drag coefficient of this cluster is shown by the light dashed line. These parachutes are basically stable in their gliding configurations and can provide a stable descent at a given flap setting from about one-half their design L/D maximum up to the maximum value. Another aspect is that the flight velocity is relatively constant over the entire range of glide angles achievable.

Other design limits have been established. Maximum parachute diameter for this study range as high as 300 ft. This is based on the present large parachute test programs involving ringsail parachutes up to 124 ft in diameter, and also upon heavy drogue system deployment tests. The original limit on flight velocity was placed at 150 ft/sec, but this is currently being explored to see if higher values can reasonably be used.

To arrive at parachute system weight estimates, the starting point was the ringsail parachute weight formulae that have been verified from specific designs. These were modified to include the various differences inherent in the aerosail system. Based on an L/D capability of 2.0, the total parachute system weight fraction has been plotted against the design flight velocity for the various Mars model atmospheres (Figure 7). This weight fraction includes only the parachute system and its controls. The plot was made against velocity to allow an optimization to be made with the primary deceleration system weight fraction which increases with the descent velocity, and later with the maneuver system where range is a function of design velocity. The number of maximum-diameter parachutes which can be clustered, and weight curves for various vehicles permit determination of a minimum design flight velocity. For instance with the 100,000-lb vehicle and a three-parachute cluster, the NASA nominal atmosphere gives a minimum descent velocity of 85 ft/sec. If the Schilling nominal model should be correct, then the minimum velocity would reduce to 40 ft/sec.

One of the major conclusions of this study so far is derived from this plot. This is, that even with the NASA minimum model atmosphere, parachute landings can be accomplished for weight fractions of less than 5%. The parachutes will be large but even the 200,000-lb vehicle with a cluster of three parachutes can be landed within this weight fraction.

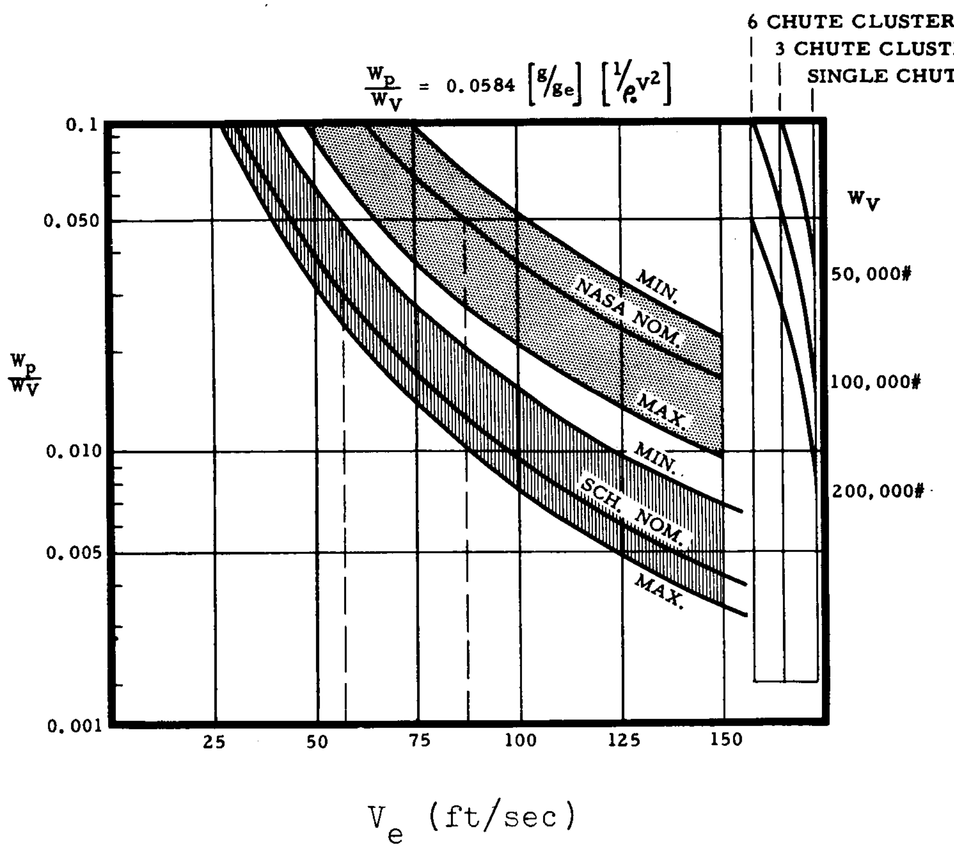
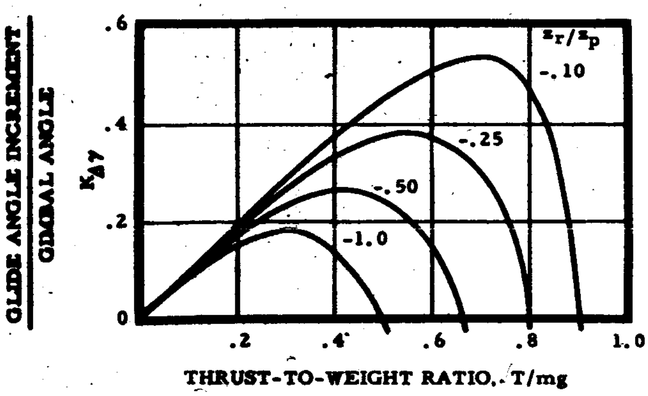


Figure 7. Weight vs Terminal Velocity, Aerosail Parachute System (Curves on the right show vehicle weight for various clusters shown.)

GUIDANCE AND CONTROL

The guidance and control aspects of the combination of the parachute and propulsion systems have been analyzed to determine the control requirements, dynamic effects, and handling characteristics of the vehicle during the landing operation. Investigation included the trim conditions as a function of rocket mounting location as well as the magnitude, direction,and control of the thrust vector. Glide augmentation from the main landing engine and also from a smaller auxilliary thrust (or maneuver) engine was analyzed separately. The main landing engine is assumed to have a large thrust-to-weight ratio with limited vector control, while the auxiliary engine has a smaller thrust but a larger allowable gimbal angle.

The main landing engine can make a small contribution to the glide angle (Figure 8). The coefficient of glide capability ($K_{\Delta\gamma}$), which is the ratio of the incremental



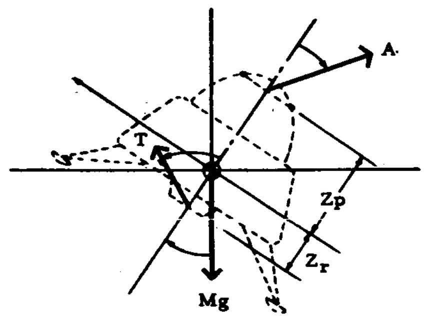


Figure 8. -Main-Landing-Engine Contribution (Sketch on right presents assumptions.)

glide angle change to the engine gimbal angle, has been plotted against the rocket thrust-to-vehicle weight ratio for various mounting locations. Assumptions are that (1) the main landing engine is mounted below the center of gravity (Z_r), and (2) the parachute is attached at a single point on the vehicle centerline above the center of gravity (Z_p). For a given configuration, as the engine location ratio (Z_r/Z_p) becomes more negative, the engine is located further below the center of gravity. T is the thrust of the engine and A is the parachute force. The main conclusion is that the incremental increase in glide angle due to the application of thrust is always less than the gimbal angle. In addition, the landing engine would have to be throttled considerably to avoid a negative contribution to

the glide angle since the curves go negative before a thrust-to-weight ratio of one is reached.

For a nominal configuration the Z_r/Z_p ratio would be approximately minus 0.50. With a gimbal angle of 10° and throttling to a thrust-to-weight ratio of 0.4 the glide angle augmentation would be less than 3°, or about 4% of a parachute system with an L/D = 2.0 capability. These results indicate that the main landing engine would be very inefficient when used for glide augmentation.

This conclusion has led to an investigation into the glide angle contribution possible from a small maneuver engine which can deliver thrust in any desired direction. The results indicate that a near optimum thrust direction could be achieved by a fixed engine delivering thrust normal to the vehicle center line. The glide angle contribution of such a system has been estimated.

In Figure 9, the actual incremental glide angle in degrees and the parachute force-to-weight ratio are plotted vs the engine thrust-to-weight ratio for locations of the maneuver engine varying between the same point as the parachute attachment, shown by $Z_r/Z_p = +1.0$, to an equal distance below the center of gravity, shown by $Z_r/Z_p = -1.0$.

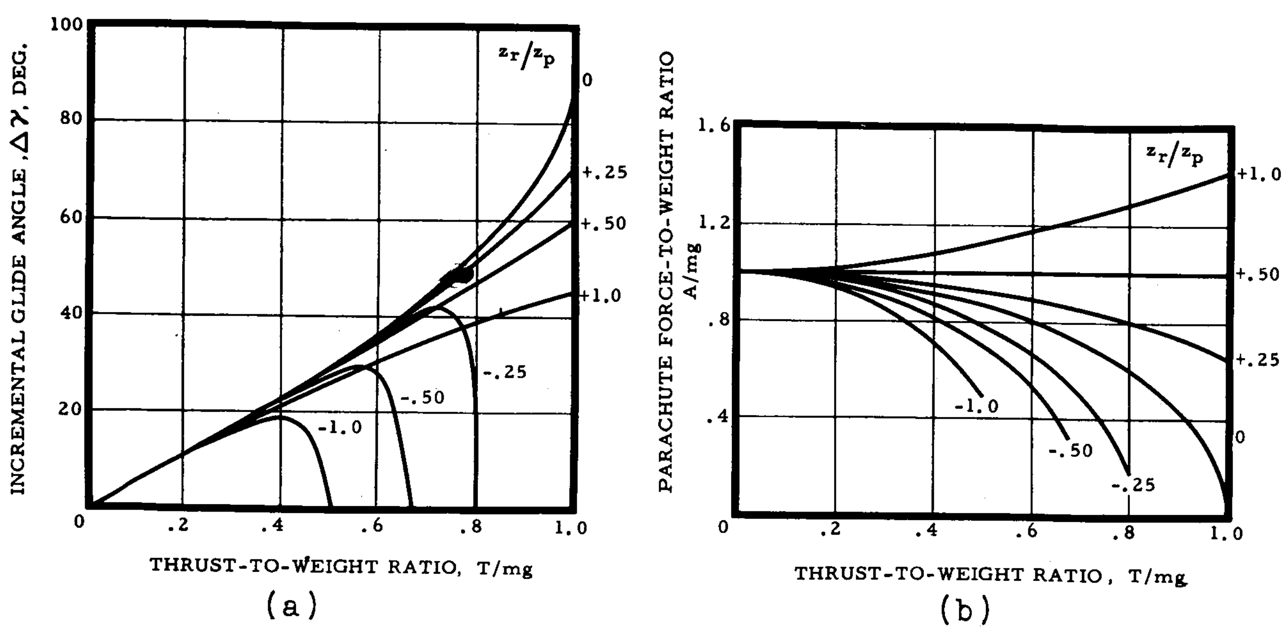


Figure 9. Maneuvering Engine Contribution

If the thrust-to-weight ratio required is less than one-half, then an engine location anywhere between the center of gravity and one-half the distance to the parachute attach point could yield a glide angle increment very near optimum (Figure 9a). Location at the halfway point (where Z_r/Z_p = +0.50) is preferred since the application of thrust at this point has no effect on the parachute force and therefore does not change the design flight velocity, (Figure 9b).

Coupling these results to the various parachute gliding capabilities considered gives the maneuver engine thrust-to-weight ratio requirements for horizontal flight vs the parachute $(L/D)_{max}$ capability for two locations of the nongimballed, normal thrusting maneuver engine (Figure 10).

These locations are on the center of gravity and at the point midway between the center of gravity and the parachute attach point. The conclusion drawn from this plot is that the thrust required decreases with increasing parachute maximum gliding capability, and that, for the ranges considered in this study (where L/D is greater than 1.5), the location of the engine to maintain a constant design velocity with no increase in the parachute loading will require very little additional thrust.

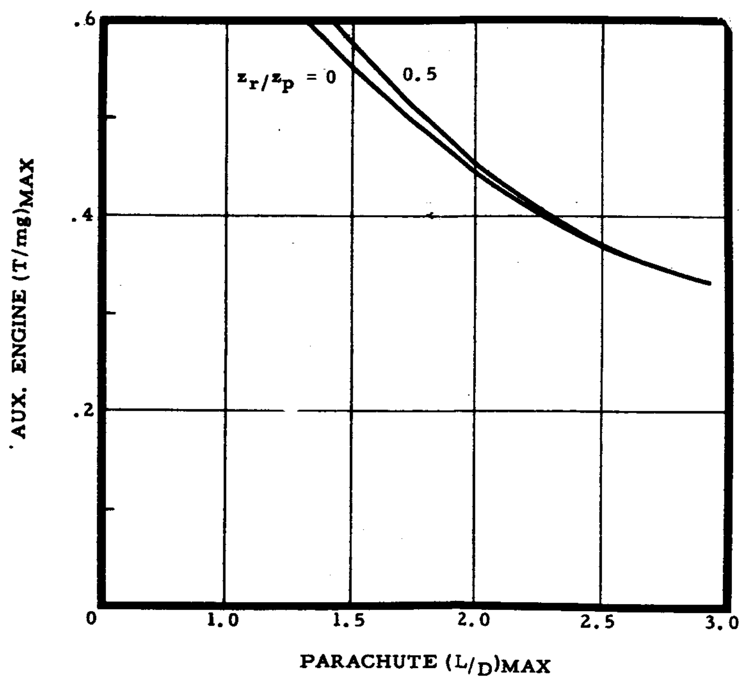


Figure 10. Horizontal Flight Requirement

Horizontal flight may be achieved by applying a horizontal thrust fraction of 0.45 the vehicle weight for a parachute design of $(L/D)_{max} = 2.0$. Increasing the parachute L/D capability to 3.0 reduces the required thrust value of one-third the vehicle weight.

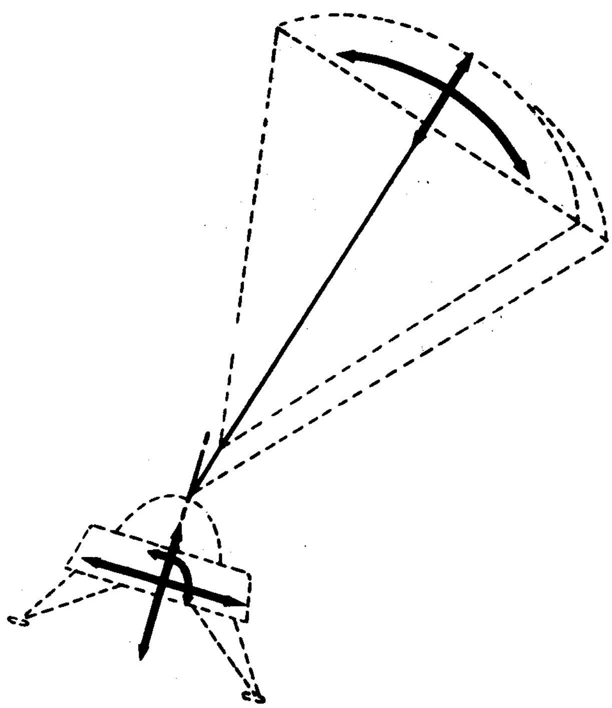


Figure 11. Planar Model

To continue the control analysis work, a mathematical model of the system has been necessary for use in the landing trajectory dynamic analysis and to demonstrate handling characteristics of the total system. A planar model (Figure 11) has been developed incorporating 3 degrees of freedom for the vehicle plus 2 degrees of freedom for the parachute. The 3 degrees of freedom of the vehicle are conventional being the motion along the two axes of the body plus a rotation. The two degrees of freedom for the parachute are line stretch, or motion toward and away from the vehicle, and a pitching or oscillating motion about the parachute attach point. The parachute force is transmitted through a single attach point and the engine thrust is applied at any arbitrary location.

The derived nonlinear equations have been linearized to compute the characteristic modes of one of the gliding parachute wind tunnel tests conducted at Ames. The computed modes for the 12-ft-diameter wind-tunnel model, tied to a fixed point, are an elastic mode of high frequency and well damped, and a pitch mode which is unstable at low L/D and stable at high L/D. The modes seem to be well separated in frequency. While the elastic mode has not been observed in the wind tunnel test (which might be due to the high frequency and the small amplitude), the pitch mode observed

exhibits the same characteristics as those computed, thus providing a check on the mathematical model.

The dynamic analysis work and an investigation into handling characteristics are presently being conducted. Only preliminary data are available at this time; however, indications are that stable descent may be achieved over the entire range of L/D (from vertical to maximum glide angle) through the use of control feedback loops in the vehicle attitude control system and the parachute flap control system.

PROPULSION SYSTEM

To provide the design and parameter data needed for the overall study, individual portions of the propulsion system also have been analyzed. Parameters of major importance have been established in terms of mission requirements, system considered, and performance available. The mission parameters are thrust-to-weight ratio, thrust duration, and vehicle weights. The system parameters are specific impulse (as a function of propellants considered and the system design), chamber pressure, expansion ratio, as well as engine location and direction of thrust application. The performance parameters are propellant-to-system weight ratio and propulsion system-to-vehicle weight ratio. A parametric analysis to establish optimum system characteristics in terms of the overall landing systems analysis is being conducted.

The physical properties and the range of performance of presently available and possible future liquid, hybrid, and solid propellants have been studied. As a result of this study, three advanced but currently available liquid propellants have been chosen for equal consideration in the system analytical work. These are OF_2/MMH, oxygen/hydrogen, and fluorine/hydrogen. Solid propellants have been considered only for the terminal deceleration phase where the high thrust level and short duration requirements result in a lighter system. The propellants selected encompass a broad range of values of specific impulse and density. During the study all three propellants are being evaluated in pressure fed systems. In addition, an expander cycle pump fed system is being studied for use with the oxygen/hydrogen propellant only.

Some of the subsystem design assumptions include (1) plug or forced deflection nozzles are assumed for the thrust chambers; (2) for the pressure-fed chambers, cooling is accomplished ablatively with a radiative skirt and

throttling is obtained through the use of variable injector areas; (3) the pump fed engine is regeneratively cooled and is throttled at the turbine; (4) for tankage, both spherical and toroidal designs are considered along with the required insulation; (5) helium in high pressure bottles stored within the oxidizer tank is assumed for the pressurization system coupled with a heat exchanger.

Performance calculations assume the basic thrust chamber and nozzle performance to be 94% of the theoretical shifting equilibrium value. The effect of a finite back pressure is included since the maneuver system operates at sea level full time in addition to being throttlable. An optimum expansion ratio exists for each chamber pressure and the performance varies with pressure. The effect of throttling on performance in an atmosphere is minimized by the use of a plug or forced deflection nozzle but losses still exist. Equations have been derived to present the propulsion system weight fraction in terms of the important vehicle, system, and design parameters. Computer programs evaluate interplay between these parameters in order to establish optimum weight systems.

Some of the preliminary results of the analytical computer programs that have been completed include a nomograph (Figure 12) for the separate, solid propellant Mars terminal deceleration system. This nomograph starts from the vehicle descent velocity, and proceeds through the propellant effective I_{sp}, the ratio of propellant weight to total rocket system weight and the allowable accelerations on the vehicle, in terms of earth g's, to arrive at a system weight fraction. An automatic control system coupled with a peripheral rocket motor design can reduce the weight fraction further through the ground augmentation effect.

This graph represents a number of design inputs; however, for a given preliminary mission analysis effort the I_{sp} may be dictated by the years and propellants being considered for the mission. A propellant mass fraction of 0.85 may be picked as representing a reasonable value for a good system design. If the allowable acceleration is assumed at a value of 4 the results are still reasonable for higher "g" loadings.

In addition, if there is a requirement for a hover mode for final evaluation of the landing site, the ground augmentation effect cannot be considered. The equations thus reduce to a simplified form for input into the overall landing system equations during the preliminary mission

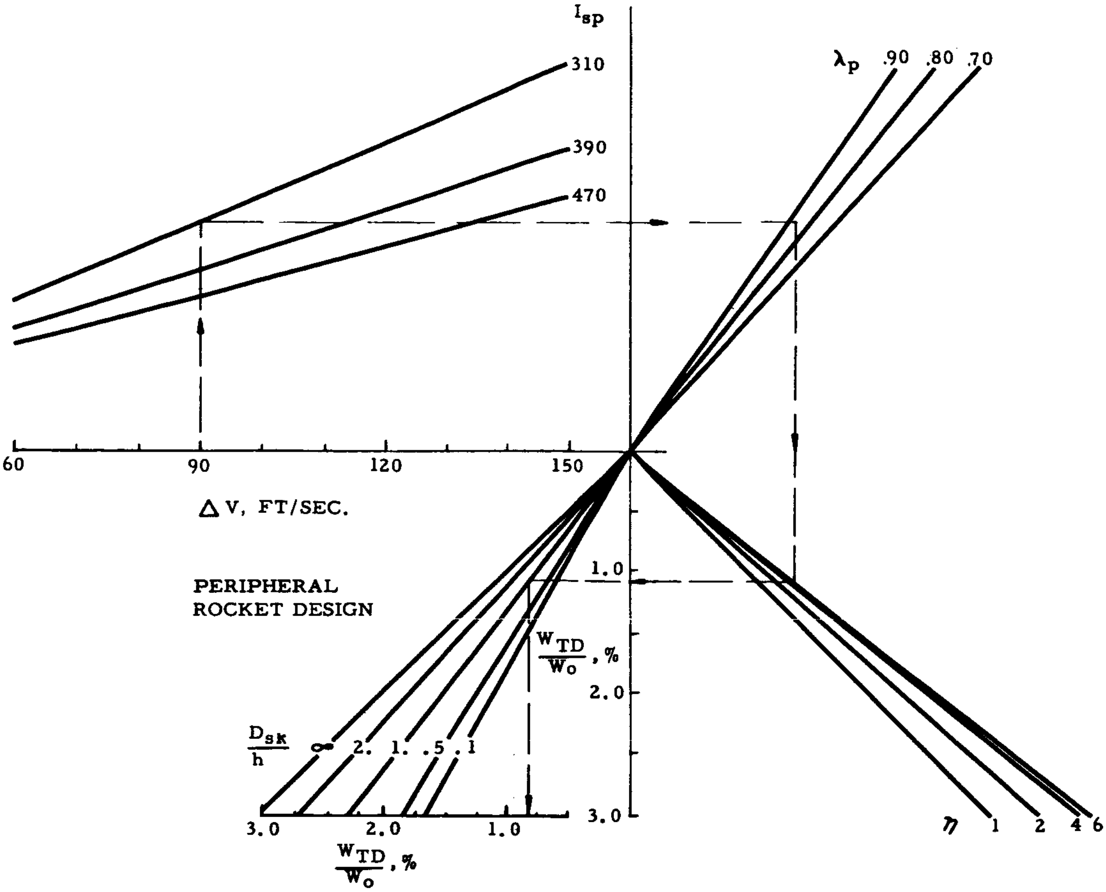


Figure 12. Terminal Deceleration System

analysis effort. The basic tool is available, however, for more refined missions analysis work and for system optimization as vehicle configuration and design progresses.

Computer programs have also been set up for the liquid propellant maneuver propulsion system. One analysis (Figure 13) shows the weight fraction of the oxygen/hydrogen pressure-fed maneuver system vs operating duration for thrust-to-weight ratios of 0.5, 0.4 and 0.3. The curves are based on optimum chamber pressures and expansion ratios for each thrust level. Starting with an assumed parachute glide angle capability, the parachute, terminal deceleration, and impact system weight fraction can be optimized. The difference in this value and an allowable fraction for a given mission is the available weight fraction assigned to

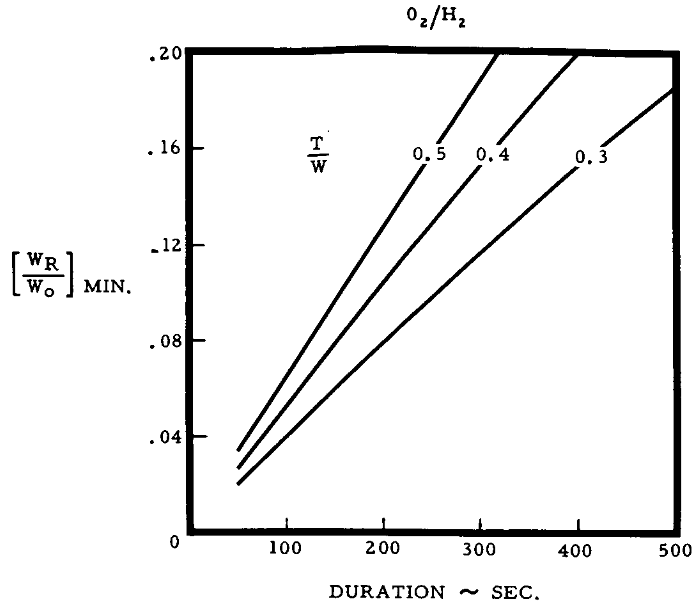


Figure 13. Maneuver Propulsion System Weight Fraction (Curves are based on optimum chamber pressures and expansion ratios for each thrust level.)

the maneuver system. Entering the plot at this weight fraction and using the thrust-to-weight ratio required for horizontal flight permits determination of the maximum duration of operation. This provides another input to be used in determining an optimum parachute design velocity to yield the maximum available horizontal translation.

For the range of weight fractions available, the curves may easily be represented as straight lines (Figure 13). This means that a simple form of equations may be included in the overall landing system parametric analysis and still yield accurate results. Note that these curves are for one propellant only, and the use of fluorine/hydrogen propellant, for example, would result in a 10 and 15% increase in duration time for the same system weight fraction.

TENTATIVE RESULTS

At this point in the program only tentative results are available. The quantitative data are approximate and are presented only to show trends the study is indicating.

For the nominal Schilling Mars model atmosphere, the horizontal range capability from an altitude of 1,000

ft vs the landing system weight fraction would be as shown in Figure 14. To prepare these curves, the altitude limit is arbitrarily picked at 1,000 ft, and an arbitrary 15% limit is placed on the landing system weight fraction. Four systems are included. System I is the minimum weight landing system, based on a ringsail parachute design with no range capability, which has a weight fraction of 3.2%. Adding a 30-sec hover requirement and translating on the main landing engine gives system II. This system weighs out at 6.5% for 0 range while the cut-off limit can give 1.3 miles of horizontal translation.

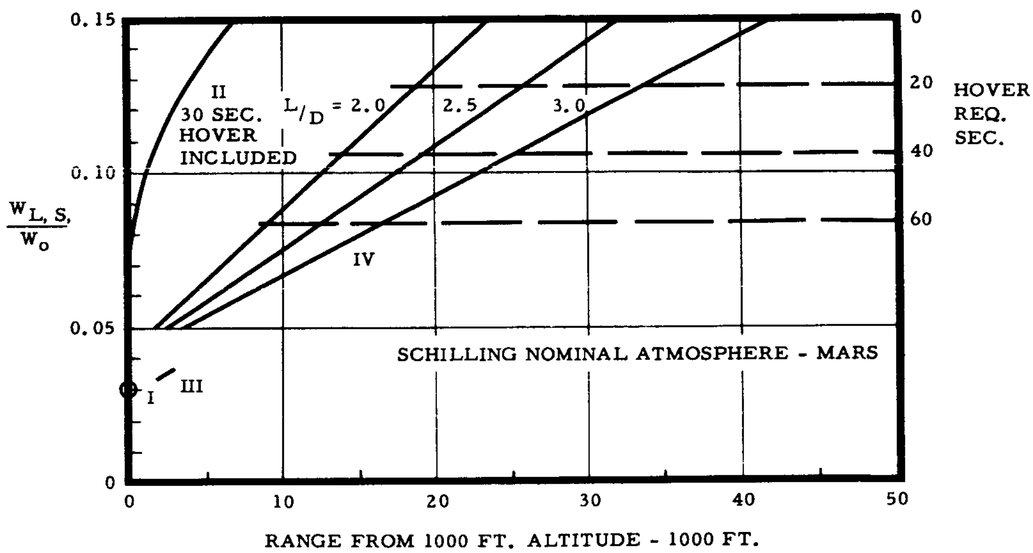


Figure 14. Range Capability

System III is the gliding parachute system without thrust augmentation. This shows an increase in weight fraction from 3.4 to 3.7% to increase the L/D capability from 2 to 3 or the range from 2,000 to 3,000 ft from the 1,000 ft altitude. Adding the maneuver system basic weight gives System IV, and the lower thrust requirement lets the system weight fraction decrease from 5.1% at L/D = 2 to 4.9% at L/D = 3. The remaining weight fraction is then used for propellant to give a range of 4.5 miles for the L/D = 2.0 system and 8 miles for the L/D = 3.0 system.

If an operational requirement exists for a hover and easing descent on the landing engine, with a 40-sec

duration assumed from an altitude of 50 ft, the weight of the propellant burned during this maneuver is deducted from the cut off limit weight fraction and reduces the maximum range capability from 8 to 5 miles for the L/D = 3.0 systems. On the other hand, if no hover requirement exists, the use of a peripheral terminal deceleration system incorporating automatic ignition and ground augmentation can effect a 25% increase in range capability.

The landing system weight breakdown for the L/D = 3.0 system is as follows:

Parachute System	0.9%
Terminal Deceleration System	1.7
Maneuver System	1.2
Impact System	1.1
Total weight fraction	4.9%

These weight fractions and the ranges shown previously are based on optimizing the design velocity in terms of the parachute and terminal deceleration system weights. The L/D = 3.0 system has a design velocity of 150 ft/sec. Further study indicates that when the maneuver system weight is also considered, an increase in this design velocity, while increasing the basic weight, may still give a greater range capability. The cut-off limit will also enter the trade-off curves as a contributing factor.

In summary, the main conclusions derived so far are that:

1) Utilization of parachutes as the primary descent system is feasible even for the minimum models of the Martian atmosphere.

2) Augmentation of the gliding capability of the parachute system can be accomplished by application of thrust of proper magnitude and direction. Through such an augmented system horizontal flight can be achieved.

3) Examination of the optimum systems for the various propulsive requirements that exist during the landing operation, indicates that separate propulsion systems will be most efficient.

4) Pilot control of the thrust-augmented gliding parachute landing system is feasible.

During the next four months of this study results presented herein will be expanded to express all portions of the landing system in a form most useful for long range mission analysis. In addition the input parameters are being re-evaluated and modified to portray the same detail for Earth Landing Systems. However, preliminary results indicate manned landing vehicles can have a descent and landing system flexible enough to overcome unforeseen local conditions and accomplish successful landings on the planet Mars.

26985

PART 8

AMES RESEARCH CENTER MARS MISSION STUDIES SUMMARY

by

H. Hornby
Ames Research Center

AMES RESEARCH CENTER MARS MISSION STUDIES

SUMMARY

By

H. Hornby, Ames Research Center

We have covered a lot of territory both in terms of technological milestones and the old fashioned kind. With scant respect for detail, I will endeavor to summarize the highlights, injecting a few thoughts from an Ames in-house study (Ref. 1) at the same time.

An objective in all of these Ames studies is to expose technical problem areas. I will start with a recapitulation of several that have been encountered: radiation shielding, Earth and Mars atmosphere entry, and Mars descent and landing. Let us attend to the general problem of radiation shielding. There are several facets to be considered: cumulative effects, acute dosage, and the integration of the shielding material into the life support requirements. Among these, the cumulative effects of the galactic secondary radiation dosage to which the men have been exposed seems to be an item over which we have least control, but in any case it is of apparently small consequence. The first figure taken from NAA's results suggests that some 25 rads per year will be accumulated from such cosmic Bremstrahlung irrespective of the precise details of the packaging. Acute or chronic effects of a high onset rate of radiation dosage that may arise because of encounters with solar flares are perhaps of more significance. A Bailey model solar proton event is a large flare intentionally set larger than any observed to date. As indicated in the top left hand corner of the figure, the maximum onset rate for this event varies from many rads per hour at a few grams per square centimeter wall thickness to about 2 rads per hour at 10 gm/cm^2 thickness and to a small fraction of a rad/hour at some 20 gm/cm^2. It may be necessary to keep the minimum wall thickness at or above some 20 gm/cm^2 to minimize any hazard from this source. Research is needed to determine the allowable thresholds that should be used since the amount of shielding required is a most significant portion of the total vehicle mass and the total dosage is quite low for shielding much less than 20 gm/cm^2.

However, if the desired radiation shield thickness is indeed on the order of 20 gm/cm^2, there may be a substantial reduction in mission complexity and system development time and cost by using an open ecology. For a sample case, indicated on the second figure, an Ames study has shown little difference in Earth orbital launch weight between an open cycle and a partially closed life support system with oxygen and water recovery. Both open cycle water and oxygen may serve for radiation protection, in a single container if the oxygen is contained as hydrogen peroxide . . . The figure has been computed for a Mars orbiter mission using nuclear propulsion and for the opposition of 1975 but the results are typical of the entire range of missions (Ref. 1). There is less than a ten percent difference in Earth orbital launch weight between the open cycle and the partially closed life support methods even for round trip times as high as 460 days; longer trips, of course, tend to favor the partially closed system. Thus, we must decide carefully the point at which we close or partly close the life system. Do we include in the ecological system only the local environment within the independently designed cabin or do we include the total environment taking radiation, meteoroids, weightlessness and habitability into account? To reject, because the requirements are presently ill-defined, any ecological system closure that includes part of these latter environmental factors certainly appears to improve engineering definition and allows advantage to be taken should these factors later seem unimportant. There is however some justification in early mission planning in selecting an approach that errs slightly on the heavy side. An open-cycle water and oxygen system is also much simpler and its implication in terms of systems reliability is obvious.

The return to Earth is accomplished at speeds substantially greater than lunar return speeds and therefore poses radiative heating and guidance problems if atmosphere braking is used. For these and other reasons, propulsive braking to escape speed has been considered by various investigators. Propulsive capture at Mars has also been considered, primarily because of ignorance of the Mars atmosphere characteristics at this time. Figure 3 indicates the advantages that may accrue from atmosphere braking. On the left of the figure, the total propulsive ΔV that must be delivered is shown for optimized 10-day stopover missions as a function of the opposition date for an all propulsive system with propulsive braking at the target planets and for a system in which atmosphere entry techniques are used to

decelerate at Mars and Earth. The ΔV requirements for the propulsive braking case are seen to increase with opposition date from a typical "good" year, 1971, to a typical "bad" year, 1980; whereas, the corresponding requirements for atmosphere braking are relatively insensitive to opposition with a maximum value in an intermediate year, 1975. On the right of the figure, these ΔV results are exponentially amplified by introduction of the various mass ratios involved to convert to Earth orbital launch weight. Two important advantages to atmosphere braking are now emphatically clear. First, overall weights are reduced some four to ten fold because the effective specific impulse of a heat shield is an order of magnitude better than that of a rocket. Second, the influence of opposition is masked. Thus the "bad" years no longer produce large amplifications of the Earth orbital launch weights so that now the launch window is opened up to include every 26-month cycle instead of the few "good" oppositions separated by 11 or 12 year "dead bands." Both for the lower atmosphere entry plot in which nuclear propulsion is used for Earth orbital departure and for the higher all-chemical propulsion curve, only a 30 to 40 percent variation in weight requirements is indicated between the minimum and maximum values. The propulsive braking results, on the other hand, show a three-to-one increase in launch requirements from a "good" year to a "bad" year. This picture is somewhat of an oversimplification. If we examine the figure, we see that only the variation of total propulsive ΔV with opposition is shown for the atmosphere braking case. A plot of Earth entry speeds would show a steady increase from good to bad years from some 44,000 fps in 1971 to over 70,000 fps in 1978 or 1980. Certain heat shield studies indicate reasonable weight penalties over this entire range of entry speeds. The guidance task, however, becomes entremely problematic at the high end of the speed range. The use of a Venus flyby during the return leg to reduce Earth entry speeds is therefore of significant interest. The dramatic result obtained by STL for the opposition of 1975 is shown in Figure 4; with most other mission parameters essentially unchanged, the passage by Venus has reduced Earth entry speed almost 23,000 fps from the optimum direct value of some 68,000 fps to a value quite typical of a "good" year, about 45,000 fps. There can be no doubt that exploitation of Venus merits further study in depth. So far, the technique has been examined for the purpose of reducing Earth entry speed primarily on the return leg. This is but a fraction of the potential of the method. It can be used on the outbound leg. In addition, by a reoptimization of the trajectories, it might be used either on the outbound or the return leg to reduce propulsive ΔV

requirements which would be reflected in a sharply decreased Earth orbital departure weight. In this sense, reductions in Earth entry speed operate against a mechanical disadvantage because the Earth entry vehicle is in general only a small fraction of the total return weight. Nevertheless, the results obtained to date indicating availability at all oppositions to maintain Earth entry speeds at or below 50,000 fps are important indeed.

Perhaps the most critical parts of the Mars landing and return mission are the actual events of descent to and touchdown on the unknown and hostile Mars Surface. It appears from Northrop studies that it is possible to combine a parachute system incorporating glide capability with a rocket propulsion system to provide a controlled landing with lateral translation and hover capability. Figure 5 shows the Mars parachute system weight ratio as a function of the terminal descent velocity. For either the Schilling nominal model atmosphere or the NASA minimum, the parachute system weight need not exceed five percent of the landing vehicle weight for terminal speeds less than 100 fps. Such terminal speeds must be taken out by a rocket system; moreover, in view of the large thrust-to-weight requirements and small ΔV, the optimum system is possibly obtained with a solid rocket. When the parachute system weight is coupled to the weights of the various rocket systems, glide augmentation, terminal and hover, the overall landing system weight ratio is increased to some ten percent of landing vehicle weight, still quite a small value. The Mars landing system is so critical to the success of the mission that it probably needs to be heavily redundant. A vital ingredient therefore is a basic weight that is low.

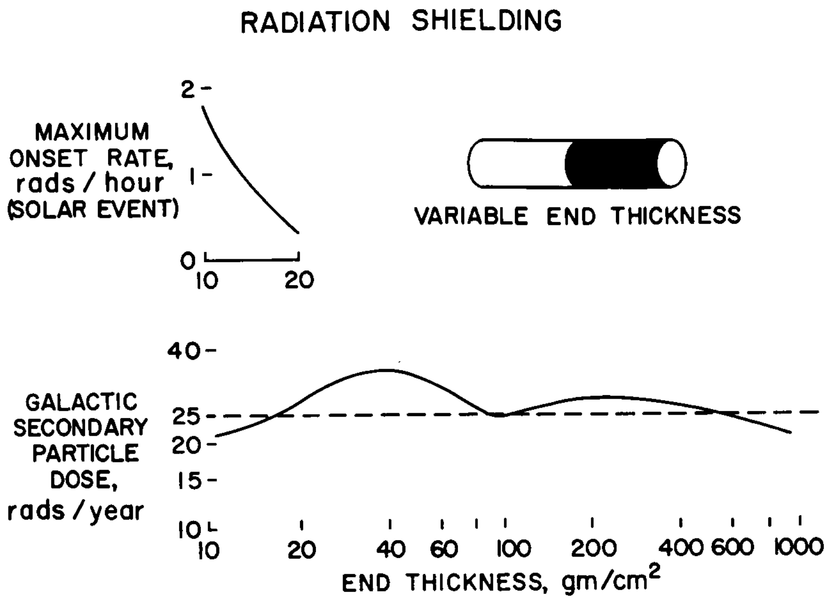
Figure 6, based on some NAA results, is important in that it shows much reduced systems sensitivity in areas where technological uncertainty is greatest, the Mars excursion module and the Earth entry module. The sensitivity characteristics shown are for the opposition of 1975 and for various other mission constraints, 480 days total trip time, 30 days stopover at Mars, chemical propulsion and so on. Undoubtedly, other conditions will substantially revise these values but the indicated trends should remain the same. Hence, changes in the Mars excursion module weight will be diminished by an order of magnitude in terms of the percentage change they cause in Earth orbital departure weight. Similarly, changes in the Earth entry module weight will be reduced three to five fold in their influence on overall system weight. These features, of course, are

most useful for mission planning. It appears that we can afford to take greater technological risks in these areas than we can, say, in the Mars departure propulsion where, as is indicated in the figure, even the smallest departure from nominal values, ΔV, specific impulse or inerta, is reflected and amplified throughout the system.

In conclusion, it may be stated that the studies have shed considerable light on the manned Mars mission. There remain many uncertainties. In particular, it is unclear that we have fully explored all potential energy saving modes. The tacit assumption that Mars orbit rendezvous is the best method needs careful examination; it restricts us to landings close to the Mars equator and is prone to precession problems similar to those encountered in Earth orbital assembly and departure. The influence of opposition date is perhaps much less of a concern now than when the studies commenced and, if we exploit fully atmosphere braking and the Venus flyby, the differences between "good" and "bad" years may be slight.

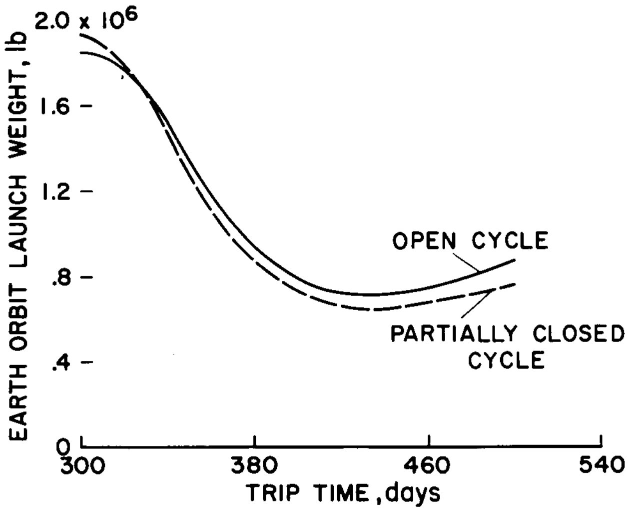
REFERENCE

1. Wong, Thomas J. and Anderson, Joseph L.: "A Preliminary Study of Mission and Spacecraft Requirements for Manned Mars Orbiting and Landing," NASA TM X-54032, 1964.



A-31875-1

Figure 1.- Radiation shielding.



A-31874-19

Figure 2.- Effect of life support system closure on launch weight: orbiter mission; 8 man crew; nuclear propulsion; 1975 opposition.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER, MOFFETT FIELD, CALIFORNIA

A-31875-3

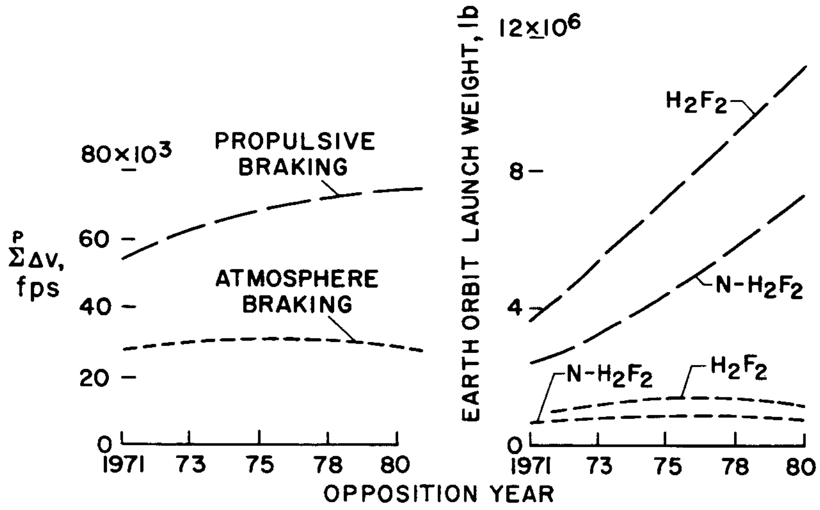


Figure 3.- Planetary deceleration.

A-31875-4

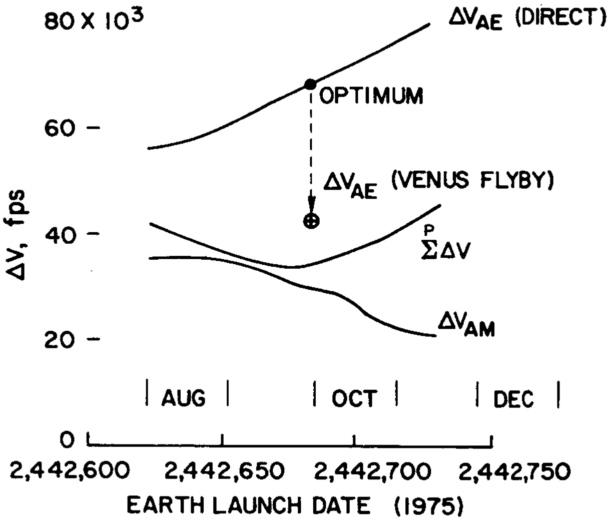


Figure 4.- Venus flyby return.

A-31875-5

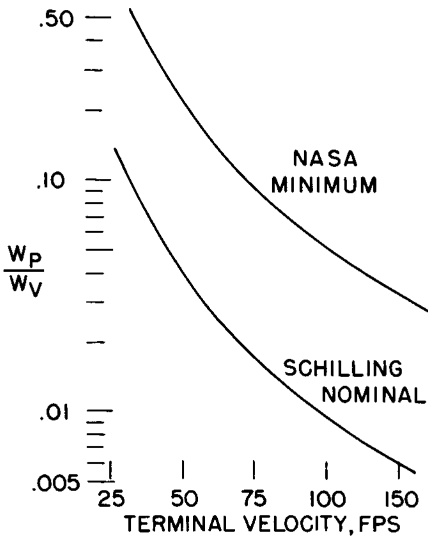


Figure 5.- Mars parachute weight ratio.

A-31875-6

PARAMETER	% CHANGE	% WT_{EO}
TRANS-MARS INJECTION ΔV	10	11.7
MARS RETROBRAKING ΔV	10	8.4
TRANS-EARTH INJECTION ΔV	10	22.4
EARTH RETURN MODULE WT	10	3.3
MISSION MODULE WT	10	6.0
MARS EXCURSION MODULE WT	10	0.6

*16-MONTH MISSION, 30-DAY MARS STAY TIME, 30-DAY EARTH LAUNCH WINDOW, OF_2/MMH INTERPLANETARY PROPULSION WITH F_2/H_2 TRANS-MARS PROPULSION

Figure 6.- System sensitivity characteristics: 1975 mission*.

26986

PART 9

A STUDY OF MANNED MARS EXPLORATION IN THE UNFAVORABLE TIME PERIOD (1975-1985)

by

Dr. R. N. Austin
General Dynamics/Ft Worth
Contract No. NAS8-11004

F O R E W O R D

This document contains a summary of the results of a study on Manned Mars Exploration in the Unfavorable Time Period (1975-1985). The analysis was performed by General Dynamics/Fort Worth for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under Contract No. NAS8-11004. The study was established by the Future Projects Office of NASA-MSFC as part of the long-range and continuing planning of the manned exploration of space.

The complete results of the study are presented in five volumes:

Volume I - Condensed Summary

Volume II - Summary

Volume III - Technical Report

Volume IV - Appendix

Volume V - Spacecraft Propulsion Systems (Title Unclassified; Document CONFIDENTIAL-RESTRICTED DATA)

1.0 INTRODUCTION

This document is a summary of a study on Manned Mars Exploration in the Unfavorable Time Period (1975-1985). The analysis was performed by General Dynamics/Fort Worth (GD/FW) for the George C. Marshall Space Flight Center. The basic objective set for the study has been the survey and definition of mission profiles for missions during 1975-1985 and the recommendation of attractive mission concepts based on selected profiles.

The most important factors in this study result from the unfavorable characteristics of the time period, i.e., relatively high energy requirements because of the eccentricity of Mars' orbit and an expected increase in solar activity which must be reflected in additional shielding to protect the crew from radiation. However, major consideration was required for weighting other criteria. For example, minimum initial mass in Earth orbit (IMIEO) is an appropriate criterion for the selection and recommendation of mission and vehicle. Nevertheless, at the inception of the study GD/FW held the opinion that, other criteria should also be considered in a realistic recommendation of missions and vehicles, i.e., (1) probability of mission and program success, (2) program costs, and (3) the technological risk associated with each concept. The outline shown in Figure 1.0-1 has been followed throughout the program to implement this approach. Analysis of technological risk, an unusual feature of the study, provides a measure of the confidence in meeting the initial performance, reliability, and schedule for a particular program.

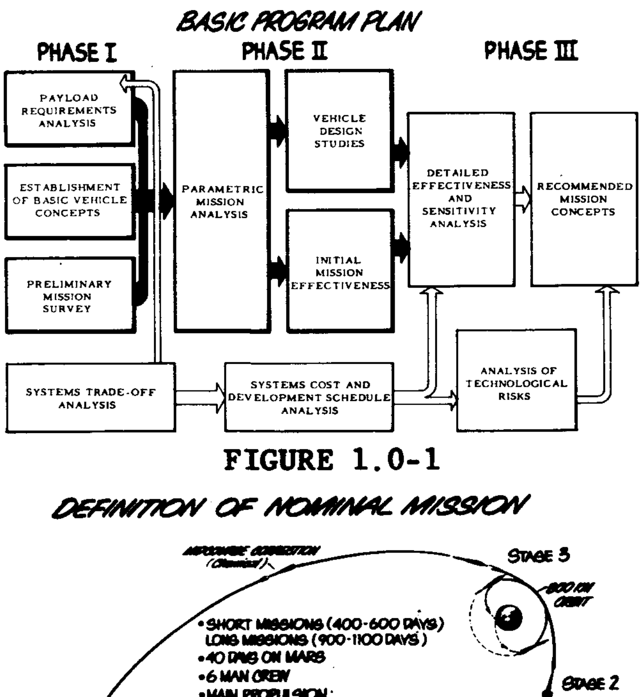


FIGURE 1.0-1

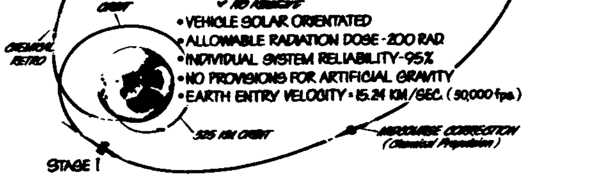


FIGURE 1.0-2

To limit the scope of the study, the parametric analysis was concentrated on the Nominal Mission defined in Figure 1.0-2. Additional vehicle concepts and mission modes were investigated only for selected points on the basis of the results obtained for the Nominal Mission. In the basic program plan, emphasis has been placed on two predominant though not independent areas of investigation: (1) the Mission and Vehicle studies and (2) the Operations Research studies, and the summary material in this document is presented in this order.

MISSION AND VEHICLE STUDIES

2.0 MISSION TRAJECTORY ANALYSIS

2.1 TRAJECTORY COMPUTATIONS

In trajectory computations, the "patched conic" technique was used to simulate the overall mission profile. Perturbations of the transfer trajectory were neglected, except for the gravitational influence of the launch and target planets.

Planetary position and velocity coordinates were approximated by means of the 2-body equations in conjunction with mean orbit elements. In order to account for secular perturbations of the planets, their mean orbit elements were expressed as linear functions of time. Lambert's theorem was then applied to calculate the elements of the heliocentric transfer conic which passed through the centers of the launch and target planets. Departure and arrival hyperbolic excess velocities were obtained by taking the vector difference between the spacecraft and planet heliocentric velocities.

In order to check the accuracy of the planetary ephemeris approximation for the period from 1975 to 1986, heliocentric position coordinates of Earth and Mars were calculated by means of the approximate equations at intervals between 1963 and 1980, and checked against the coordinates in published ephemerides. The comparison indicated maximum radial and angular errors of approximately 2.0×10^{-4} A.U. and 0.02°, respectively, within this time period.

2.2 NOMINAL MISSION PROFILE

A nominal round-trip trajectory profile was defined at the beginning of the study for the purpose of surveying the mission requirements for the 1975-1986 time period. The nominal profile was characterized by (1) planar ballistic heliocentric transfer orbits, (2) circular parking orbits of 325 and 800 km altitude, respectively, at Earth and Mars, and (3) direct atmospheric entry at Earth return. A maximum allowable Earth entry velocity of 15.24 km/sec at 121.92 km altitude (50,000 fps at 400,000 feet altitude) was specified. If the unbraked entry speed exceeded this limit, a propulsive ΔV was computed for the Earth capture maneuver. All departure and capture maneuvers were assumed to be planar.

2.3 MISSION MAPS AND MISSION CLASSIFICATION

An extensive survey of Earth-Mars-Earth missions, based on the ground rules described in subsection 2.2, was made in the early phases of the study. The primary purpose of the survey was to determine initial trial trajectories for use in analytical programs for mission optimization. An IBM 7090 computer program was used to map impulsive, one-way total ΔV requirements, along with other pertinent trajectory data, at 10-day intervals on an Earth-date versus Mars-date grid.

Figure 2.3-1 is a typical ΔV contour chart drawn from the mission maps.

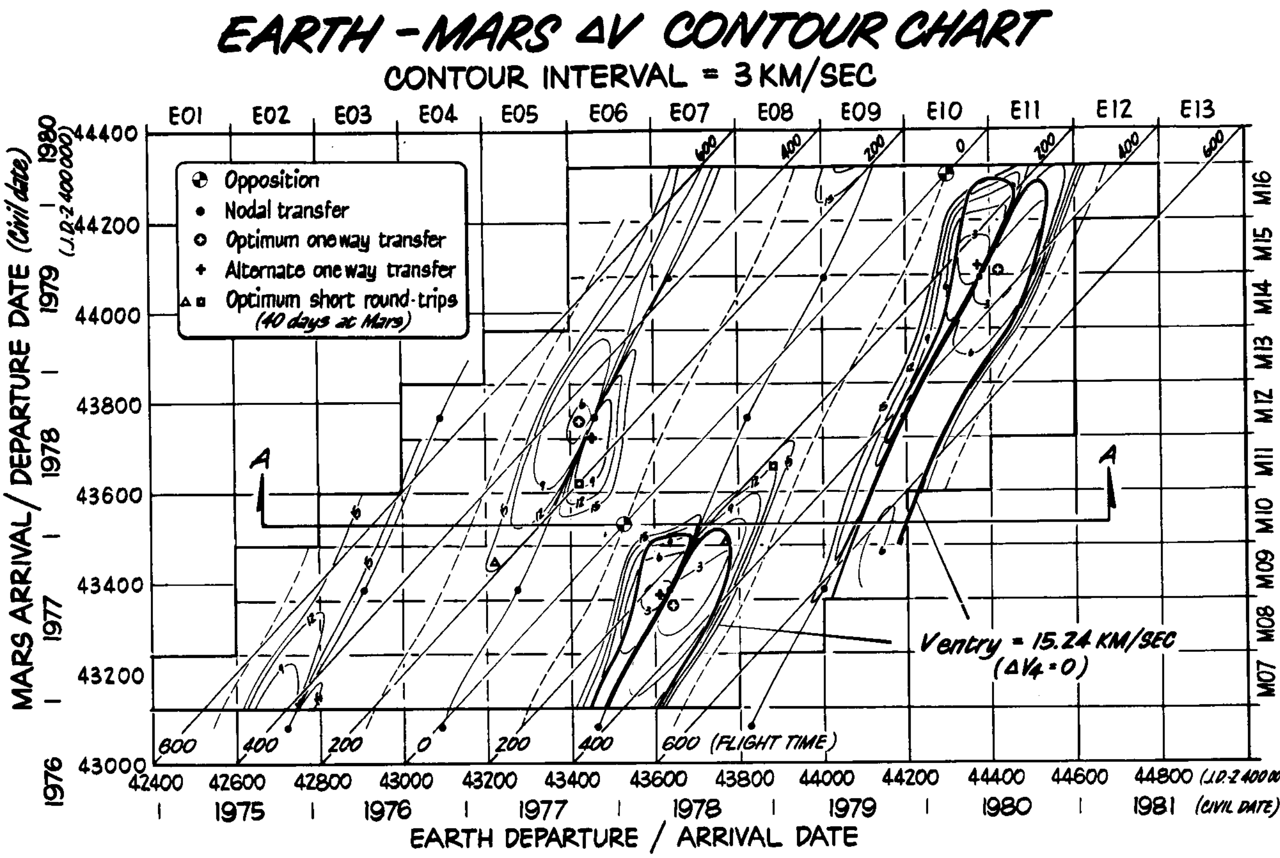


FIGURE 2.3-1

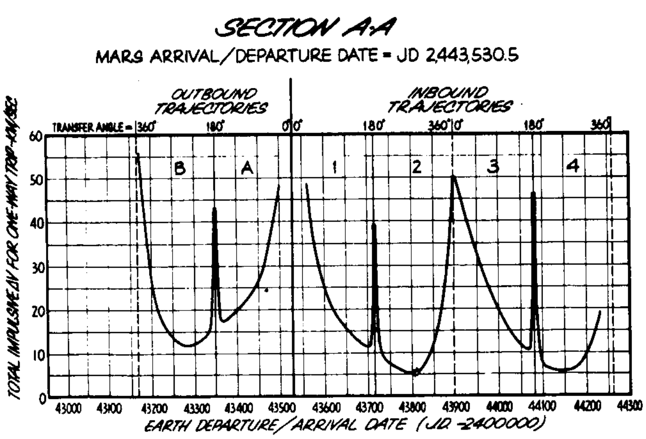


FIGURE 2.3-2

The inclination of Mars' orbit causes the ΔV "surface" to be characterized by a series of roughly parallel valleys, separated by ridges which correspond to heliocentric transfer angles of 180° or 0° (360°).

Each point on the Earth-date versus Mars-date grid represents a unique one-way transfer. A complete round-trip mission is defined by 2 points, one on each side of the equal-date line. For missions launched near a given opposition of Mars, all of the practical outbound trajectories are found to lie in the first 2 valleys (designated A and B) to the left of the opposition point on the mission map. All practical inbound trajectories are found to lie in the first 4 valleys (designated Type 1, 2, 3, & 4) to the right of the

opposition point. Any round-trip mission may be classified by a combination of the alphabetic and numeric characters which define the ΔV valleys in which the one-way transfer points lie. Thus designated, missions of a given classification are found to have similar characteristics for all opposition periods.

Of the 8 practical types of missions, two classes were selected for detailed study and optimization. The first of these, broadly categorized as "minimum-time class," consists of missions utilizing Type-2 return trajectories. Optimum stay-time for this class of missions is zero; therefore, the suffix S40 has been added to the A2 and B2 classifications to indicate that a 40-day stay-time in Mars orbit was applied as a mission constraint. The second class of missions is referred to as the "minimum-ΔV class" and consists of missions utilizing a Type-3 or Type-4 return trajectory. In this report, missions belonging to the latter class are often referred to simply as "long missions." The A2 and B2 types are commonly called "short-short" and "long-short" missions, respectively, and "short missions," collectively.

2.4 ΔV-OPTIMIZED MISSIONS

Total propulsive ΔV is a significant parameter for mission evaluation. Besides being relatively easy to compute, it is highly indicative of required total vehicle mass, and yet independent of vehicle design parameters. Consequently, this parameter was used as a criterion for preliminary selection. In general, the ΔV-optimized mission of a given type is very nearly optimum from the standpoint of IMIEO. (Significant exceptions to this generalization are discussed in Section 7.)

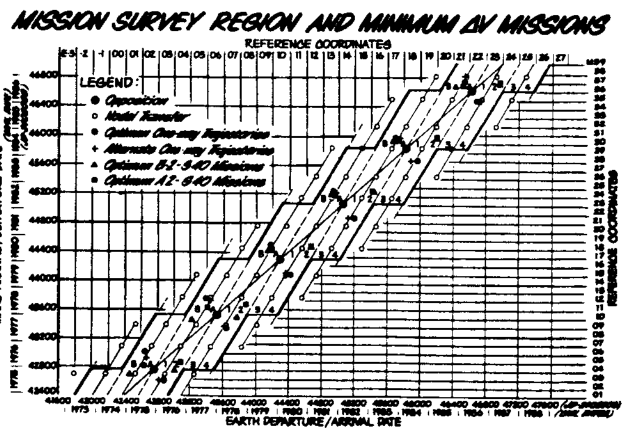


FIGURE 2.4-1

For each opposition period between 1975 and 1986, minimum ΔV as a function of launch date was determined by an IBM 7090 analytical optimization program for the 2 classes of missions defined in subsection 2.3. The sum of impulsive velocity increments for the 4 primary mission maneuvers was the optimization criterion for this phase of the study. The ΔV-optimized missions were determined in the computer program by iteration on the following set of independent variables:

1. Launch date
2. Outbound transfer time
3. Mars stay-time
4. Mission duration.

Provisions are made in the computer program for equality constraints on one or more of the independent variables. For instance, a constraint on stay-time was necessary for the minimum-time class missions (A2-S40 and B2-S40) because optimum stay-time for this class of mission is zero.

The effect of Mars' orbit eccentricity on mission ΔV requirements is readily apparent in Figure 2.4-2. At opposition in 1980, Mars is near aphelion; in 1988, it is near perihelion. Optimum launch-date missions have been numbered in this figure for easy reference in following sections of this report. Typical heliocentric geometry of the 2 mission classes is shown in Figure 2.4-3

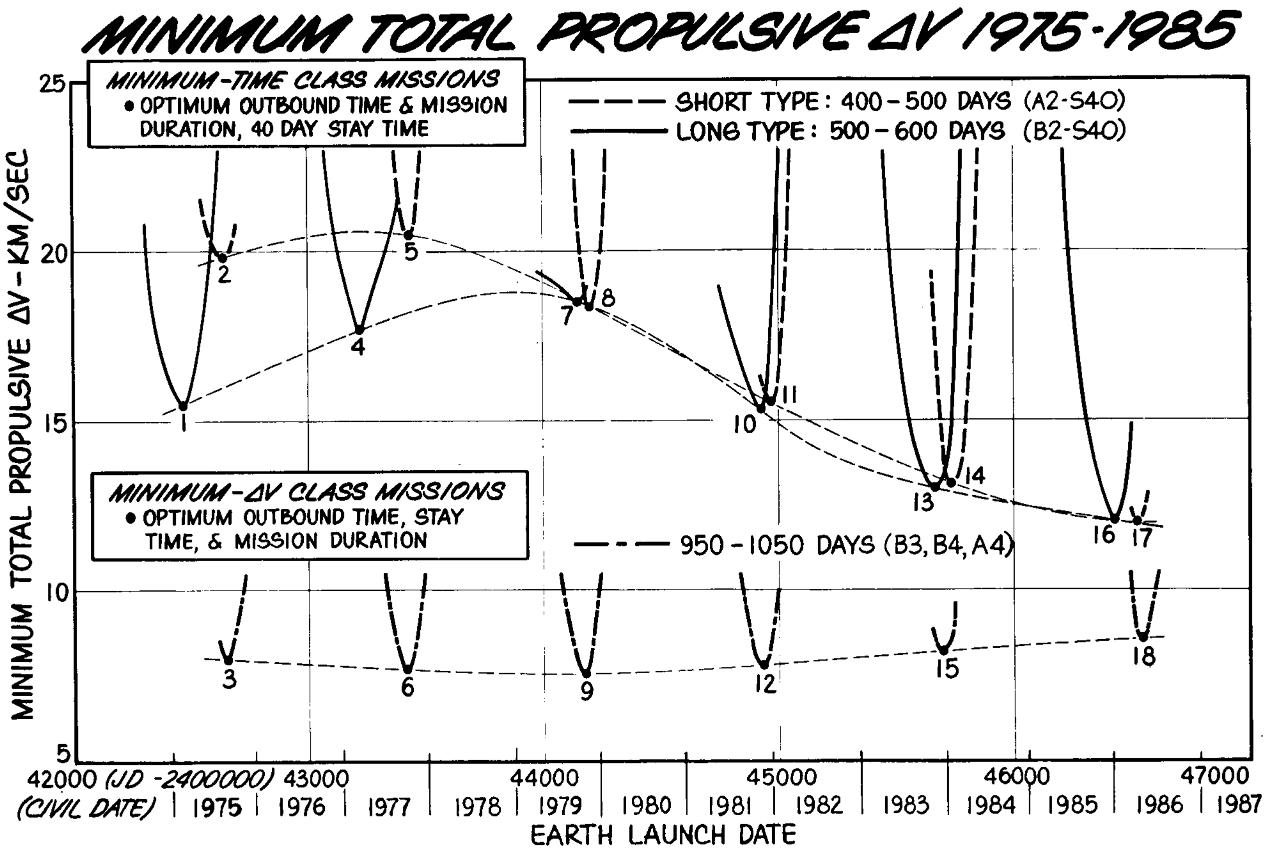
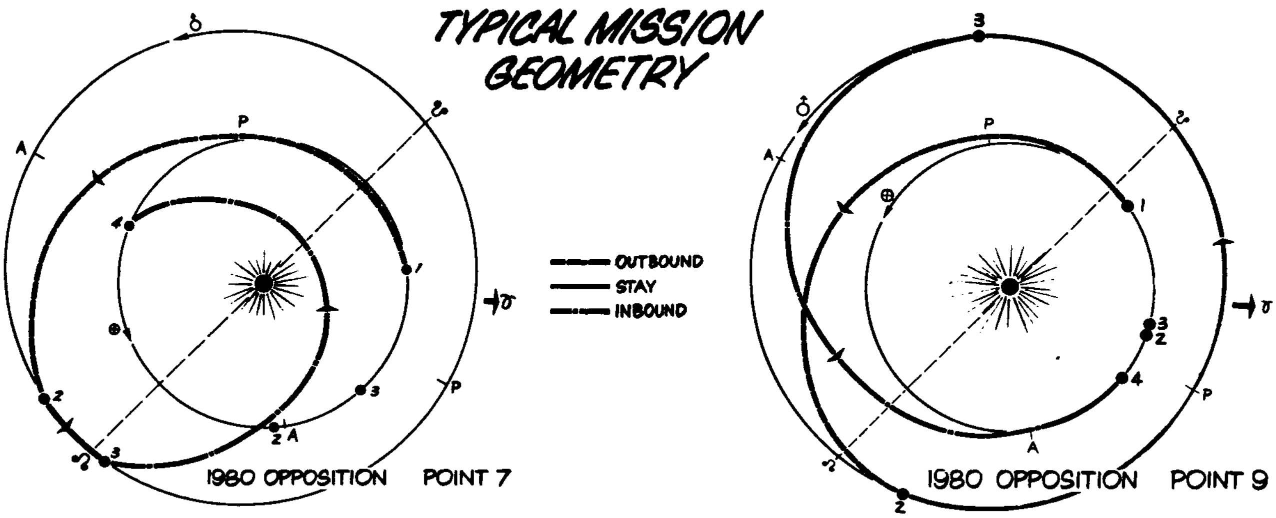


FIGURE 2.4-2



1980 Opposition Point 7

LOCATION	1	2	3	4	TOTAL
CIVIL DATE	OCT 4, 1979	JULY 1, 1980	AUG 10, 1980	MAR 3, 1981	
TIME - DAYS	271	40	208		517
ΔV - KM/SEC	4.05	2.91	6.03	5.51	18.51
PERIHELION DISTANCE - A.U.	0.97		0.44		
COMMUNICATION DISTANCE - A.U.		1.50	1.76		
ASYMPTOTE DECLINATION	2.84	-8.58	-11.23	12.83	

1 EARTH DEPARTURE
2 MARS ARRIVAL
3 MARS DEPARTURE
4 EARTH ARRIVAL

1980 Opposition Point 9

LOCATION	1	2	3	4	TOTAL
CIVIL DATE	NOV. 4, 1979	SEPT 9, 1980	SEPT 14, 1981	AUG 20, 1982	
TIME - DAYS	310	370	340		1020
ΔV - KM/SEC	3.60	2.04	1.95	0	7.59
PERIHELION DISTANCE - A.U.	0.99		0.99		
COMMUNICATION DISTANCE - A.U.		1.92	2.17		
ASYMPTOTE DECLINATION-DEG.	29.54	-17.60	0.22	17.71	

FIGURE 2.4-3

2.5 EFFECT OF MISSION CONSTRAINTS

The effects of certain mission constraints were investigated in selected opposition periods. In addition to the constraint of the independent variables (stay-time and mission duration), a limited study of the effect of a perihelion distance constraint was included.

The results of the perihelion constraint study are shown in Figure 2.5-3. The 1980 opposition was chosen for investigation in this study because the smallest unconstrained values of perihelion distance (approximately 0.43 A.U.) were obtained for optimum missions during this period. The application of an identical perihelion constraint (0.5 A.U.) in other opposition periods would have a lesser effect, or no effect at all, on total propulsive ΔV.

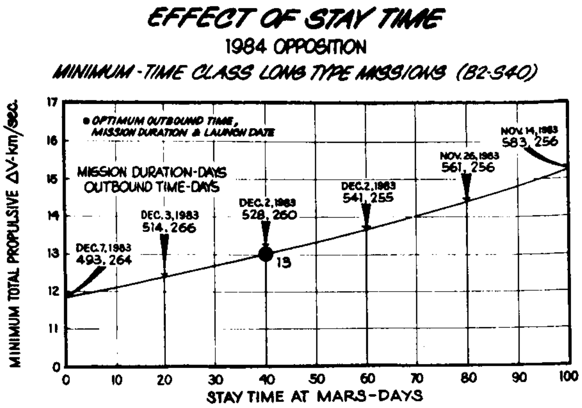


FIGURE 2.5-1

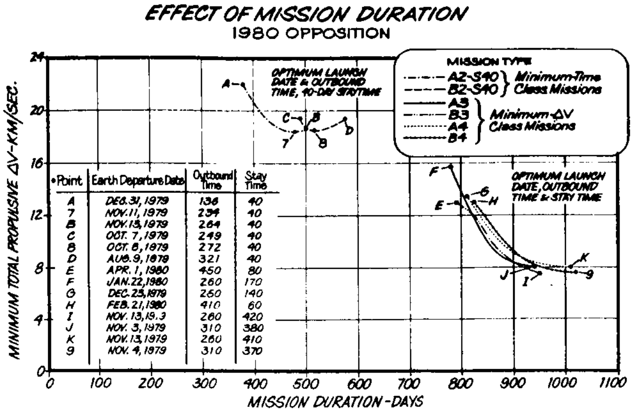


FIGURE 2.5-2

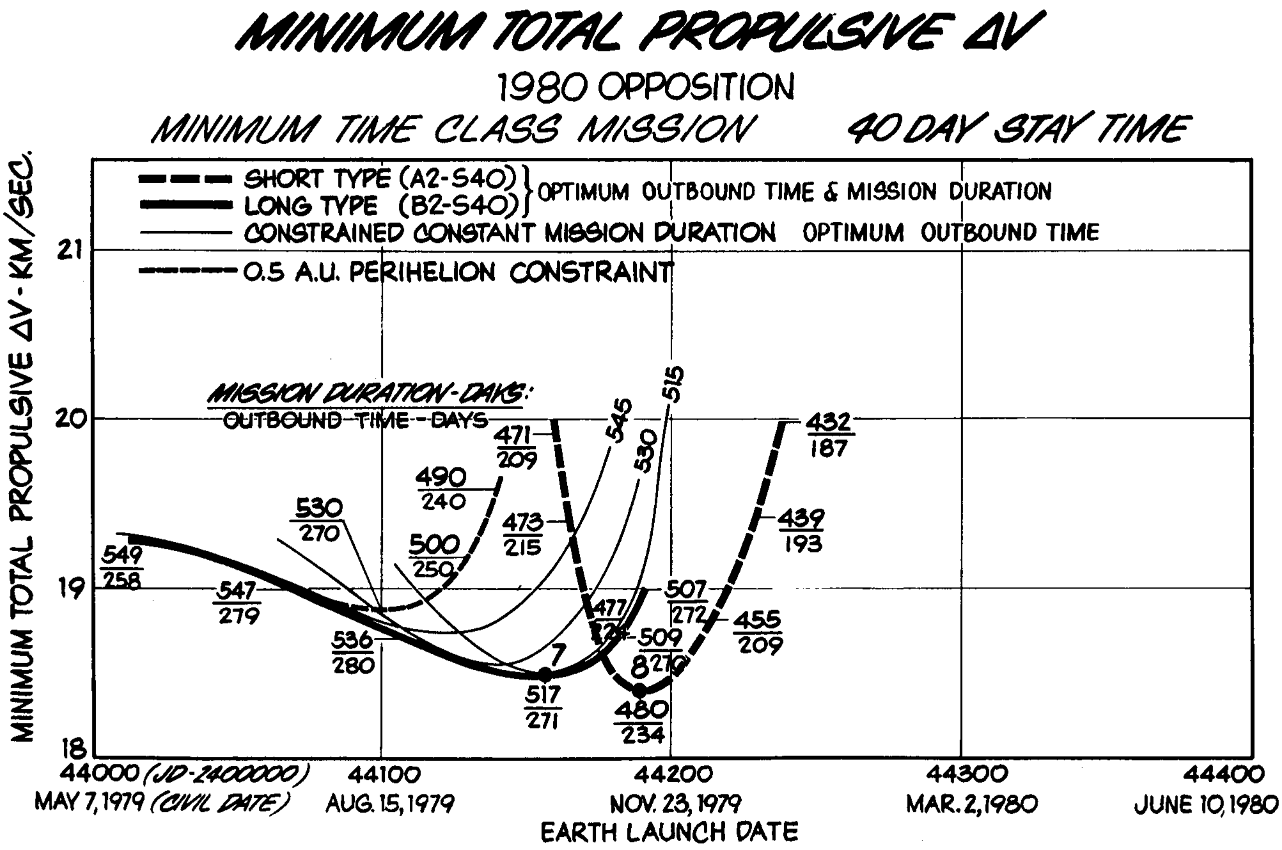


FIGURE 2.5-3

2.6 HELIOCENTRIC PLANE CHANGE

A limited study was made of the desirability of using a heliocentric plane-change maneuver to decrease ΔV requirements for one-way trajectories having transfer angles near 180°. The analytical approximation developed by Fimple was used to optimize the location of the mid-course maneuver.

The results of this study indicate that use of the plane-change maneuver does not yield total mission ΔV's lower than those obtained with optimum planar trajectories. However, the plane-change maneuver does offer a means of improving launch-window quality for some types of missions. For instance, it may be desirable to plan missions so that the Mars arrival date remains constant if the launch is delayed. In such a case, it is possible to add as much as 20 days width in the center of the launch window if a modest mid-course ΔV capability is provided.

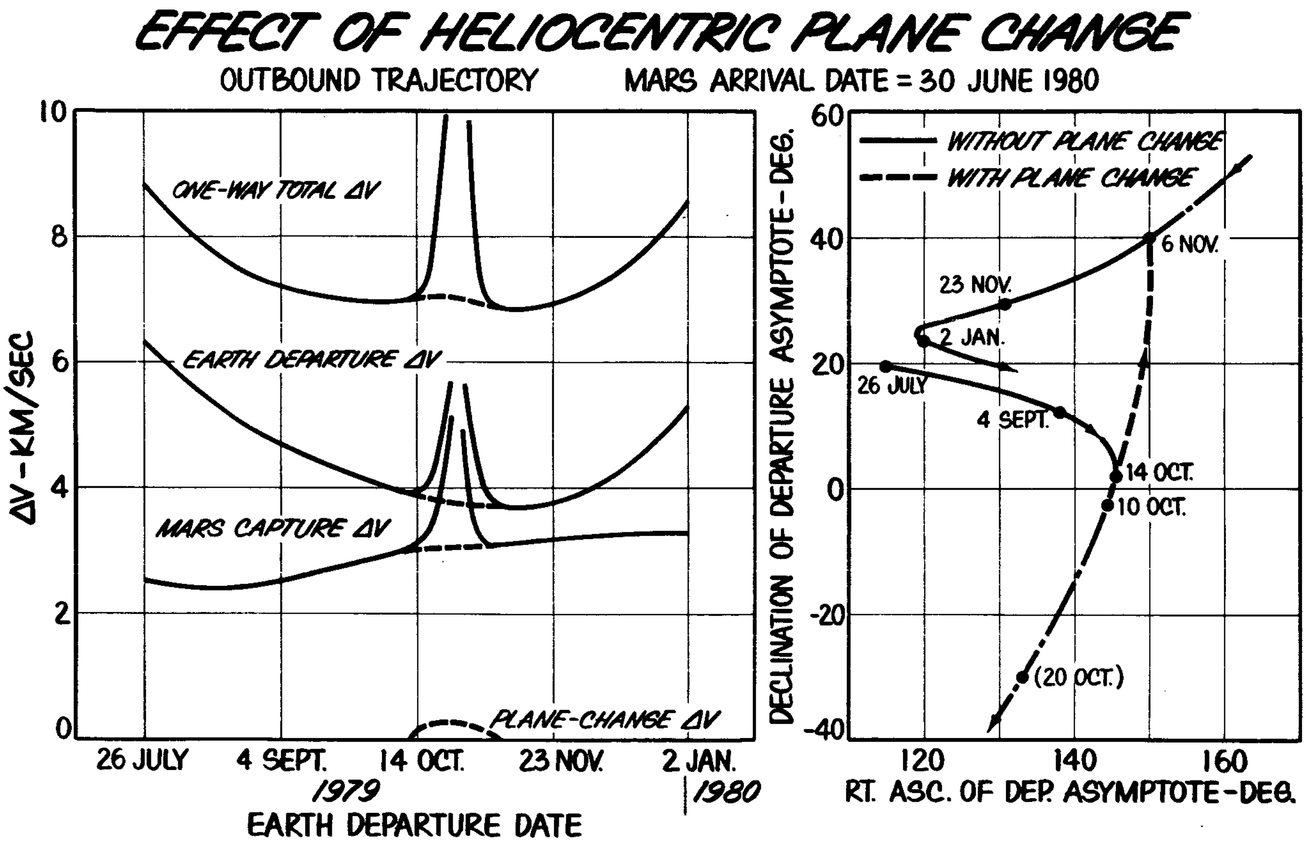


FIGURE 2.6-1

2.7 CALENDAR EFFECTS

The true anomaly of Mars at opposition exhibits a cyclical variation with a period of approximately 16 years. This fact, coupled with the relatively large eccentricity of Mars' orbit, causes a similar cycle for round-trip mission ΔV requirements (Figure 2.4-2). In those synodic periods during which Mars is near perihelion at opposition, minimum ΔV for short missions (Type-2 return trajectory) is comparatively low. When Mars is near aphelion at opposition, minimum ΔV for this class of missions is comparatively high. The high ΔV requirements for short missions in the 1978 and 1980 opposition periods, and the expected peak of solar activity in 1980 have caused the 1975 to 1985 time period to be characterized as "unfavorable."

It is well-known that ΔV requirements for long missions (Type-3 or -4 return trajectory) are significantly lower than those for short missions, and that the cyclical variation of ΔV requirements is smaller in magnitude and opposite to that of the short missions. The highest values of required mission ΔV for the long missions occur when Mars is near perihelion at opposition, and lowest values occur when Mars is near aphelion at opposition.

While the ΔV requirement is a significant parameter for mission evaluation, it is by no means adequate as a basis for final trajectory selection, especially when missions of different classes are compared. Since IMIEO is highly correlated with program cost, it is a much better parameter for use in overall mission evaluation. For this reason, the optimization criterion used in the final phases of this study was IMIEO The results of the IMIEO-oriented study are discussed in detail in Section 7. However, since an understanding of the IMIEO results is necessary for a meaningful discussion of trajectory characteristics, they are presented in brief below.

2.7.1 Mission Duration. On the basis of total propulsive ΔV, it is evident that the best method of alleviating calendar effects on Mars missions, insofar as heliocentric trajectory selection is concerned, is the utilization of missions of long duration. Although support for this conclusion is provided by data on mission ΔV requirements, the conclusion is by no means definitive. On long missions, additional vehicle mass will be required for radiation and meteoroid shielding, life support, propellant boil-off, systems redundancy, etc. The IMIEO-optimization study, in which these factors are taken into account, indicates that the mass requirements for long missions are appreciably lower than those for short missions in the unfavorable years. Moreover, another factor, which is possibly more significant from the standpoint of long-range planning, must be considered. It was verified on the IMIEO-optimization study that long mission mass requirements are relatively constant for all opposition periods. It is recognized, therefore, that there are serious problems as well as some advantages in long missions, and these cannot be measured in terms of either mass or cost. These considerations are discussed in Section 11.

A limited investigation of intermediate-length missions in the 1980 opposition period has been made. These missions result from constraining mission duration to shorter than optimum for missions having Type-3 or -4 return trajectories. Minimum practical mission duration for the Type-3 or -4 return trajectories is on the order of 800 days. The required ΔV for these missions is approximately 4 km/sec lower than that for the short missions (Figure 2.5-2), but IMIEO values for these ΔV-optimized, duration-constrained missions were higher than those for the short missions The higher IMIEO values resulted in part from mission-duration effects on system masses, and in part from unfavorable distributions of the velocity increments.

Since IMIEO-optimized trajectories for these constrained-duration missions were not obtained because of convergence difficulties in the computer program, it cannot be said categorically that they are worse than those for the short missions. On the basis of this limited investigation, however, the use of them does not seem to be advantageous.

2.7.2 Planetocentric Trajectories. A number of variations in the planetocentric segments of the overall mission profile are possible. Among those which have been proposed are (1) elliptical Mars-capture orbits, (2) atmospheric braking at Mars, (3) higher allowable Earth-entry velocities, and (4) hyperbolic Earth rendezvous.

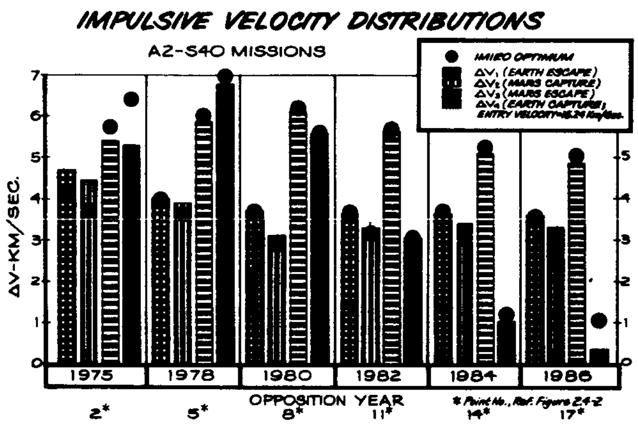


FIGURE 2.7-1

The distribution of velocity increments for typical ΔV-optimized short missions (Type A2-S40) is shown for the various opposition years in Figure 2.7-1. It is apparent that the cyclical variation of the first 3 velocity increments (Earth departure, Mars capture, and Mars departure) is small in comparison to the fourth increment (Earth capture). Further, it is indicated that the effectiveness of elliptical parking orbits and atmospheric braking at Mars is not strongly dependent on opposition year. These methods represent a means for potential improvement of any mission; they cannot be considered, however, as techniques for use in alleviating the calendar-year effect.

There is a wide variation with opposition year in the Earth-capture velocity increment. This increment is highly correlated with mission total ΔV and with IMIEO. Thus, it is indicated that the most profitable area of study for the alleviation of the calendar year effect for short missions is in Earth-capture techniques. Because Earth atmospheric braking is desirable, a somewhat optimistic entry velocity limit of 15.24 km/sec (50,000 fps) was assumed for purposes of defining the Nominal mission concept in this study. In addition to investigating the desirability of using complete atmospheric braking at Earth arrival, the hyperbolic Earth-rendezvous technique also warrants additional study. In the use of this technique, a spacecraft is met by a pickup vehicle as it approaches Earth. Although no investigation has been made during the present study, data has been generated which indicates that future studies of this technique would be worthwhile.

3.0 PAYLOADS REQUIREMENTS ANALYSIS

3.1 GENERAL

The data which will be described and illustrated in the following subsections have been generated for two reasons: (1) to establish, as parametrically as possible, data which will reflect projected state-of-the-art increases in systems performance, and decreases in system mass; and (2) to provide data which can be curve-fitted with analytical expressions to be used in an iterative computer program to determine the vehicle mass buildup. Since the variation in initial mass in Earth orbit (IMIEO) with payload mass will be investigated as part of the Operations Research studies, the data generated during these payload studies were considered as baseline numbers only. It should be pointed out that the systems data presented in this section which reflects calendar-year variations in performance are for a nominal reliability level of 95%.

3.2 STRUCTURES SYSTEM

3.2.1 General. In the structures systems study, methods were established for use in predicting the structural masses of the various components of the composite spacecraft. The methods were preliminary ones, by necessity, and provisions were made for flexibility in their use in the parametric mission analysis.

3.2.2 Cryogenic Tank Structural Weight Analysis. Equations were provided to generate the structural weights required for non-meteoroid- and meteoroid-protected tanks fabricated from both aluminum and titanium. The equation for non-meteoroid-protected tanks was based on differential gas and liquid head pressure requirements; contingency factors were applied as needed to cover other requirements, such as those for slosh baffles, hatch seals, reinforcements, fittings, bending, axial loads, etc. The equations for meteoroid-protected tanks were based on the non-meteoroid-protected equation with the meteoroid protection requirements incorporated. The method of meteoroid protection used in these equations was based on double-wall theory; in the use of this technique, the outer wall or skin acts as a bumper to shatter the majority of the meteoroids encountered. Actual penetration of the inner wall was based on a 99% probability that there will be no more than one penetration in a 30-day period. Shown in Figure 3.2-1 are typical plots of Tank Fractions vs Tank Volumes, with and without meteoroid protection, for titanium and aluminum. The curves do not contain contingency factors.

3.2.3 Mars Mission Module Structural Mass. A unit structural mass of 22.0 kg/m^2 (4.5 lb/sq ft) of module surface area was used to obtain a generalized structural mass for the mission module, including its meteoroid protection.

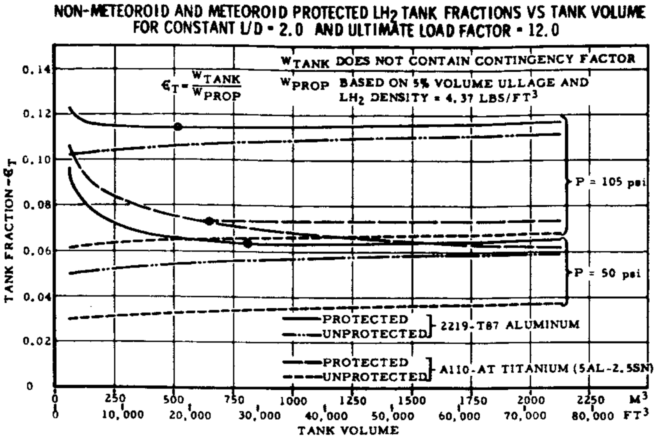


FIGURE 3.2-1

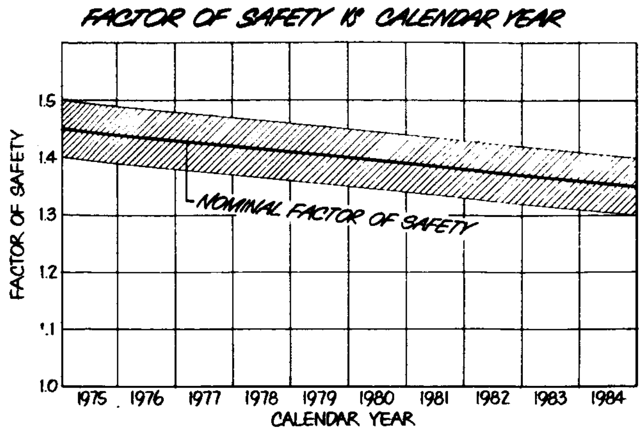


FIGURE 3.2-2

3.2.4 Structural Support Masses. The structural weight required to support the individual components in the composite spacecraft was based on 1½% of the individual component gross weight. This value was restricted to missions where the spacecraft's accelerations or decelerations do not exceed 2.0 Earth G (ultimate) from Earth-orbit escape to just prior to Earth reentry. Structural support mass for both chemical and nuclear engines was based on 5% of the engine weight and accessories supported.

3.2.5 Nominal Factor of Safety. To incorporate a state-of-the-art decrease in structural weight, a decrease in the nominal factor of safety with calendar year, as shown in Figure 3.2-2, has been postulated. Assumptions were made regarding improvements in (1) material dimension tolerances, (2) accuracy in design loads prediction, and (3) component and vehicle testing methods.

3.3 RADIATION SHIELDING ANALYSIS

3.3.1 General. The radiation environment to which the crew members will be exposed while on Mars missions consists of space radiation and man-induced or on-board radiation. Shielding weights for the protection of the crew were calculated parametrically for 3 total mission doses: 100, 150, and 200 rads. A quasi-optimum partition of each total dose was made for cosmic rays, solar flares, main propulsion reactors, and the nuclear auxiliary propulsion unit. Although a specific dose was allocated for cosmic radiation, no provision was made to shield against this radiation because of the higher energy secondary events.

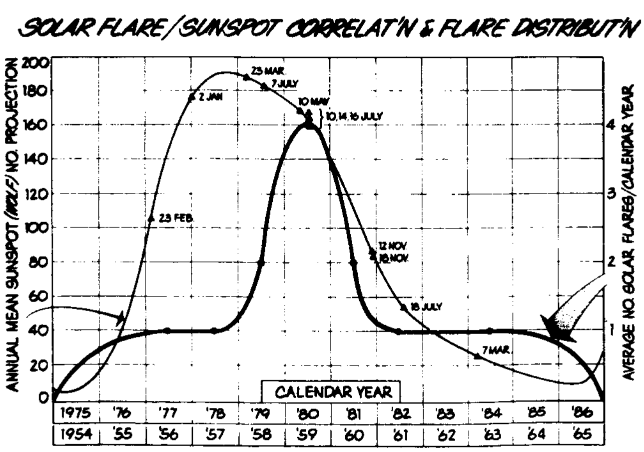


FIGURE 3.3-1

3 3.2 Solar Flare Shielding. The periodic maxima and minima of

solar activity, which occur on the average of an 11-year cycle, can be represented by a plot of mean annual sunspot (Wolf) numbers as a function of calendar year, as is shown in Figure 3.3-1. Although the appearance of active regions producing major flares is not strongly correlated with the maximum of the 11-year cycle of solar activity, in this study the flares of cycle 19 (the current cycle) were projected as cycle 21 on the assumption that the flares will occur on the same month and day in both cycles. This overall model was based on 12 solar-flare events, each with an integrated dose greater than 50 rad. The model flare used was the 10 May 1956 solar flare. This event was a low-energy, high-intensity flare having an energy range up to about 500 Mev, and a total integrated dose of 440 rads. To facilitate calculation of shield weights, the curve showing the number of flares per year was proposed for use. The number of flares, N, to be shielded against was determined from the following expression: $N = \eta D$, where D is the mission duration in years. Since the variation in IMIEO with shield mass was to be investigated in a sensitivity analysis later in the study, the weights resulting from this model were considered as baseline numbers only.

3.3.3 Utilization of Intrinsic Shielding. In determining the required thicknesses of shielding material, the presence of intrinsic shielding was taken into account, including (1) neutron attenuation from the reactors by the available liquid hydrogen propellant, (2) space radiation attenuation by approximately 15 gm/cm^2 of the mission module structure and equipment, and (3) protection afforded by the on-board water supply.

3.3.4 Solar Flare Protection. Since radiation from solar flares is essentially isotropic, the "storm cellar" concept was employed in order that protection could be afforded on all sides of the crew. Borated polyethylene was selected as the primary shielding material. The effect of displacing an equal amount of polyethylene by the minimum amount of on-board water is shown in Figure 3.3-2. An aluminum structure, having a thickness of 2.0 gm/cm^2, was provided on each side of the cellar to add strength and durability.

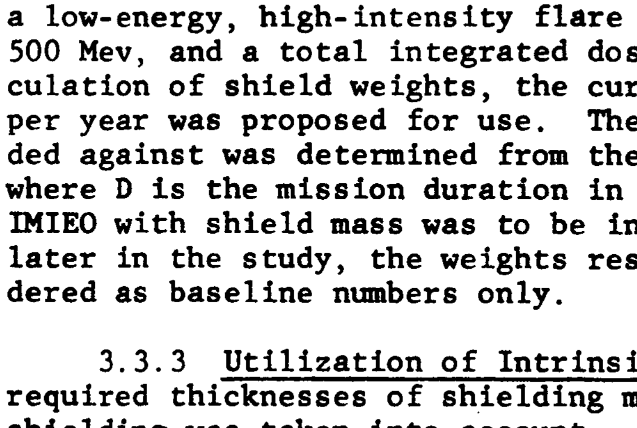


FIGURE 3.3-2

3.3.5 Shielding from Reactors. The radiation from on-board nuclear reactors that presents a hazard to the crew consists of gamma rays and neutrons. On the assumption that sufficient hydrogen would be available at all times to attenuate the neutron radiation, additional shielding was provided for gamma radiation only. Uranium was chosen

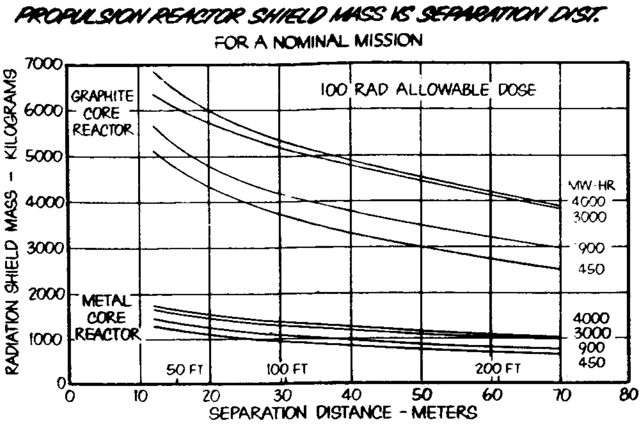


FIGURE 3.3-3

as the shielding material because of its high Z-number and good structural properties. The shields for the propulsion and auxiliary power unit are shadow shields, placed adjacent to their individual reactors. The shield mass for various mission module/reactor separation distances and reactor energy levels is shown in Figure 3.3-3.

3.4 LIFE-SUPPORT SYSTEM

3.4.1 General. A life-support system was defined for the Mars mission module to provide thermal and atmospheric control of the cabin, food, and waste-handling systems, and the required crew-support equipment. The various components and processes to accomplish these functions were evaluated, and a reference life-support system was selected. In this system, food was stored (80% dehydrated) rather than regenerated. Carbon dioxide was reduced by the Sabatier process to recover oxygen. The oxygen supply was stored as water and integrated with the water-management system. An ion exchange membrane water electrolysis cell was utilized to generate oxygen from water. Odor and contaminant control were achieved by a catalytic burner, absorption, and filtration. Waste products were stored aboard the spacecraft. Thermal control was provided by a heat-transport, fluid-space radiator system.

3.4.2 Reference Life-Support System. The reference life-support system served as a base for the generation of parametric data on mass, volume, and electrical power requirements. Mass requirements for the selected life-support system are shown in Figure 3.4-1 as functions of mission duration and crew size. The contribution of the major items to life-support system mass is shown in Figure 3.4-2 for a 6-man crew.

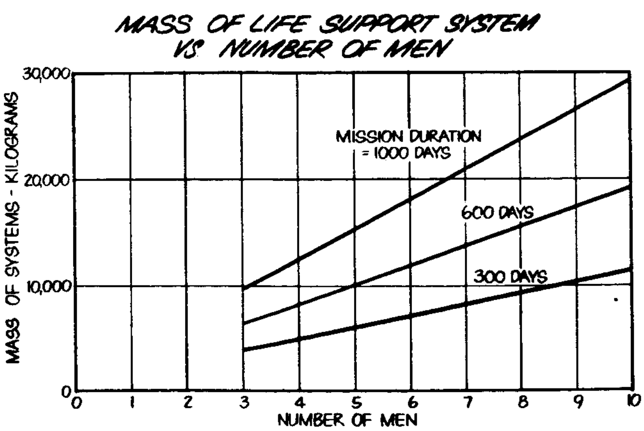


FIGURE 3.4-1

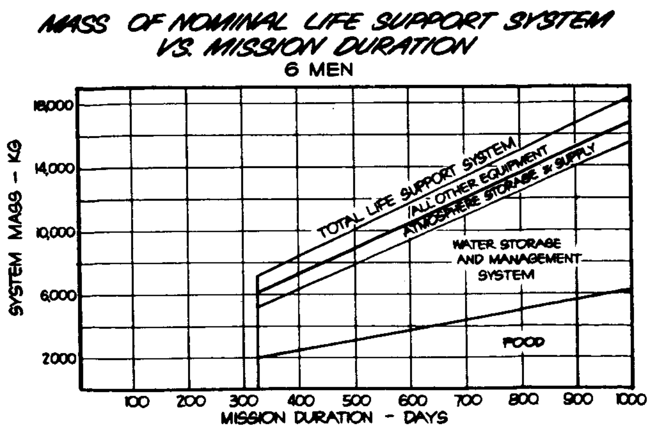


FIGURE 3.4-2

Launch date has a negligible effect on life-support system mass; it is, however, reflected in mission duration. The volume of packaged equipment was estimated by using an average density of 0.48 gm/cc.

Additional studies and developments will be required before specific definition can be made of the following: (1) complete carbon dioxide collection and reduction systems, (2) atmosphere constituent storage schemes, (3) micrometeoroid protection requirements, (4) odor and contaminant control systems, and (5) water purification processes. Biological regeneration of food and oxygen was not considered feasible in the 1975-1985 time period.

3.5 PROPELLANT STORAGE ANALYSIS

3.5.1 General. The heat sources considered in the analysis were (1) solar radiation, (2) planetary infrared and albedo, and (3) heat shorts.

3.5.2 Propellant Storage. In order to provide a uniform comparison of the effects of the various mission and trajectory parameters, propellant mass storage penalties (insulation mass plus boil-off mass) were evaluated on the basis of tanks vented at constant pressure wherein all heat absorbed by the tank is rejected by boil-off. Insulation thickness was determined so that the combined insulation mass plus the boil-off mass was a minimum. The analysis was based on an assumption that the vehicle is solar-oriented so that only the ends of the propellant tanks are exposed to solar radiation. An expression was developed which permits evaluation of the integrated time variation in solar heating for each trajectory, thus making it possible to accurately compare propellant storage mass penalties for the various trajectories. This method involves integrals of the vehicle's motion that can readily be evaluated in terms of standard trajectory parameters.

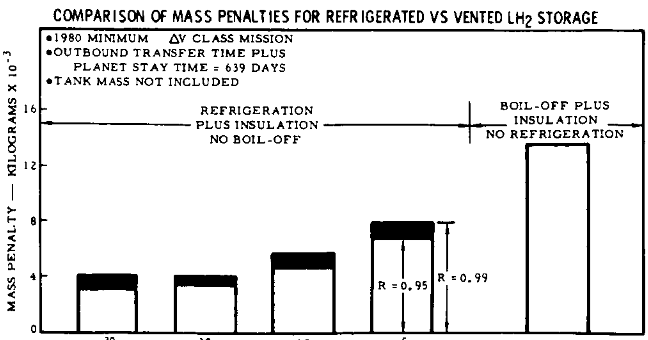


FIGURE 3.5-1

3.5.3 Refrigeration. For a number of select missions, propellant storage mass penalties were evaluated by postulating the use of refrigeration systems to reject the absorbed heat. The total refrigeration plus insulation mass penalties were calculated parametrically in terms of required reliability and assumed values of unit mean-time-before-failure. Sample results are compared with the boil-off scheme in Figure 3.5-1. The refrigeration unit masses increase at lower values of mean-time-before-failure and

at higher required reliability values because redundant refrigeration units are included in order to achieve the required reliability. Thus, a question is raised about the desirability of using refrigeration units because of the uncertainty attendant upon estimates of their mean life Refrigeration unit masses were calculated on the basis of data presented in General Dynamics/Astronautics Report No. AOK63-0001, "A Study of Early Manned Interplanetary Missions - Final Summary Report" (1963).

3.6 NAVIGATION AND GUIDANCE SYSTEM

3.6.1 The Guidance System. In order to meet the requirements considered for the orbital, planetary departure, heliocentric transfer, and planetary approach phases of a mission, the space navigation and guidance system selected in the study consisted of a variety of optical sensors, an atomic reference clock, a gyroscopically stabilized platform with accelerometers, a computer, and the required display and control equipment. The system is shown in Figure 3.6-1. Although the basic mission parameters, such as mission duration and crew size, have very little effect on the physical characteristics of the system, launch date (calendar year) variations which will reflect state-of-the-art increases in performance were considered.

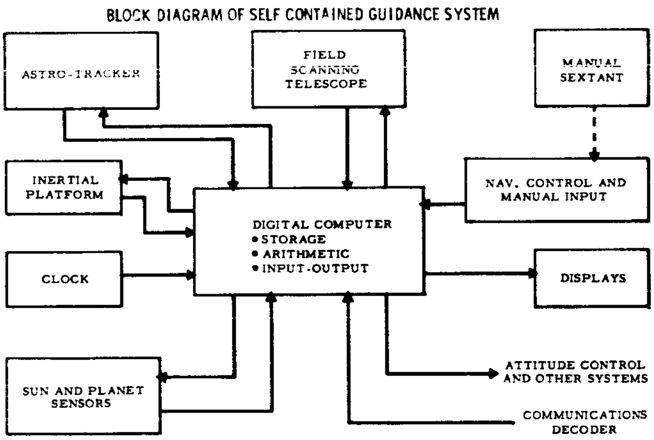


FIGURE 3.6-1

3.2.6 Guidance Corrections. The guidance scheme to be implemented was based on the so-called explicit or path-adaptive method. This scheme may be used to accommodate relatively large departures from the nominal trajectory, while the guidance correction ΔV requirements remain modest, and the correction application, infrequent. Based on a nominal injection error of 15 fps and a mid-course correction of 3 to 4 fps, the total, one-way guidance correction ΔV is shown as a function of one-way transfer time in Figure 3.6-2. A constant ΔV value of 0.25 km/sec was used throughout the study. The various guidance corrections are accomplished by a LO_2-LH_2-Be propulsion system.

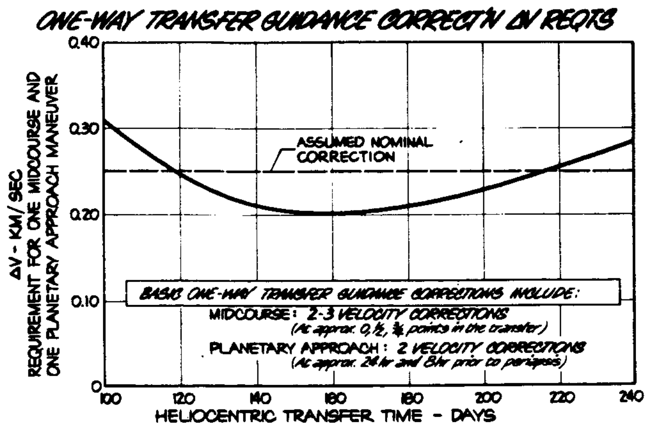


FIGURE 3.6-2

3.7 ATTITUDE CONTROL SYSTEM

3.7.1 <u>General</u>. Attitude control for a Mars mission vehicle may be achieved by use of a conventional control system which incorporates position and rate feedback and reaction jets for control torque sources. Present state-of-the-art attitude control system components and angular rate and position sensors can be used to satisfy the requirements for maintaining attitude to an accuracy of $\pm 5^{\circ}$ during coast, and $\pm 0.25^{\circ}$ during thrusting periods. Spinning a section of the vehicle to produce artificial gravity adds some complication to the analysis of the control system performance. It appears, however, that no significant problems should be encountered in providing adequate attitude control and damping for the artificial-gravity vehicle configuration.

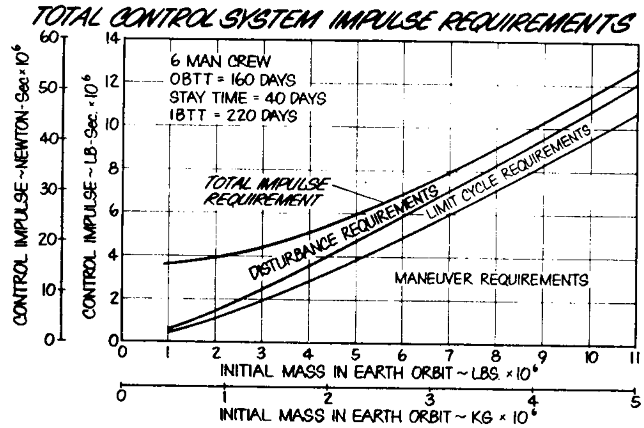


FIGURE 3.7-1

3.7.2 <u>Impulse Requirements</u>. The estimated nominal impulse requirement for attitude control based on analysis of maneuver, limit cycle, and crew and equipment disturbance requirements is shown in Figure 3.7-1. These masses include the solar orientation requirements imposed for the propellant storage analysis, and a 25% reserve. The requirements resulting from spinning a section of the vehicle to produce artificial gravity will add approximately 1400 kg to the system mass for a 6-man mission module.

3.7.3 <u>Attitude Control System</u>. Because of the sensitivity of the attitude control system to vehicle configuration, the overall system was considered rather than specific staging arrangements. In the mission analysis computer program, a semi-arbitrary percentage of the system was assumed to be staged at various points along the trajectory. The propellant masses were based on the LO_2-LH_2-Be system having a vacuum Isp of 500 seconds and a pulsing mode Isp of 390 seconds. The masses shown in Figure 3.7-2 include tankage, pressurization, feed system, and chambers. The redundant system includes completely redundant chambers and liquid feed systems.

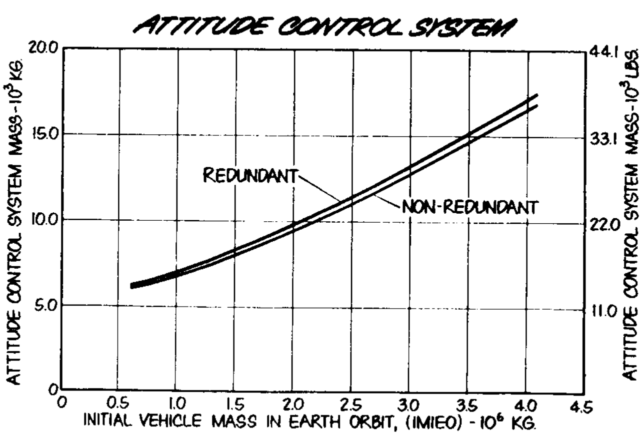


FIGURE 3.7-2

3.8 SCIENTIFIC INSTRUMENTATION

3.8.1 General. The basic objectives set for the scientific instrumentation provided for a manned Mars mission would be (1) to measure the properties of interplanetary space, (2) to select a suitable landing site for the Mars Excursion Module, and (3) to measure the properties of Mars, its environs, and its satellites. The approach taken in the study was not to select one optimum set of instruments to obtain these objectives, but to determine 3 sets of scientific experiments and corresponding sensor complements which could be categorized as minimal, nominal, and desirable, respectively.

3.8.2 System Payload Requirements. For use in the mission analysis program, system masses of 660, 940, and 2600 kg were established; these correspond to the masses estimated for the minimal, nominal, and desirable systems, respectively. The desirable system includes both a surface rover and a "Mars Pipercub" for large-scale surface exploration.

3.9 COMMUNICATIONS SYSTEM

3.9.1 General. The communications equipment provided for the spacecraft will allow communication between Earth and the Mars Mission Module (MMM), and between the MMM and the Mars Excursion Module (MEM). The distances between Earth and the MMM will result in a high space attenuation of RF energy. Typical missions for the 1975-1985 time period will involve a maximum transmission distance of 3×10^8 km. Thus, the system selected must be designed to transmit with sufficient power to overcome an attenuation of 163 db at a frequency of 8450 megacycles.

3.9.2 Equipment. Transmissions to Earth will involve a 2000-watt 100-kilocycle bandwidth, RF transmitter at 8450 megacycles and a 10-foot parabolic antenna for the MMM, and a 210-foot receiving antenna and a sensitive 100-kilocycle bandwidth receiver on Earth. Transmissions from Earth to the MMM will involve a 200-watt, 10-kilocycle bandwidth, RF tranmitter and a 210-foot antenna on Earth, and a 10-foot parabolic antenna and a sensitive 10-kilocycle bandwidth receiver for the MMM. Communications between the MMM and the MEM were considered to be provided by a UHF link and a HF backup link. Identical receivers and transmitters were considered for use on the MMM and MEM to reduce the diversity of spares. Other communications equipment was also considered to be provided, including transponders, homing radars, rendezvous equipment, television, and intercoms.

3.9.3 System Mass. The total communications system mass was estimated to be 435 kg for a mission which will commence in 1975. This mass was assumed to be reduced through state-of-the-art increases in system performance to a value of 260 kg by 1985. The data link bandwidth of the MM-to-Earth transmitter was a major factor in defining

maximum input power for the communications system. On the basis of an assumed data link bandwidth of 100 kilocycles, the maximum input power required for system operation was estimated to be 10 kilowatts. Maximum heat dissipation from the equipment at this input level was estimated to be 8 kilowatts.

3.10 AUXILIARY POWER SYSTEM

On the basis of a preliminary loads analysis of the various vehicle subsystems considered in the study, the peak electrical power input levels for the nominal payload equipment would total approximately 22 kilowatts. For this anticipated power level and a nominal mission duration of 10,000 hours, a nuclear electrical power system appeared to be the most practical system to be considered for the 1975-1985 time period. The estimated electrical load for the Mars mission is on the same order of magnitude as the anticipated output of the SNAP-8 system, which is being developed by AEC and NASA. However, current effort on the SNAP-8 system is not being directed toward the development of a system suitable for planetary exploration missions. If a nuclear electric auxiliary power system is to be developed in time to be successfully utilized in a 1975 Mars mission, a program to develop this power system must be started immediately. Nevertheless, such a program has been assumed in this study. A mass of 7000 kg was assumed for a redundant nuclear Brayton-cycle type of system (one system on 100% standby). This mass does not include the shielding mass; however, it is included in the analysis of radiation shielding requirements.

3.11 EARTH REENTRY MODULE

3.11.1 General. In order to determine the feasibility of accomplishing direct aerodynamic Earth reentry at velocities compatible with those which might be experienced on returning from Mars missions in the 1975-1985 time period, in-house studies were performed at General Dynamics/Fort Worth prior to the present study. The purpose of these studies was not to define an optimum entry vehicle, but to obtain the environment to be encountered by a typical entry configuration.

EARTH REENTRY MODULE
MODIFIED LANGLEY SHAPE

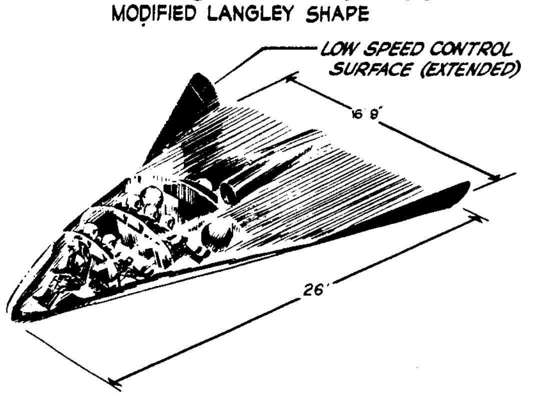


FIGURE 3.11-1

3.11.2 <u>Reentry Vehicle Studied</u>. The configuration chosen for the analysis was a modified Langley HL-10 vehicle having a hypersonic L/D_{max} of approximately 1.0. Excluding its heat shield and 6-man crew, the basic vehicle with a small turbofan engine, as shown in Figure 3.11-1, weighs approximately 4000 kg (8800 lbs). Entry trajectories were run at entry velocities of 12.2, 14.3, 18.3, and 21.3 km/sec

(40,000, 50,000, 60,000, and 70,000 fps, respectively) for L/D_{max} and $C_{L\ max}$ entry modes.

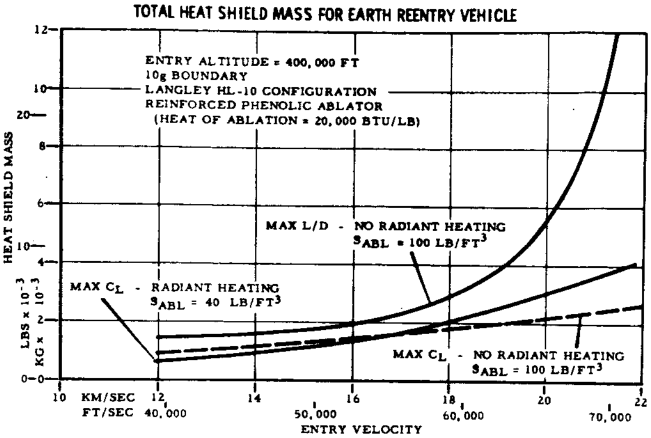


FIGURE 3.11-2

3.11.3 Heat Shield Analysis. Preliminary heat transfer calculations were performed by taking into consideration only the convective heating of the vehicle. A constant effective heat ablation of 20,000 BTU/lb, and a density of 100 lb/ft^3 were used in determining the heat shield mass. Based on these results, more detailed calculations (Figure 3.11-2), including the effect of radiant heating, were made for the $C_{L\ max}$ entries for a more realistic ablation density of 40 lb/ft^3.

3.12 MARS EXCURSION MODULE

3.12.1 General. Prior to the present study, in-house studies were performed at General Dynamics/Fort Worth for the purpose of establishing design concepts for 2 types of vehicles, ballistic and glide. These vehicles were suitable for use in entering into the Martian atmosphere from orbit, landing on the Mars surface, and eventually returning from the surface to orbit. The intent of these design studies was to provide direction for future vehicle analysis, not to define optimum configurations. In the results of these studies, it was demonstrated that both types of vehicles were capable of performing the MEM mission.

3.12.2 Vehicles. The configurations shown in Figure 3.12-1 for the glide- and ballistic-type vehicles have a maximum gross weight of 32,800 kg (70,000 lbs), a crew of 3, and the capability of being stored in the interplanetary spacecraft within a 7.6 meter (25 ft) diameter.

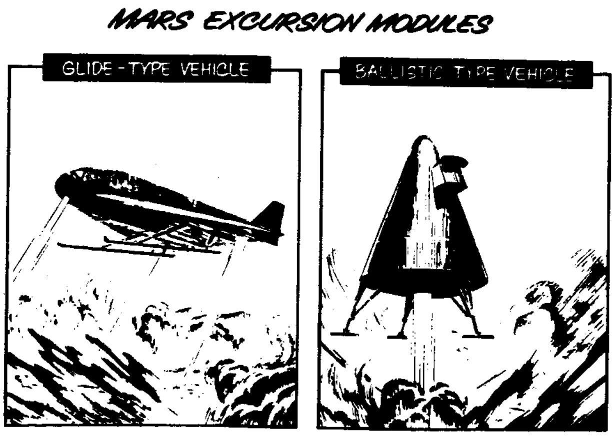


FIGURE 3.12-1

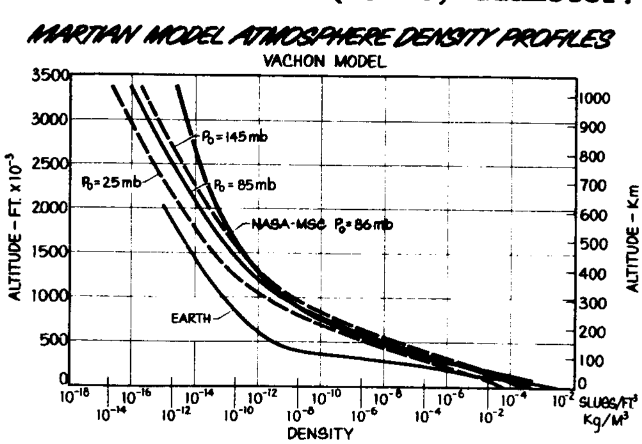


FIGURE 3.12-2

3.12.3 Mars Atmosphere Model. The Mars atmosphere model proposed by Vachon has been used throughout the Unfavorable Time Period study. The basic model, as indicated by the 85 mb surface pressure curve in Figure 3.12-2, has a mean surface temperature of 273°K (0°C) and a tropopause at 24 km. To account for the atmospheric uncertainties, selected points were investigated for the other models shown in the figure.

3.13 SPACECRAFT MAIN PROPULSION

3.13.1 Nuclear Systems. A basic specific-impulse growth curve has been established for a second-generation, graphite-core engine with the characteristics shown in Figure 3.13-1. The shaded area indicates the range of variation in Isp (approximately 10%) that was investigated. The broken line indicates that a metal-core reactor, having approximately the same Isp as the graphite engine but different physical dimensions, could also be available by 1975. Although additional development could also produce the gaseous core reactor, it must be accepted that, in all probability, funding will not be made available to provide 3 operational engine systems in the 1975-1985 time period. Consequently the majority of the effort in the study has been concentrated on the graphite engine.

3.13.2 Chemical Systems. In the investigation of chemical propulsion systems to be used in the mission studies, the most advanced of the feasible bipropellant systems was considered. Included as representative examples were the propellants $LOX\text{-}LH_2$, $LF_2\text{-}LH_2$, and $LOX\text{-}LH_2\text{-}Be$. Since the $LOX\text{-}LH_2\text{-}Be$ system offers an Isp of 500 seconds and has been operated experimentally, this system has been selected as representative of the bipropellant engines of the subject time period. (See Figure 3.13-2) Both Earth entry and mid-course correction propulsion will be provided by this system.

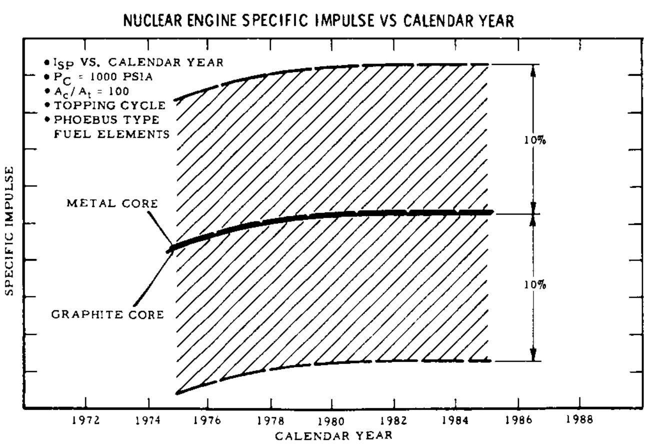


FIGURE 3.13-1

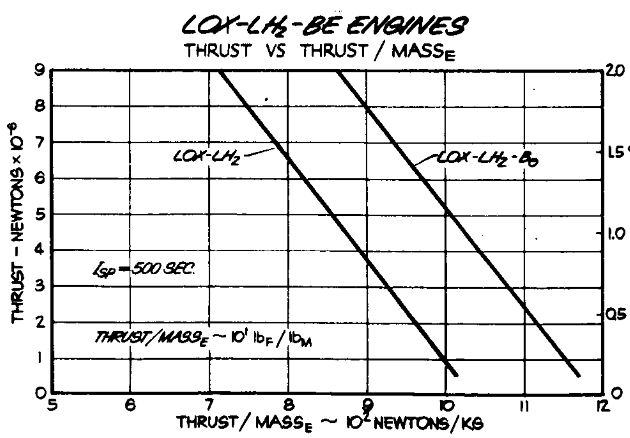


FIGURE 3.13-2

4.0 SYSTEMS TRADE-OFF ANALYSIS

4.1 OBJECTIVES

The basic aim in the systems trade-off analysis was to establish estimates of performance and reliability based on the state-of-the-art for all of the major systems required to accomplish a manned Mars mission. These estimates were compiled in parametric form so that the complete range of significant alternatives could be readily weighed and traded. Functional state-of-the-art relationships, involving mass, performance reliability, and time, were developed for each critical system. In Figure 4.1-1, a listing is given of the systems treated in the analysis, and the type of trade-off developed for each system is illustrated.

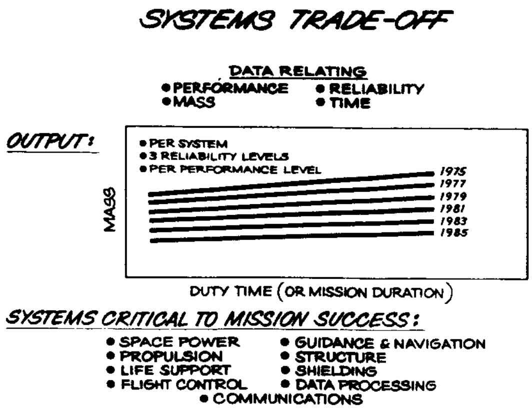


FIGURE 4.1-1

4.2 METHODS

For each subsystem, reliability estimates were derived from mean-time-between-failure (MTBF) growth curves for the critical subassemblies on the basis of existing values, target specifications, and engineering judgment. Reliability for any calendar year was regulated by changing the redundancy or spares complement. In determining overall system reliability, an allocation was used to optimize reliability with respect to system mass. In order to establish the relationships among total system mass, reliability, calendar year, and duty time, these results were integrated with (1) the anticipated reduction in system mass, and (2) estimates of improvements in performance. The procedures which were utilized in determining these relationships are outlined in Figure 4.2-1.

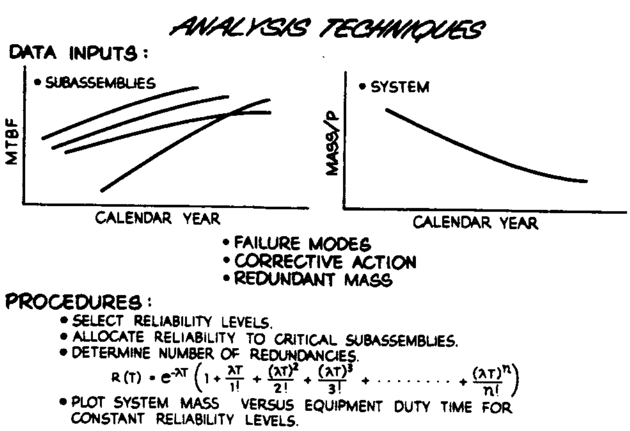


FIGURE 4.2-1

4.3 RESULTS

Some of the typical results are shown in Figure 4.3-1 and Figure 4.3-2. The total system mass in Figure 4.3-1 includes all equipment

that can program or relate radio frequency propagation, and the necessary self-test equipment and handbooks. Reliability was regulated only for vital communications links equipment. However, the increase in mass required to raise the system reliability of this equipment from 95% to 99.5% is small in comparison to increases required for most of the other major systems.

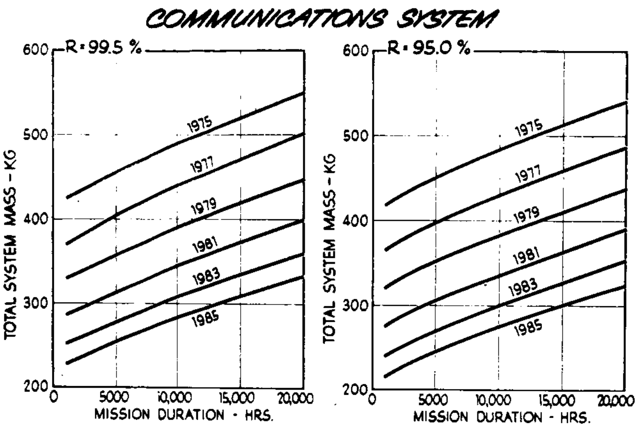


FIGURE 4.3-1

Trade-off relationships for the life-support system are illustrated in Figure 4.3-2. Although a considerable amount of redundancy is required to raise reliability from 95% to 99.5% for the active equipment (mechanical or chemical), the percentage of total system mass is small. However, because of the greater total mass for this system, the increment required to increase the reliability results produces a significant increase in IMIEO.

In the case of the primary propulsion system, repair or total redundancy is impractical. A high inherent reliability is essential and is provided for in the development cost estimates. Structural reliability and calendar-year effects were translated into allowable safety factors, as indicated in subsection 3.2.5. Some of the systems defined in less detail, in particular those for space power and life support, may require reliability, performance, and mass adjustments after firm designs become available. However, for the purposes of this study, these estimates are assumed to be adequate.

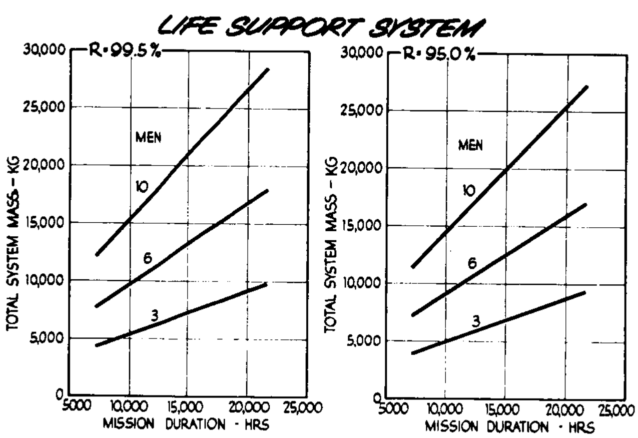


FIGURE 4.3-2

The assumption that the crew will have adequate time to perform the maintenance functions along with the routine monitoring functions is supported by the preliminary results of an independent study at General Dynamics/Fort Worth. It has been indicated that, under design conditions, a crew of 3 can comfortably perform the maintenance as well as the operations. The reliability of each concept of the total system was derived on the basis of these data and used as inputs in analysis to determine the probability of mission success.

5.0 CONCEPTUAL AND PRELIMINARY VEHICLE DESIGN STUDIES

5.1 GENERAL

In the initial phase of the study, the primary effort in configuration design was directed toward generating basic vehicle concepts. From these various concepts, a nominal configuration, compatible with the mission defined in Figure 1.0-2, was selected to provide a basis for the parametric mission analysis. It was necessary to key the mission analysis to a specific vehicle because many of the vehicle and payload parameters were extremely sensitive to configuration characteristics.

The remaining phases of the study were devoted to the investigation of artificial gravity concepts and arrangements of the mission module, additional propulsion systems, and the overall vehicle. Particular emphasis was placed on the investigation of vehicles which would be suitable for establishing an orbit at Mars by use of partial atmospheric braking.

5.2 NOMINAL INTERPLANETARY VEHICLE

The Nominal manned Mars vehicle is shown in Figure 5.2-1. Basically, it is composed of 3 manned spacecraft components and the propulsion system required to enable these components to perform their specific function during the Mars mission. The 3 manned components consist of (1) the Mission Module, which provides living quarters for the crew throughout the mission; (2) the Mars Excursion Module, which is used for transportation between the Mars' orbit and surface of the planet; and (3) the Earth Reentry Module, which provides a capability for atmospheric entry, landing, and the return of the crew to Earth. The Nominal vehicle is composed of 6 propulsive stages; each is jettisoned after its particular thrust application. Nominally, 3 stages are chemical-type propulsion systems, and the other stages employ nuclear propulsion arrangements. Included in the chemical systems are (1) the outbound and inbound mid-course correction propulsion, and (2) the Earth-braking propulsion. The Earth-escape, Mars-braking, and Mars-escape stages are nuclear rocket systems.

Each of the Earth-escape and Mars-braking propulsive stages is composed of a nuclear rocket engine and 4 propellant tanks. The 8 liquid hydrogen propellant tanks of these 2 stages are clustered around a central tank which contains the Mars-escape propellant. This single, central liquid-hydrogen tank and a nuclear rocket engine make up the Mars-escape stage, which serves as the structural core of the entire interplanetary vehicle. Each of these liquid-hydrogen tanks has a length-to-diameter ratio of 2.

The Earth-braking propulsion system, basically composed of a

fuel tank, an oxidizer tank, and an engine, is situated forward of the Mars-escape stage. The 3-man Mars Excursion Module is attached alongside the Earth-braking stage by means of a mechanical trapeze arrangement. The Mission Module, a two-story cylindrical structure, is located forward of the Earth-escape stage. Provisions are made for crew transfer to Excursion Module when the trapeze is in the retracted position. The 6-man Earth Reentry Module is located just forward of the Mission Module, and access is also provided for crew transfer between these two components. The propulsion system for outbound mid-course correction is located forward of the Earth Reentry Module.

NOMINAL VEHICLE CONFIGURATION

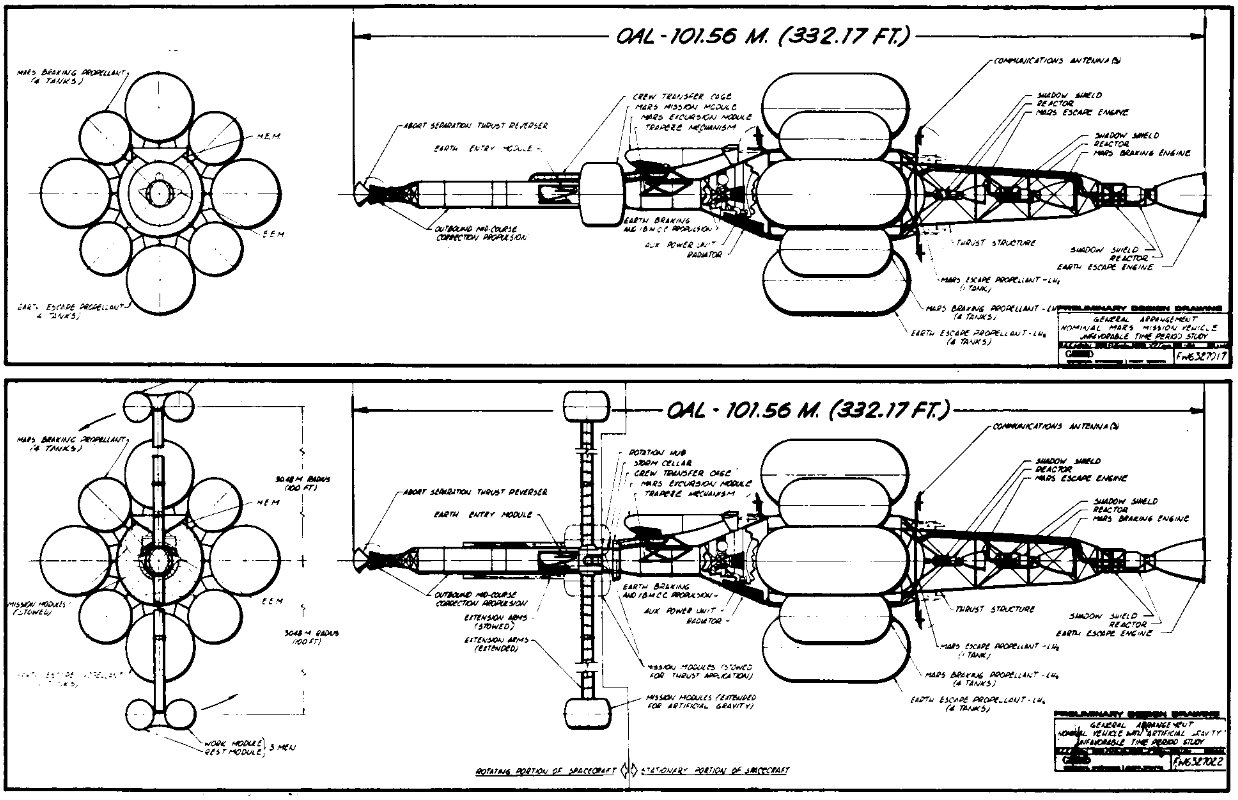


FIGURE 5.2-1

The inbound mid-course correction system (MCC) will be located forward of the Mission Module adjacent to the outbound MCC or combined with the Earth-braking system, depending upon the size of the system. Each of these systems is composed of a fuel tank, an oxidizer tank, and a rocket engine. The Snap-8 auxiliary power system is located just aft of the Earth-braking propulsion stage so that the fuel tank will provide some of the required shielding from the reactor.

In the vehicle illustrated in Figure 5.2-1, a thrust-reversing arrangement in conjunction with the propulsion system for outbound mid-course correction provides for separation of the Earth Reentry Module from the main spacecraft in case abort is required during the Earth-escape maneuver.

No attempt was made during this study to determine whether artificial gravity is required for a mission of this type. However, concepts for providing the addition of this capability to the Nominal vehicle were investigated to estimate the cost of the system in terms of weight and complexity. The arrangement shown in Figure 5.2-1 is reasonably compatible with the Nominal vehicle, and only minor changes in the overall vehicle are involved. In this arrangement, all of the Mission Module volume can be rotated except that of the "storm cellar." The storm cellar is housed in a cylindrical container which is rigidly attached to the main vehicle at the aft end and mated to a pair of bearings at the forward end. The remaining volume is divided into 2 sets of dual-cylinder compartments, each housing 3 men. These compartments are attached to the central rotating hub structure by a folding parallelogram arrangement of the extension-arms; the arms are motor-driven and gear-actuated to provide controlled deployment. Intermodule crew access is provided by ladders on the extension arms. Intermodule crew movement is assumed to be minimum. Direct access to the storm cellar is possible at all times whether the crew modules are rotating or are stowed for thrust applications. The rotational drive system is arranged so that the portion of the vehicle located forward of the hub bearings is rotated to the desired angular velocity by reaction jets located at the extremities of the extended modules. Bearing friction, which causes the aft portion of the vehicle to rotate, is overcome through torque applied by a synchronous electric motor. This system enables the aft portion of the vehicle to remain fixed with respect to inertial space; thus, navigational observations can be accomplished from a non-rotating platform.

5.3 ATMOSPHERIC BRAKING AT MARS

The investigation of the concept of employing atmospheric braking as a means of accomplishing the Mars-capture maneuver was restricted to ballistic-type entry configurations. The 2 basic entry shapes, shown in Figure 5.3-1, were examined during the course of this study.

A configuration employing an Apollo-type entry shape is illustrated in the upper portion of Figure 5.3-1. This arrangement is similar to the Nominal vehicle, except as follows. The nuclear Mars propulsive capture stage has been replaced by both a heat shield and a chemical propulsion system employed in circularizing the vehicle's orbit about Mars. The vehicle is basically conical with the propulsion systems for Earth-escape and outbound mid-course attached aft of the heat shield. The heat shield is attached to a spherical, Mars-escape propellant tank through the use of a conical structure. The nuclear engine for Mars escape and the propulsion system for establishing Mars orbit are also enclosed within this structure.

The conical entry shape concept, illustrated in the lower portion of Figure 5.3-1, is essentially the same with respect to the overall vehicle design as that of the Apollo-type configuration. The

differences are in the heat shield shape and the Mars-escape stage tankage arrangement.

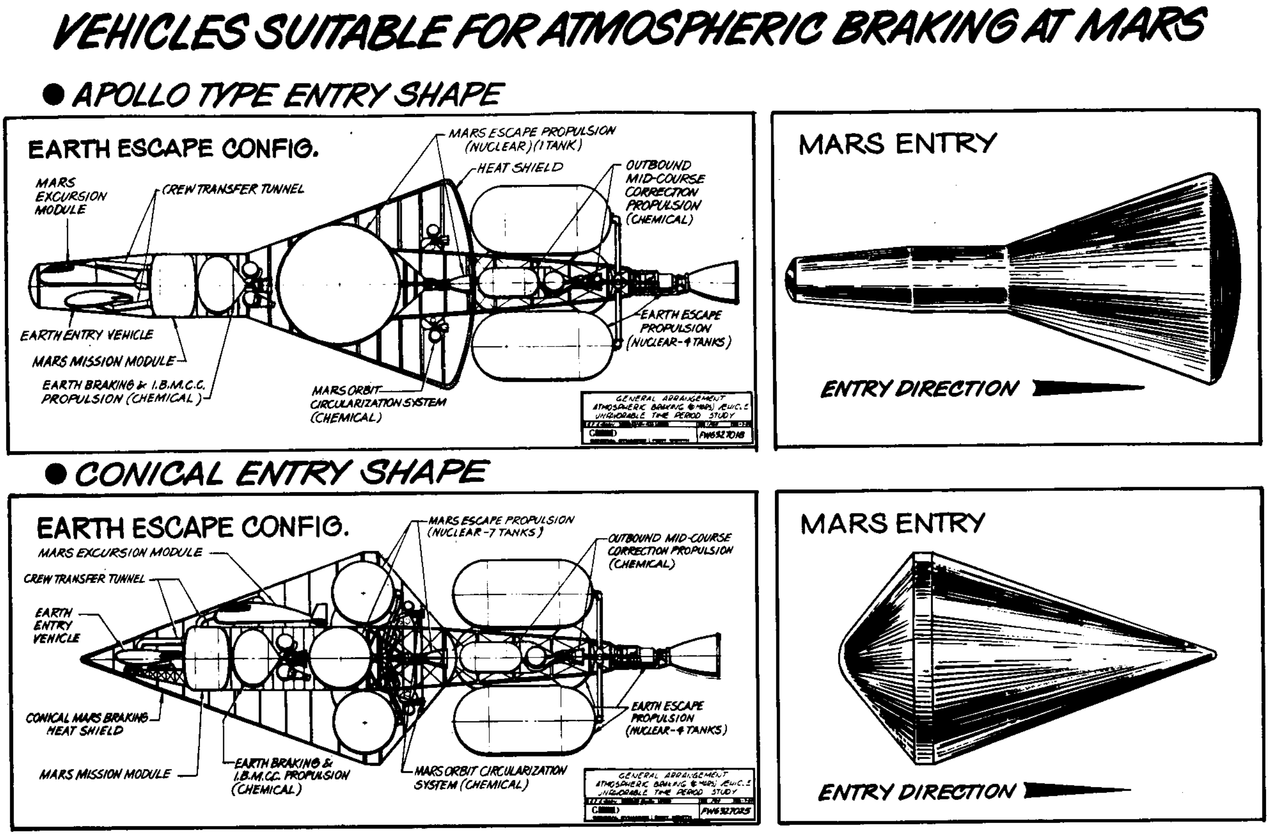


FIGURE 5.3-1

6.0 ATMOSPHERIC BRAKING AT MARS

6.1 GENERAL

One technique which could be used to reduce the total initial mass in Earth orbit (IMIEO) requirements for Mars missions would be to establish an orbit at Mars by the use of partial atmospheric braking rather than by total propulsive braking. Discussed in this section are a preliminary investigation of vehicles suitable for accomplishing atmospheric braking at Mars, and an analysis of the corresponding entry environment encountered. Because of the preliminary nature of the study, only the ballistic-type entry vehicles shown in Section 5.0 were investigated. The basic objective of the study was to determine heat shield requirements and the corresponding vehicle IMIEO. The analysis was based on a 1979 mission, and a Mars entry velocity of 8.93 km/sec (29,300 ft/sec). Vehicle stability during entry was not considered.

6.2 FLIGHT MECHANICS

The geometry and conditions assumed for the Mars atmospheric braking trajectory analysis are shown in Figure 6.2-1. The atmosphere definitions which were used are presented in Figure 3.12-2. They correspond to those of the Vachon Model, in which a surface pressure of 85 mb is assumed, and Vachon Model variations with surface pressures of 25 mb and 145 mb, respectively. All of the curves shown in Section 5.0 are based on the 85 mb Vachon Model. Some of the assumptions made about the 25 mb and 145 mb variations are presented below.

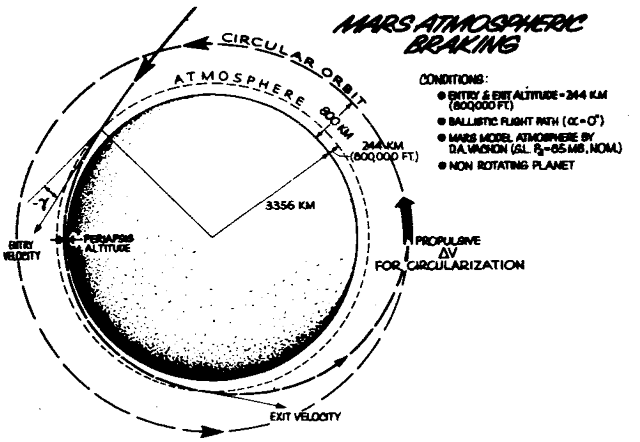


FIGURE 6.2.1

From a trajectory standpoint, the feasibility of Mars atmospheric braking depends upon

1. Achieving a substantial velocity reduction without impacting or requiring excessive deceleration, and

2. Maintaining an entry corridor width consistent with guidance limitations.

Based upon the analysis performed, the following conclusions were made relative to these critical points:

1. For the Vachon Model Atmosphere and the 2 variations noted above, Mars atmospheric braking down to circular velocity (for entry velocities up to 10 km/sec) can be accomplished in a single pass without impacting or exceeding a deceleration of 8 Earth surface G.

2. Entry corridors are very narrow; exit velocity variation in circular velocity up to escape velocity corresponds to entry angle corridor widths of from 1° to $1/4^{\circ}$. Moreover, the nominal entry angle values vary with atmosphere definition to the extent that there is no corridor overlap for the range of possible atmospheres assumed. (No corridor-widening maneuvers were investigated).

The above results apply to a range of $W/C_D A$ values of approximately 300 to 5000 kg/m^2.

The circularization maneuver into an 800 km orbit, following the pass through the Martian atmosphere, is illustrated in Figure 6.2-1. A representative value for the ideal velocity requirement associated with this propulsion phase is 165 m/sec. Exit at velocities in excess of the approximately circular value used should be investigated in subsequent studies, particularly as a corridor-widening technique.

6.3 HEAT SHIELD ANALYSIS

This subsection contains a brief description of the thermal analysis directed toward the evaluation and comparison of heat shield requirements for the conical and the Apollo-shaped entry bodies shown in Figure 5.3-1.

6.3.1 Convective Heating. Convective heating at the stagnation point during entry into the Mars atmosphere was based on Detra and Hidalgo's simplification (Ref. 1)* to Fay and Riddell's theory (Ref. 2). The stagnation point convective heating was conservatively computed for a cold wall. The convective heating along the cone was based on the laminar and turbulent convective heat transfer correlations of Blasius (corrected for conical flow) in connection with Eckert's reference temperature method (Ref. 3). The following conditions were established for convective heating on the sharp cone:

*The references indicated in this section refer to those discussed in the Technical Report

(1) the transport properties were considered to be those for air; (2) the pressure along the cone was predicted by the modified Newtonian theory; (3) the velocity along the cone was given by $V_\infty \cos\beta$, where V_∞ is free stream velocity and β is the shock angle obtained for conical flow of a perfect gas (Ref. 4); and (4) for reference temperature calculations, the wall temperature was considered to be the radiation equilibrium temperature. The local Reynolds number (based on length) at which transition from laminar to turbulent flow occurs was considered to be 5×10^5.

6.3.2 Radiative Heating. The heating (at the stagnation region and along the conical surface) produced by the radiation emitted from cyanogen (CN) was based on the work of G. S. Massingill at General Dynamics/Fort Worth;radiation from other chemical species is small compared to that from CN for the range of shock layer thermodynamic conditions encountered by these vehicles. For the gaseous radiation calculations, the Mars atmosphere composition was considered to be 5% CO_2 and 95% N_2. The composition (including the amount of CN) and the thermodynamic properties of the gaseous mixture in the shock layer are based on thermodynamic equilibrium calculations by O. R. Brock of General Dynamics/Fort Worth.

Radiation incident on the conical surface was determined as a function of cone half angle for an altitude of 45 km, velocities of 5, 7, and 9 km/sec, and distances from the cone apex of 3 and 30 meters. Surface absorptivity was assumed equal to unity. These data indicate that radiation is negligible along the conical surface for cone half-angles of 20° or less; therefore, a value of 20° was chosen for the conical vehicle.

6.3.3 Heat Shield Weights. The following paragraphs contain descriptions of (1) the convective and radiative heating encountered by the selected vehicles during atmospheric entry, (2) the heat shield performance, and (3) the resulting heat shield weights.

Entry Heating. Both convective and radiative heating during entry were computed by the methods described above for entry velocities from 5 to 10 km/sec. In each case, the entry altitude was 244 km and the entry angle was selected so that the exit velocity (at 244 km) was slightly above circular velocity. Typical convective and radiative heating rates during entry are shown in Figures 6.3-1 and 6.3-2. In Figure 6.3-1, the discontinuity in the convective heating rate on the cone at 96 seconds is due to the transition from laminar to turbulent flow. In all of the entry conditions considered in this study, the Apollo-shaped vehicle heat shield is exposed primarily to radiative heating, while the cone is exposed primarily to convection.

Heat Shield Performance. The heat shield weights are based on conservative, but realistic, heat shield performance. The material is assumed to be a low density phenolic (0.65 to 0.80 gm/cc, controlled

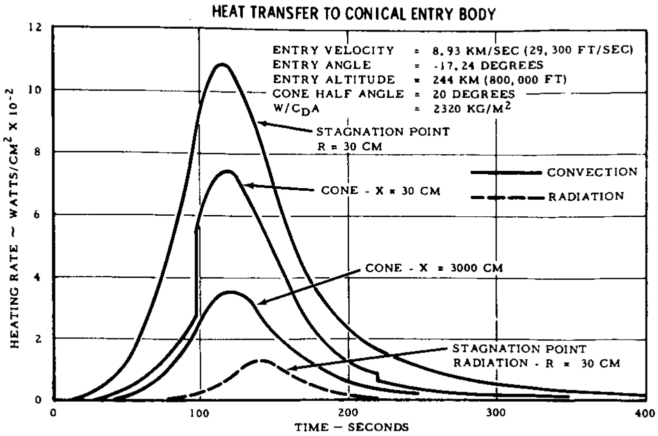


FIGURE 6.3-1

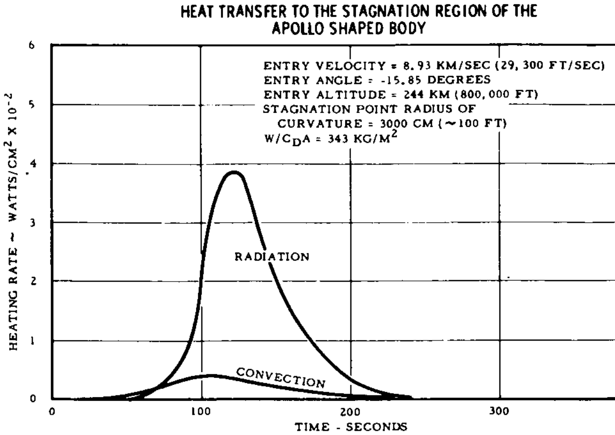


FIGURE 6.3-2

by dispersion of phenolic micro-balloons in the material) reinforced with randomly oriented quartz fibers. For purposes of this preliminary analysis, the ablation rate of the heat shield material is base on an approximate energy balance at the ablative surface. This meth accounts for the reduction in ablation material performance when exposed to radiative heating. The reduction is due to the fact that t gaseous products of ablation are highly effective in "blocking" convective heat transfer, but have no effect on radiative heating. The performance benefit due to radiation emitted by the body was not con sidered, since the surface temperature was not known. An appreciabl increase in heat shield performance can be shown by considering the effects of surface re-radiation.

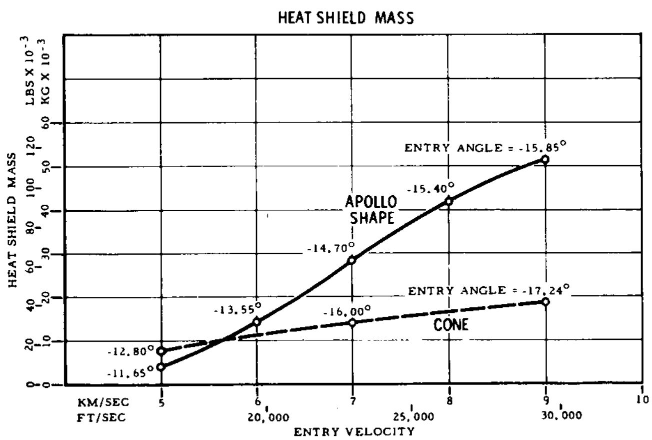


FIGURE 6.3-3

Heat Shield Weights. The estimated heat shield weights for the conical and Apollo-shap bodies as a function of entry velocity are shown in Figure 6.3-3. These weights represent the amount of heat protection material required on the forwar portion of the vehicles, i.e., the stagnation region of the Apollo shape and conical surfac plus the stagnation region of conical vehicle; the convective heating and required heat protection in the base region of the cone and conical section of the Apollo shape has not been considered. The greater rate of increase of heat shield weight with respect to entry velocity for the Apollo shape is due to its dependence on radiative heating, which increases more rapidly with entry velocity than does convective heating. Although the total heating (convective plus radiation in kilocalories) encountered by the Apollo shape during entry from a given velocity is less than

that encountered by the conical shape (a factor of approximately 3 at 9 km/sec), the heat shield weight is greater than that of the cone because of the less effective performance of the heat shield in a radiative environment.

Comparison With Earlier Results. As has been noted in earlier reports during this study, there are a number of uncertainties which exist in the prediction of radiation from cyanogen. A detailed, in-house study was made of the problem, and the radiative heating was recomputed for the 30-meter diameter, Apollo-type vehicle considered earlier. The magnitude of the change in radiative heat flux as compared to the values shown in earlier analysis is illustrated in Figure 6.3-4. The earlier data were based on the method of analysis of Kivel and Bailey (Ref. 5); in this analysis, it was assumed that electronic oscillator strengths for the CN violet and red band systems are equal (f = 0.10). The present calculations were based on the following: (1) an oscillator strength of f = 0.027 from Bennet and Dalby (Ref. 6) for the violet system, (2) the basic spectroscopic constants from Herzberg (Ref 7), and (3) the Franck-Condon factors from Frazer et al (Ref. 8) and Pillow (Ref. 9). The red band radiation was included by obtaining a ratio of red to violet intensity from James (Ref. 10) because no data were found on the oscillator strength for the red system. By the use of this ratio, which is a function of one gas temperature, the prediction based on the present calculation is as follows: equal red and violet radiation at 3300° K, and a ratio of red to violet intensity of 0.1 at 5000° K. Kivel & Bailey predict equal radiation from the red and violet band system at 7700° K, and a ratio of 4.9 at 5000° K. Based on the revised methods of computation, the heat shield mass for the 30-meter Apollo vehicle was estimated to be 6.8 x 10^4 kg as compared with 10.0 x 10^4 kg as determined by the earlier analysis.

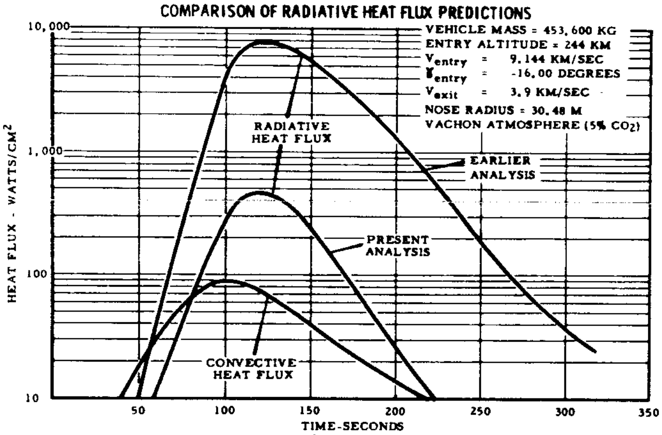


FIGURE 6.3-4

7.0 MISSION/VEHICLE PERFORMANCE ANALYSIS

7.1 CONCEPT AND SCOPE

The fact that mission analysis based on total ΔV is both convenient and useful has been discussed in subsection 2.4. However, the use of IMIEO as a criterion for mission selection is generally more satisfactory than the use of total ΔV, since IMIEO provides a reasonable measure of both overall mission difficulty and cost. Alternately, cost itself, or some other parameter, could serve as the criterion. The advantages of using IMIEO or cost are apparent; the disadvantages are (1) an increased complexity in computation, (2) the loss of generality, and (3) the dilution of results because numerous assumptions are involved. For the purposes of this study, IMIEO has been used as the basic optimization (selection) criterion.

Trajectory computations were conducted as described in subsection 2.1. IMIEO was calculated by means of a detailed iterative computer program for vehicle sizing. The IMIEO-optimized trajectories were obtained by coupling an analytical optimization program to the vehicle-sizing program. In the optimization routine, the ordinary calculus of maxima and minima, with constraints imposed with Lagrangian multipliers was utilized.

On the basis of assumptions made with regard to variations in the requirements of vehicle systems mass, various conclusions may be drawn about the relative merits of different mission types and launch windows. In the early phases of this study, estimates were made of systems' masses as functions of (1) pertinent mission vehicle parameters (mission duration, stay-time, nuclear propulsion reactor power, etc.), and (2) calendar-year effects (state-of-the-art advances, and solar activity). Curve fits of these data, presented in Section 3.0, were used to define the coefficients of the equations used in the computer program.

In this area of the study, the following 3 analyses were made:

1. A detailed definition of minimum IMIEO missions as a function of Earth launch date for the nominal mission/vehicle definition.

2. A determination of corresponding IMIEO sensitivities and variations for mission and vehicle parameters.

3. An approximate IMIEO determination for alternate mission/vehicle definitions for use in comparisons.

7.2 NOMINAL MISSION/VEHICLE DEFINITION

The Nominal mission was defined in Section 1.0. The primary characteristics are (1) maximum allowable Earth-entry velocity of 15.24 km/sec (50,000 ft/sec), (2) propulsive braking at Mars, (3) 40-day stay-time for short missions, and (4) optimum stay-time for long missions.

The Nominal vehicle is defined on the basis of the data in Section 3.0 and a number of assumptions. These assumptions, which are presented in detail in the final Technical Report, are listed in brief below.

1. Additional ΔV allowances: 3% general reserve; 250 m/sec mid-course correction for each leg; curve fits of integrated gravity losses; and no allowances for planetary orbital operations.

2. Initial stage acceleration: 0.3 Earth-surface G for nuclear, and 0.5 for chemical.

3. Crew size: 6 men

4. Artificial gravity provisions: none

5. Meteoroid protection: used for all components except Earth departure tanks.

6. Cryogenic propellant storage: insulation and boil-off determined for all tanks except Earth departure.

7. Allowable radiation dose: 200 rad.

8. Propulsion: single graphite-core nuclear engines, one each for Earth departure, Mars arrival, and Mars departure; no restart; single cryogenic chemical engines, one each for Earth arrival and each mid-course stage.

9. Reliability: 0.95 for each system.

7.3 MINIMUM IMIEO DATA FOR THE NOMINAL MISSION/VEHICLE DEFINITION

As discussed in subsection 2.4, the 4 independent variables -- Earth launch date, mission duration, stay-time, and outbound time -- have been used to define a mission.

Each of the missions is optimum (in the sense of minimum IMIEO) with respect to outbound time and mission duration. Stay-time is optimum for the long missions, and 40 days for the short missions. The launch date is as specified in Figure 7.3-1.

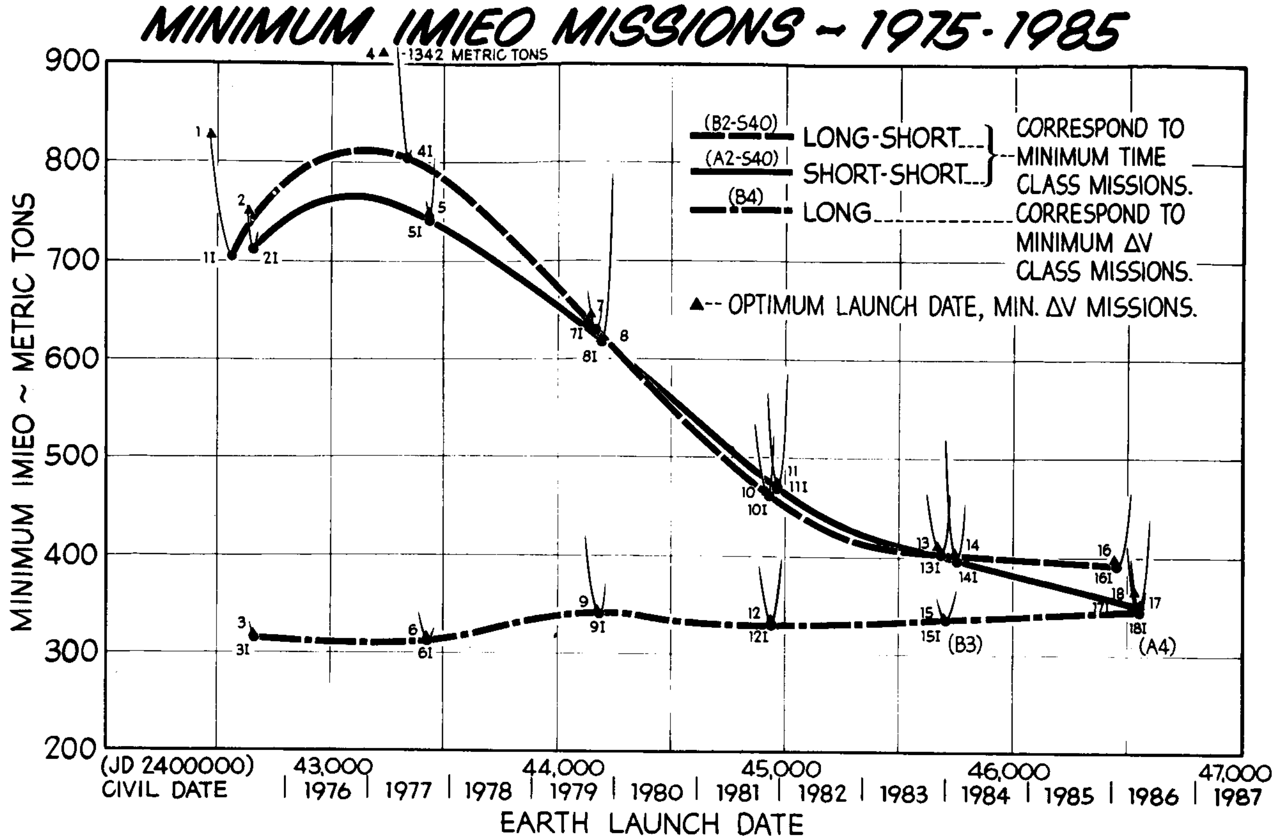


FIGURE 7.3-1

Three general classes of missions were investigated: short-short, long-short and long. Short-short and long-short missions correspond to the minimum time class missions of Section 2.3 while long missions correspond to the minimum ΔV class. These mission classes nominally bound the various other possible classes both from a ΔV and an IMIEO standpoint and from a mission duration standpoint. Figure 7.3-1 presents minimum IMIEO data for these missions for the six opposition periods from 1975 through 1986. In addition, IMIEO's are indicated for the optimum-launch date, minimum- ΔV points in each window. These points correspond to those noted in Section 2.4.

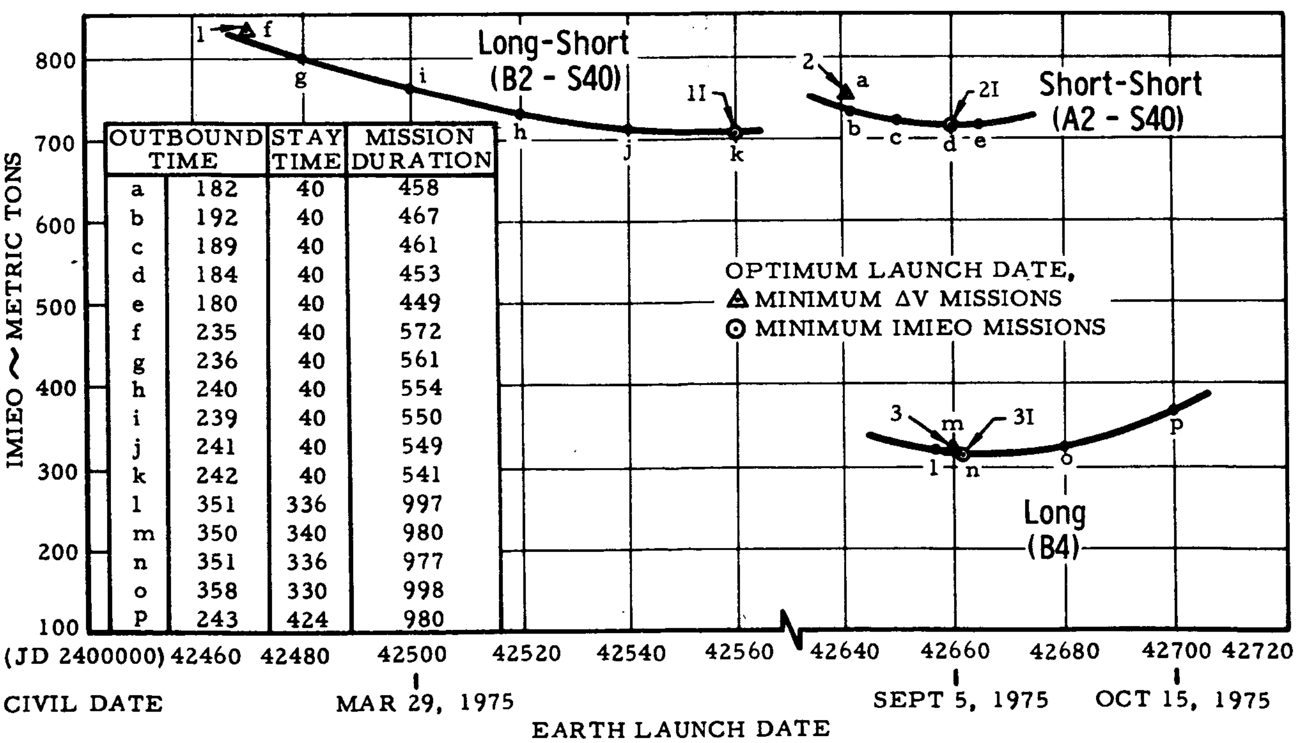
The primary calendar-year effect is the result of the eccentricity of the Martian orbit and the approximate 16-year cycle of repeated opposition locations (discussed in detail in subsection 2.7). This effect, coupled with peak solar activity in 1980 (large flare shield mass), accounts for the 1978-centered IMIEO maximum for the short missions. Because of factors such as advanced state-of-the-art and specific impulse, the IMIEO's slope downward with later oppositions (as compared with total ΔV, Figure 2.4-2). On longer missions, the larger mass requirements (i.e., those for radiation and meteoroid shielding, life support, propellant boil-off, systems redundancy, etc.) reduce the IMIEO advantage of the long missions (as compared with total ΔV); however, these missions are superior in 1980 by a factor of approximately 2; in 1986, they are approximately equal.

Minimum IMIEO launch dates are displaced from the minimum ΔV launch dates by from zero to 107 days. The corresponding IMIEO values generally differ less than 6%. However, a notable exception to this similarity in IMIEO occurs for the 1978 opposition. For the long-short mission in that opposition period, IMIEO corresponding to minimum ΔV is 67% larger than the minimum IMIEO. This increase is due to an extremely poor ΔV distribution, which is very unequal with respect to large ΔV at Earth departure. This effect could be overcome by replacing the single Earth-departure stage with 2 stages. Moreover, an 18% difference exists for the 1975 long-short mission. In general, it can be stated that total mission ΔV is very useful, but as a replacement for IMIEO, it should be used with reservations.

Except for the opposition windows of 1975 and 1982, the long-short missions exhibit larger IMIEO values than those of the short-short missions. For the 1978 window, the short-short mission has a significantly lower IMIEO than that of comparable long-short missions, although total ΔV requirements are significantly lower for the long-short. Primarily, this effect is the result of a poor ΔV distribution on the long-short mission, although increased requirements in systems mass are also a contributing factor.

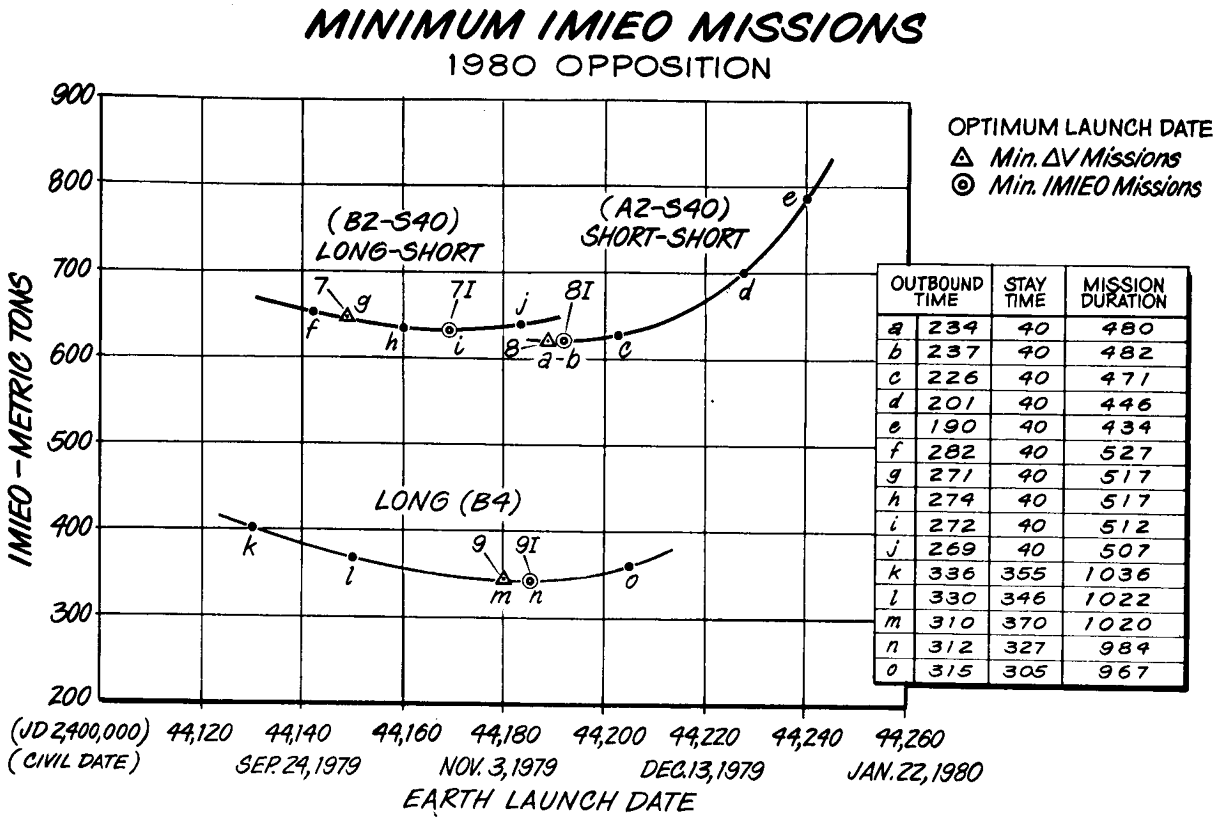
Figures 7.3-2 through 7.3-4 contain coverage in more detail of the 1975, 1980, and 1986 opposition periods. These missions are completely

MINIMUM IMIEO MISSIONS - 1975 OPPOSITION



	OUTBOUND TIME	STAY TIME	MISSION DURATION
a	182	40	458
b	192	40	467
c	189	40	461
d	184	40	453
e	180	40	449
f	235	40	572
g	236	40	561
h	240	40	554
i	239	40	550
j	241	40	549
k	242	40	541
l	351	336	997
m	350	340	980
n	351	336	977
o	358	330	998
P	243	424	980

FIGURE 7.3-2

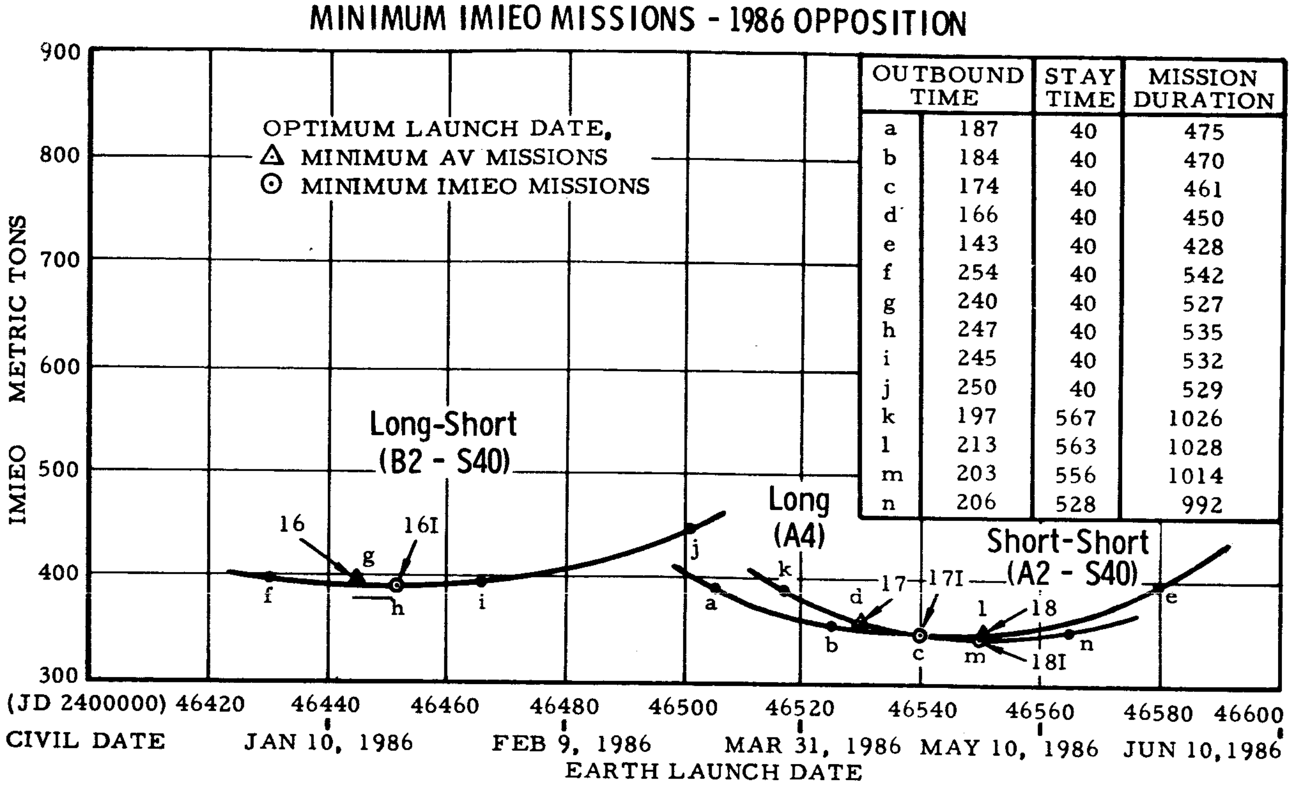


	OUTBOUND TIME	STAY TIME	MISSION DURATION
a	234	40	480
b	237	40	482
c	226	40	471
d	201	40	446
e	190	40	434
f	282	40	527
g	271	40	517
h	274	40	517
i	272	40	512
j	269	40	507
k	336	355	1036
l	330	346	1022
m	310	370	1020
n	312	327	984
o	315	305	967

FIGURE 7.3-3

defined by specification of Earth launch date, mission duration, stay-time, and outbound time. All of the IMIEO values presented above correspond to point-optimized missions. That is, each IMIEO point represents not only a specific mission (Earth launch date, mission duration, stay-time, and outbound time), but a specific vehicle which will be used in accomplishing this mission only, even if other vehicles have the same IMIEO. This general approach was considered appropriate for this study; however, future investigations should be made of vehicles which will operate over a range of Earth launch dates, both within a single window and within several windows. In a point check on the magnitude of this problem, a vehicle was defined for a 40-day window on the 1986 long-short mission. The IMIEO was 12% greater than that of the minimum IMIEO point, but this increase is excessive, since the approach used was a non-optimum one.

It is recommended that other types of missions be investigated in future studies. (The details of an intermediate time mission are discussed in brief in subsection 2.7.1.)



	OUTBOUND TIME	STAY TIME	MISSION DURATION
a	187	40	475
b	184	40	470
c	174	40	461
d	166	40	450
e	143	40	428
f	254	40	542
g	240	40	527
h	247	40	535
i	245	40	532
j	250	40	529
k	197	567	1026
l	213	563	1028
m	203	556	1014
n	206	528	992

FIGURE 7.3-4

7.4 IMIEO SENSITIVITIES AND VARIATIONS

Initial mass sensitivities, which are partial derivatives of IMIEO with respect to mission and vehicle parameters, were computed for the optimum launch-date points of each of the 3 types of missions in both the 1980 and 1986 opposition periods. Wide variations in IMIEO with respect to selected parameters were also developed. In every case, all of the independent variables that were not being arbitrarily varied for illustration were optimized. The data which are illustrated in Figures 7.4-1 through 7.4-3, are useful in extending the nominal minimum IMIEO results to off-nominal conditions, and for indicating areas of potential vehicle improvement (i.e., IMIEO-sensitivity parameters). In the subsequent analyses of probability of mission success and program success, direct use was made of these data.

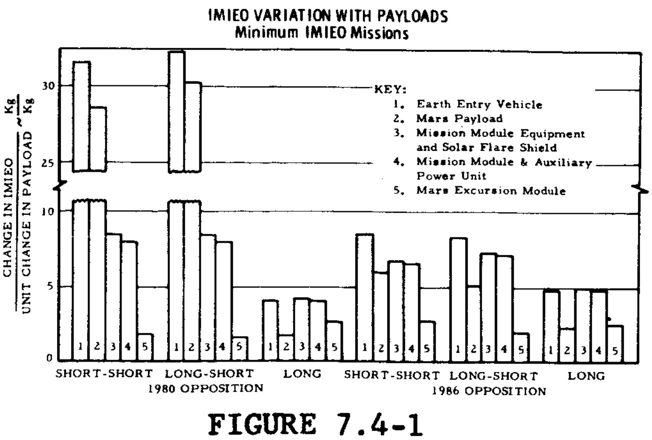


FIGURE 7.4-1

As might be expected, IMIEO was found to be more sensitive to specific impulse, particularly in the case of the nuclear engines, than to any other parameter.

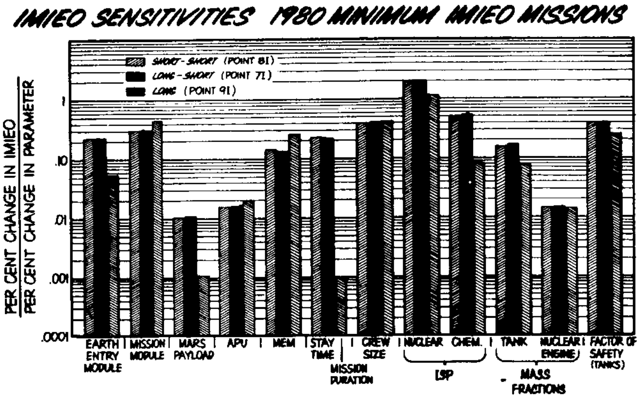


FIGURE 7.4-2

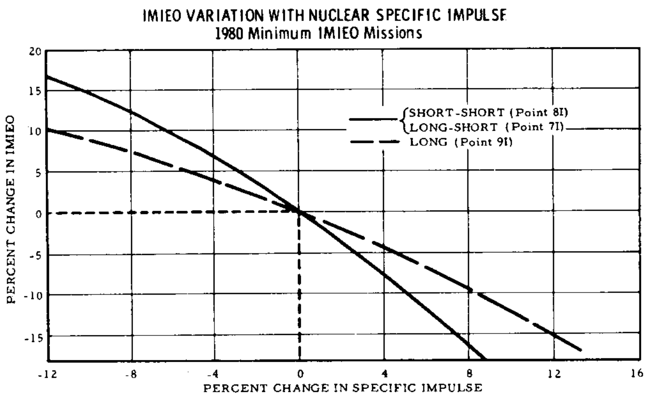


FIGURE 7.4-3

The partial derivatives of IMIEO with respect to payloads in the stages of Mars-braking, inbound mid-course, and Earth-braking were found to be essentially linear within the ranges investigated, even though mission re-optimizations were involved. Although the partials are associated with specific payloads, they may be applied to other mass staged independently at the same point on the trajectory. Partial derivatives of IMIEO with respect to solar flare shielding and equipment mass within the mission module are higher than those for the module itself because of the associated changes in structural mass.

IMIEO changes with respect to mission and other vehicle parameters were found to be non-linear. In the investigation of these changes, perturbations were made in parameters such as crew size, specific impulse (nuclear and chemical), stay-time, mission duration, Earth-entry velocity, mass fractions (engine and tank), and the factor of safety (tank).

Non-dimensional sensitivites (i.e., percent IMIEO per percent variable) were computed for overall comparisons. These sensitivities are shown in Figure 7.4-2 for the 1980 launch window.

7.5 IMIEO DATA FOR ALTERNATE MISSION/VEHICLE DEFINITIONS

In addition to the parameter variations of the previous section, IMIEO values were determined for a number of alternate mission/vehicle definitions. The results correspond to the values associated with the minimum IMIEO, optimum launch date, 1980 short-short mission (point 81). During the study re-optimizations were conducted where possible on the independent variables.

In general, these computations were less detailed than those used in the nominal definition work; therefore, they should be regarded as approximate comparisons only. In Table 7.5-1, the various definitions are delineated. The corresponding IMIEO values are presented in Figure 7.5-1. An IMIEO value for the 1980 long-mission (Point 9I) is included to provide a comparison.

The use of an unmanned vehicle to boost the Mars escape stage into Mars orbit is planned in the Convoy concept. Mating of the manned vehicle is then accomplished prior to Mars departure. Included in this operation are (1) the mission module, (2) the Earth-entry vehicle, (3) the inbound mid-course propulsion, and (4) the Earth-arrival propulsion. The total IMIEO for the 2 vehicles is about 6% greater

Table 7.5-1

1980 Short-Short*, Minimum IMIEO Mission/Vehicle Definitions

CONCEPT NUMBER & CONCEPT	DEFINITION
1. NOMINAL	1980 Short-Short Minimum IMIEO Mission (Point 8I); Nuclear earth escape, Mars braking, & Mars escape stages; chemical (Isp = 500) earth braking & mid-course guidance stages; one engine per stage (no re-start); earth entry velocity of 15.2 Km/sec (50,000 fps)
2. LONG MISSION	Nominal vehicle defined for 1980 long, minimum IMIEO mission
3. ALL NUCLEAR	Last chemical stage on nominal vehicle replaced by nuclear stage
4. CLUSTERED ENGINES	4 earth escape engines, 2 Mars braking engines, 1 Mars escape engine
5. NUCLEAR RE-START	Retain & re-start nuclear engines
6. MARS ATMOSPHERIC BRAKING	Atmospheric braking at Mars, chemical Mars orbit circulation stage (ΔV = 0.165 Km/sec)
7. CONVOY (MANNED-UNMANNED)	Mars escape stage boosted into Mars orbit and mated to manned vehicle which is boosted separately
8. EARTH ENTRY: 11.0 Km/sec (36,000 fps)	Maximum allowable earth entry at Apollo conditions
9. EARTH ENTRY: 11.0 Km/sec (36,000 fps)/STORABLE LAST STAGE	Maximum allowable earth entry velocity reduced to Apollo conditions, storable last stage (Isp = 376), no solar flare shield (last stage propellant acts as shield)
10. EARTH ENTRY: 20.7 Km/sec (68,000 fps)	Maximum allowable earth entry velocity increased, no earth braking propulsion required
11. ALL CHEMICAL	All chemical propulsion, Isp = 500
12. ARTIFICIAL GRAVITY	Provisions for 0.35 g's artificial gravity
13. MULTI-NOMINAL	Two vehicles, each with a 6 man crew and capable of completing mission with crew of disabled buddy vehicle
14. NUCLEAR PULSE	Single constant thrust nuclear pulse engine staged after Mars escape
15. ENGINE OUT	2 Earth escape engines, one engine out, initial acceleration 0.15 g's
16. TANKS LAUNCHED UN-FILLED	Propellant tanks filled in Earth orbit
17. TANKS LAUNCHED UN-FILLED/NO METEOROID PROTECTION	Propellant tanks filled in Earth orbit, no meteoroid protection for propellant tanks

*Except concept Number 2

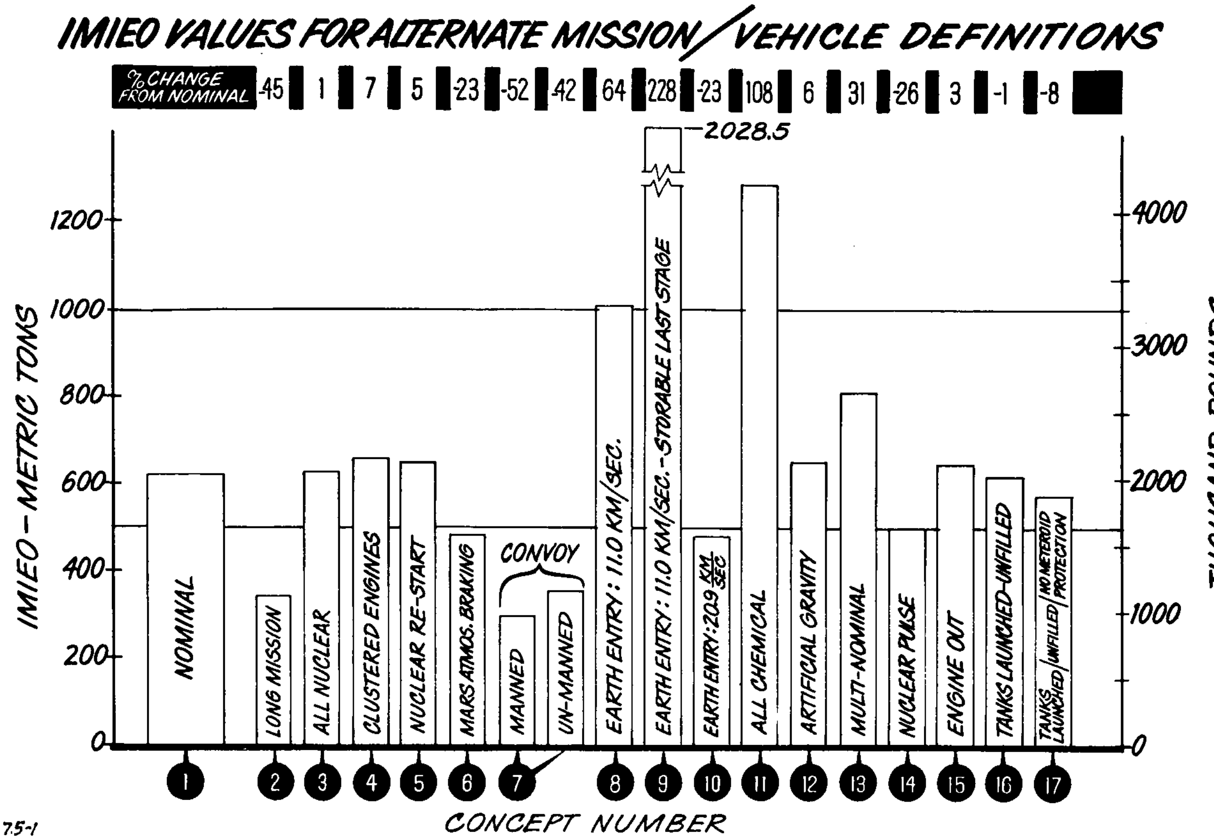


FIGURE 7.5-1

than that of the Nominal vehicle. Individually, the IMIEO of each vehicle is less than that of the Nominal one.

Two vehicles of Nominal configuration, either of them capable of performing the mission with a 12-man crew, are employed in the Multi-Nominal concept. In the event of an emergency, the 6-man crew of the disabled vehicle can transfer to the Multi-Nominal vehicle and complete the mission.

In reducing the maximum allowable Earth-entry velocity to Apollo conditions (11.0 km/sec), a tremendous IMIEO penalty was produced because of the resultant large propulsive requirement. This penalty, however, was expected. The substitution of a storable last stage of lower specific impulse (376 sec) doubled the Apollo-entry IMIEO, even though the propellant carried was assumed to replace the polyethylene in the storm cellar.

IMIEO was significantly less for the Nuclear Pulse vehicle, based on a constant-thrust engine staged after Mars escape. The source of Nuclear-Pulse performance data is referenced in the Propulsion Systems section of the Technical Report.

The chemical propulsion used for the chemical engines on the nominal vehicle (LO_2/LH_2-Be, Isp=500 sec) was substituted for the

nuclear stages of the nominal vehicle, and an IMIEO increase of approximately 100% was produced. This increase is not as large as might be expected because of a high chemical Isp, the elimination of propulsion shielding, and the smaller tank fractions for the LOX. The All-Chemical concept appears to be considerably more attractive from an IMIEO standpoint in 1986 because the IMIEO is only 65% higher than that of the Nominal vehicle for the short-short mission. (This effect is not illustrated herein.)

The Nominal concept was based on the use of one engine per stage; the Clustered Engine concept provided a more realistic vehicle for comparison. Nuclear restart IMIEO was higher because of the cool-down propellant requirement (assumed to be 5%). The Earth entry IMIEO, 20.7 km/sec (68,000 fps), is, of course, significantly less. Obviously, the use of artificial gravity results in a penalty. In the case of Tanks Launched Unfilled and Tanks Launched Unfilled/No Meteoroid Protection, reduced IMIEO values are shown for purposes of comparison. The value for Engine-out (1 out of 2 on the first stage) corresponds to that of increased reliability because all propellant can be used in the remaining engines; the IMIEO penalty results from the clustered engines and increased gravity loss.

7.6 BOOSTER/SPACECRAFT COMPATIBILITY

A booster/spacecraft compatibility study was conducted to provide a basis for the determination of the costs of Earth launch and orbital assembly. These costs are associated with a number of selected Mars-mission configurations. Each of the selected configurations was investigated to determine the minimum number of launch vehicles required to provide orbital delivery through the exclusive use of either the Saturn V booster or a Post-Saturn-type booster (or combinations thereof). For the purposes of this study, the net payload launch capability of the abovementioned launch vehicles was assumed to be 113.4 metric tons (250,000 lbs) and 453.6 metric tons (1,000,000 lbs), respectively. In addition to investigations of launch requirements, each configuration was examined to ascertain the number of major units for which assembly in orbit would be required.

Figure 7.6-1 is an illustration of the booster and orbital-assembly requirements for a Nominal 1980 short-short class, minimum-IMIEO Mission. In this case, the Saturn V requirements were examined for a condition in which both payload mass and payload volume were limited, and a condition in which payload mass was the only limiting factor. The Post-Saturn arrangement was limited by payload mass and volume.

BOOSTER/SPACECRAFT COMPATIBILITY

NOMINAL 1980 SHORT-SHORT CLASS, MINIMUM IMIEO MISSION

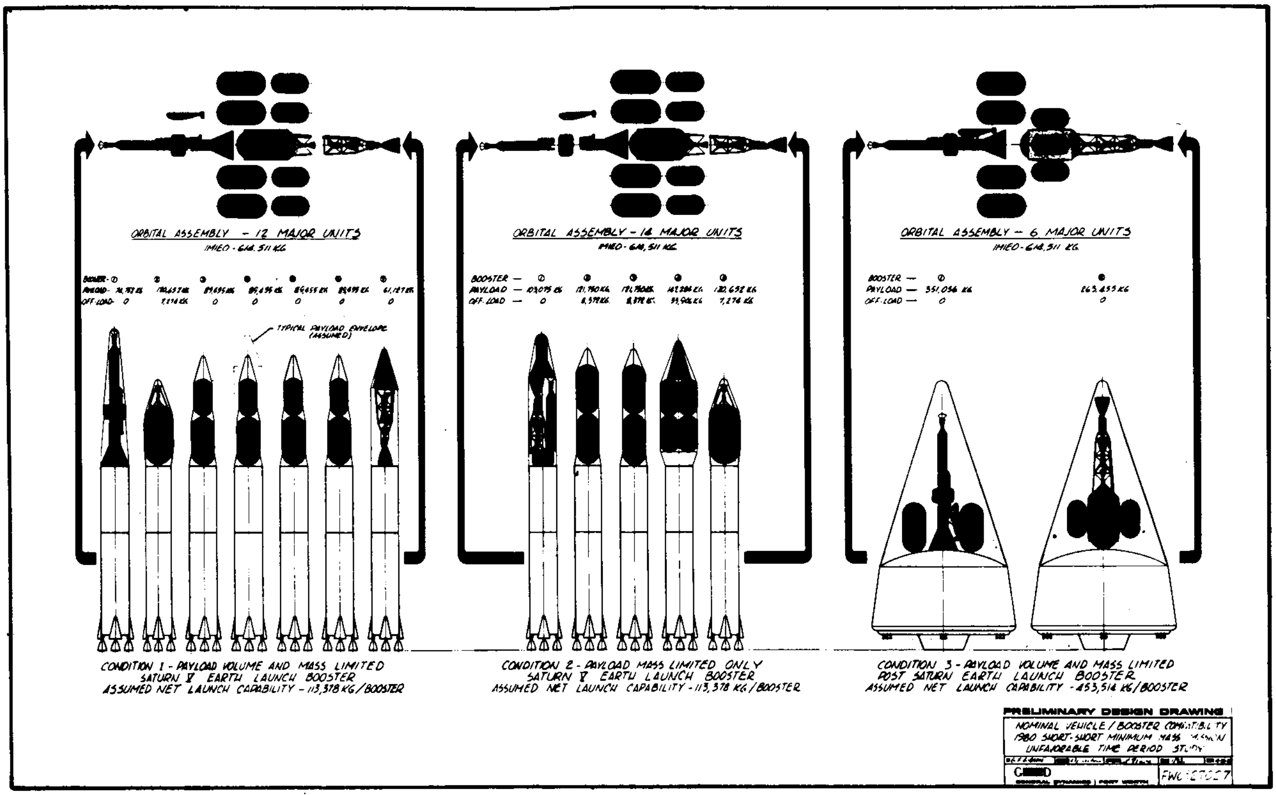


FIGURE 7.6-1

OPERATIONS RESEARCH STUDIES

8.0 SYSTEMS COST ANALYSIS AND DEVELOPMENT

8.1 GENERAL

In this portion of the study, the cost estimates for the space vehicle were determined. These costs were combined with the lofting and operations cost in Section 11.0 to obtain total program costs. Basic conclusions concerning the costs of various programs are presented in Section 11.0. Specifications of the physical and performance requirements relating to the missions under study were obtained from analyses of payload requirements. Through the use of functional relationships of cost estimating, the systems development and procurement costs were determined.

8.2 PROCEDURES

In establishing an analytical framework for the cost analysis, a number of criteria were adhered to which relate items such as systems design, development, and hardware costs to the expected state-of-the-art relevant to the time period of a mission. It was anticipated in the Mars mission (1) that all applicable existing facilities and other resources developed for prior missions would be utilized, (2) that maximum use would be made of previously developed systems, (3) and that hardware and unit production costs would tend to diminish as production rates increase.

In the use of the cost estimating relationships, cost was computed as a function of typical variables such as mass, thrust, specific impulse, and reliability. These cost estimating relationships were derived for each major system from statistical and empirical sources. The costs for space system R&D, production, and assembly were obtained with functional relationships, some of which were derived by General Dynamics/Fort Worth in an independent research program. These predictive equations were developed on the system level rather than on a more aggregate level in order to provide greater flexibility and accuracy in the cost analysis. Illustrated in Figure 8.2-1 is the cost estimating relationship used for pressurized chemical propulsion development cost. Depending on the thrust level and reliability objectives selected, development costs are expected to range between $20 million and $120 million. Of the 2 functional parameters, reliability accounts for more variation in costs than

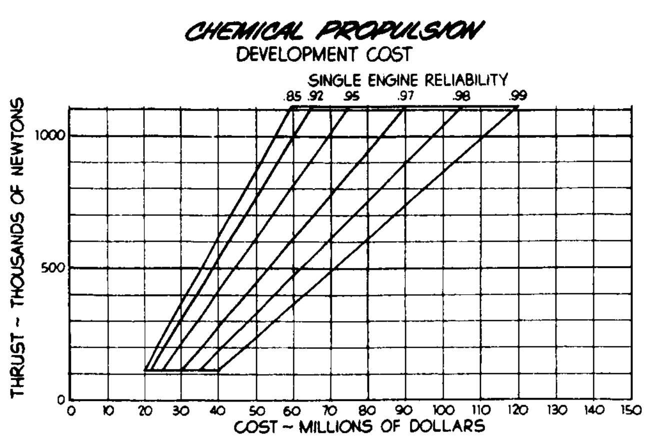


FIGURE 8.2-1

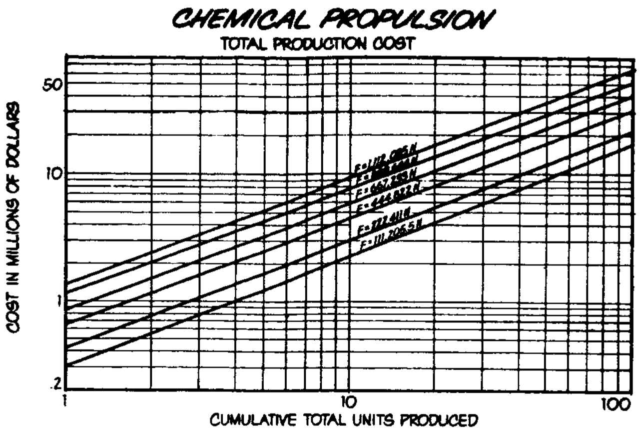


FIGURE 8.2-2

thrust level. Production costs for chemical propulsion systems are shown in Figure 8.2-2. The estimating relationship excludes the cost of tanks, since it was estimated separately. A 90% learning curve was assumed for all of the thrust levels shown. A separate treatment such as this of systems is especially desirable since the variation in the required state-of-the-art advance among systems creates a broad spread in the level of R&D.

The costs for life-support system R&D were estimated as a function of mission time, crew size, and the initial quantity of stored potable water. The graphite-core nuclear engine R&D costs for the Mars mission were functionally estimated on the basis of thrust levels and reliability. It was assumed that these engines would be second generation engines and that during the NERVA program, a substantial amount of successful R&D work would be accomplished. The development costs of propellant tanks were found to correlate with tank mass and mass fraction. These parameters also appear to account for variations in propellant combination and other design considerations. Propellant tanks production costs were estimated as a function of mass, mass fraction, and reliability.

The systems for life-support, nuclear engines, propellant tanks, and instrumentation were found to be the most expensive individual systems required for the Mars mission. Each system's cost estimating relationship was determined independently, with the exception of instrumentation, which was estimated as a function of the cost and relative complexity of all other payload systems. Cost relationships for R&D and production charges were also developed for the following: guidance and navigation; attitude control; mid-course correction; data processing; auxiliary power units; module structures; the "storm cellar"; communications; and scientific and experimental equipment. Estimates were made of the spares required to provide operational reliability for each system.

8.3 PARTITION OF PROGRAM COSTS

The largest general cost category for the Mars mission is in systems R&D, which represents approximately one-half of the total cost, excluding lofting and orbital assembly. The remainder of the costs include those for contractor support, test hardware, and operational vehicles. The estimates made for contractor support costs were based on NASA experience with Mercury and Gemini, and projections for Apollo. Test hardware costs were determined by constructing a

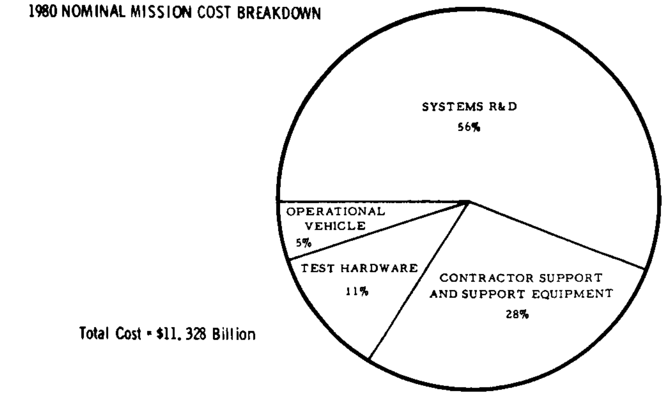


FIGURE 8.3-1

mission development plan which detailed in sequence all test hardware requirements, and the production cost of this hardware was estimated through the cost estimating functional relationships described earlier. Operational vehicle costs include the costs of 2 complete vehicles with required spares. The relative magnitude of these cost categories are shown for the 1980 Nominal mission in Figure 8.3-1.

8.4 RESULTS

Detailed estimates were made for total mission costs for the variety of mission configurations considered in the performance analysis. Differences in mission costs were noted whenever parameters such as launch year, number of crew members, Mars stay-time, propulsion variations, and artificial gravity were systematically varied. An example of the results is presented in Figure 8.4-1 where long and short-short missions are compared by calendar year. It will be noted that the long missions are more expensive than the short, although IMIEO for the long missions is much less. In Figure 8.4-2, a comparison is given of the effect that crew size has on a Nominal mission cost. The marginal or added cost per crew member is approximately $500 million. Differences in systems cost associated with stay-time on Mars are shown in Figure 8.4-3. These costs were computed for variable stay-times for a 1980, 6-man short mission, with a Mars landing crew of 3 men. An extension of stay-time from 20 to 60 days on Mars would cost approximately $800 million.

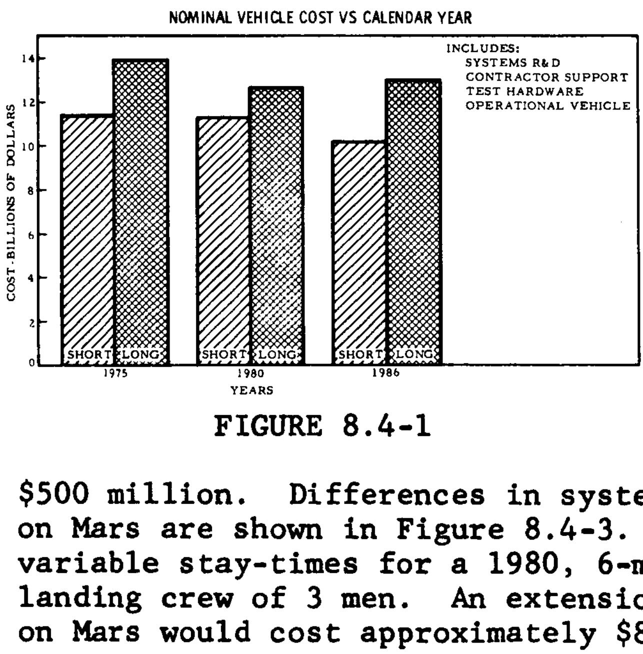


FIGURE 8.4-1

Mission costs were compared for the 1980 Nominal Vehicle (with varying Isp), an All-Chemical system, an All-Nuclear system, Clustered Engines, and Nuclear Engine Restart. The resulting cost variations were rather insignificant and reflected the change in cost with respect to vehicle and engine size more than the changes in engine development costs. Primarily, these slight variations are due to the fact that information is not available at this time which would allow the development of cost relationships which are sensitive enough to account

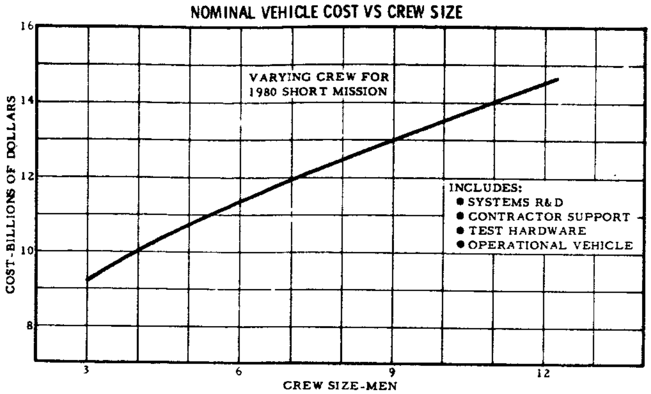


FIGURE 8.4-2

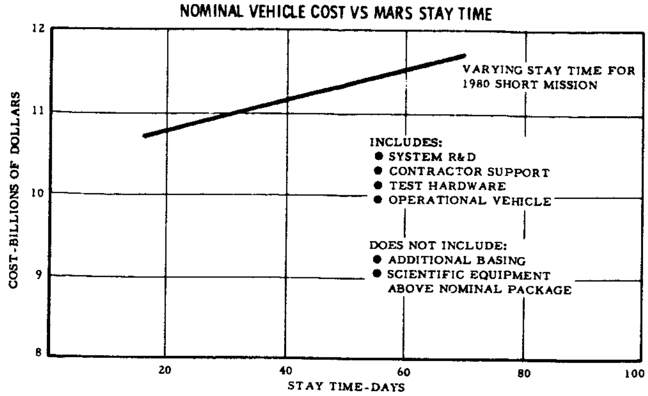


FIGURE 8.4-3

for improvements in engine parameters such as Isp. The problem associated with this lack of sensitivity is resolved in the risk analysis, however.

Systems costs for variations of the 1980 Nominal Vehicle are shown in Figure 8.4-4. Systems costs rise by $842 million for the 12-man Multi-Nominal concept over those for the 12-man Nominal Vehicle. This cost increase is due primarily to the purchase of a second operational vehicle. The other vehicle variations do not differ radically from the 6-man Nominal. Provision for artificial gravity adds $420 million to the mission cost; Convoy adds $625 million; removal of meteoroid protection and the use of partially filled tanks adds $739 million because of the additional tanks required; and the use of Mars atmospheric braking increases mission expense by $1,083 million over that for the Nominal vehicle as a result of increased structural costs.

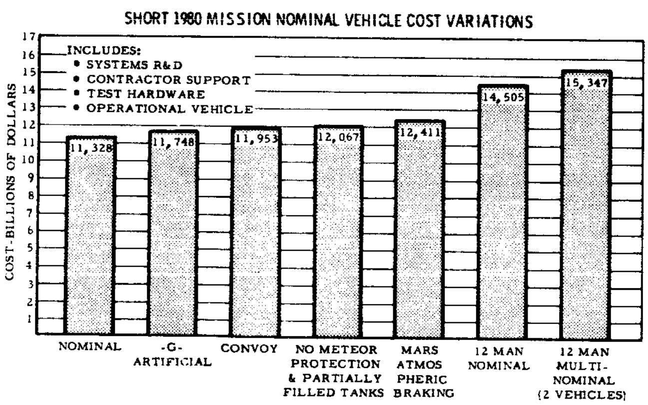


FIGURE 8.4-4

9.0 EFFECTIVENESS AND SENSITIVITY ANALYSIS

9.1 INITIAL EFFECTIVENESS ANALYSIS

Results of the initial effectiveness analysis enabled a preliminary evaluation of mission concepts and a concomitant definition of the scope of the task needed for a detailed appraisal of the more promising concepts. At this point, the evaluation parameters consisted of calendar year, IMIEO, and probability of mission success (Pms). Pms is defined as the likelihood that the essential systems will perform their design functions and that all necessary operations will be executed according to plan. Figure 9.1-1 contains the initial effectiveness plot for the Nominal Vehicle. Pms can be increased with relative ease, up to around 90%, by increasing the complement of low mass spares and redundancies. Advances above this point become expensive in terms of IMIEO since duplications of entire systems or high mass redundancies are required. The change in IMIEO required to increase Pms from 69.4% to 87.6% for the long mission is approximately equal to that required for the short missions although the actual equipment mass required for the long missions is significantly greater. This approximate equality is due to the fact that the long missions are less sensitive to changes in payload mass than the short missions.

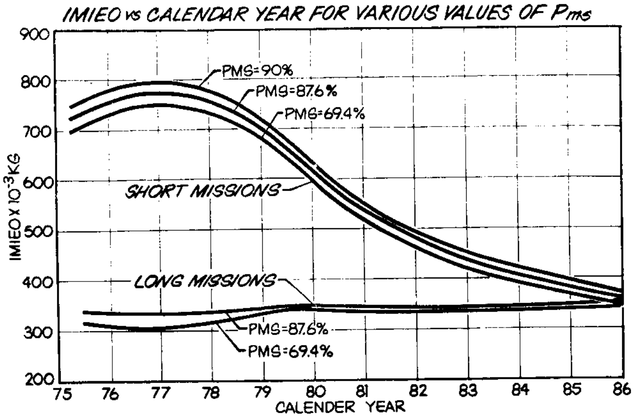


FIGURE 9.1-1

Because of the demands for IMIEO on the early short missions, the majority of the sensitivity was confined to post-1980 missions. It must be pointed out that, should a significant increase in system performance be obtained above the estimates made in this study, this present conclusion could be revised.

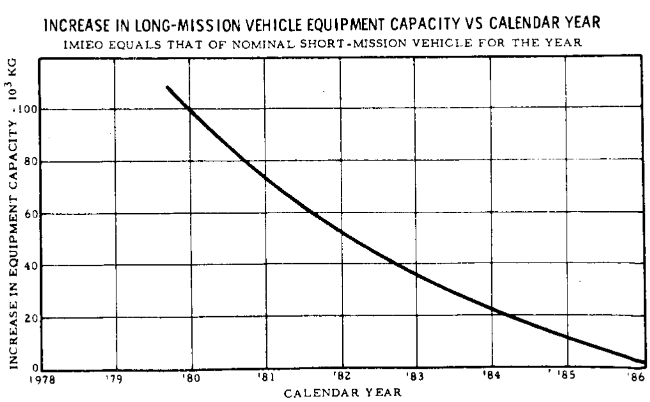


FIGURE 9.1-2

The IMIEO's shown in Figure 9.1-1 for the long mission include all necessary life support provisions, but they have not been adjusted to include extra scientific equipment which would probably be desirable in order to take full advantage of the longer stay-times. The amount of such

equipment which could be taken to the Mars surface with the IMIEO differential gained by the long mission relative to the short mission is shown in Figure 9.1-2. It is noted that this differential capability ranges from 108 x 10^3 kg for the 1980 mission to 2 x 10^3 kg in 1986. This is equivalent to 36 and 0.6 kg of equipment per man per day, respectively.

9.2 RADIATION PROTECTION

The IMIEO penalty for portage of the storm cellar is shown in Figure 9.2-1. It is noted that constraint of the whole body dose to 150 rad instead of 200 rad is dependent on a relatively small increase in IMIEO. Metabolic water was utilized as shielding in this illustration. A crossover in IMIEO between long and short missions occurs around 1984.

To simplify the amount of calculation, IMIEO computations were made by using the solar flare model discussed in Section 3.3. To account for variations in flare occurrence, other than calendar-year effects, a statistical analysis was made to establish the confidence which can be placed in the adequacy of the shielding provided to meet expected dose level constraints. These results are shown in Figure 9.2-2. It is noted that confidence rises rather sharply with dosage and a high level of confidence can be obtained with a reasonable crew dosage, particularly for the 150 rad constraint design.

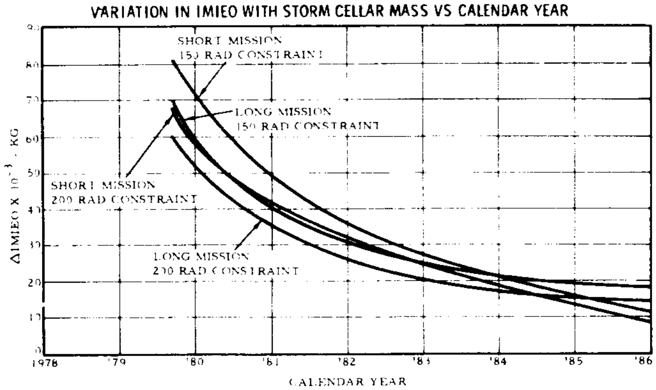


FIGURE 9.2-1

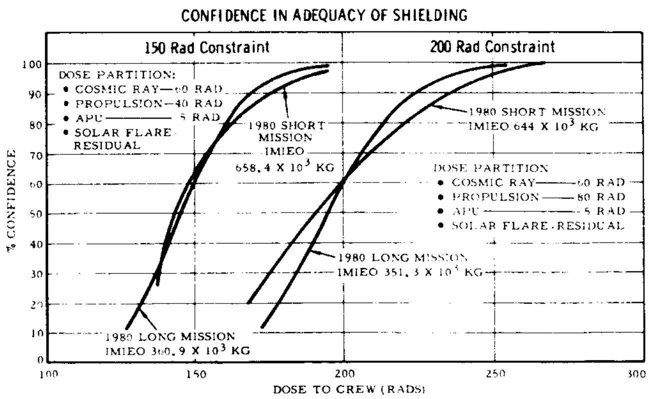


FIGURE 9.2-2

The crew dosage shown in the figure is based upon a quasi-optimum partition between on-board and extravehicular sources. It is assumed that a perfected flare warning system will be available and that all necessary vehicle functions can be performed in the storm cellar during periods of high solar activity.

9.3 SENSITIVITY OF MISSION EFFECTIVENESS PARAMETERS

The required IMIEO for various crew sizes is shown in Figure 9.3-1. As indicated, the "unfavorable time period" is further accentuated by larger crew sizes. In this comparison, the size of the Mars landing party and provisions therefor were not varied with crew size. The sensitivity of IMIEO to changes in Mars stay-time from the nominal 40 days is evident in Figure 9.3-2.

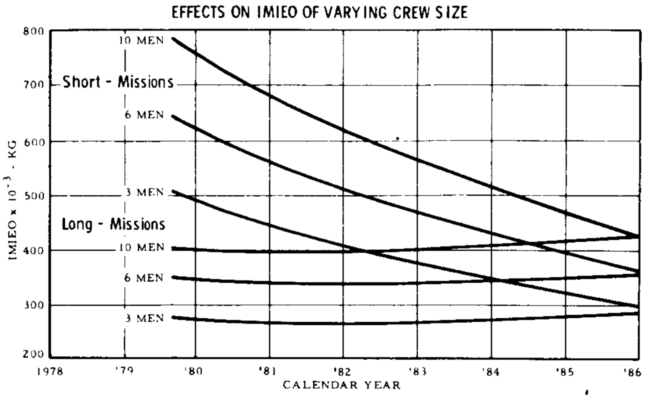


FIGURE 9.3-1

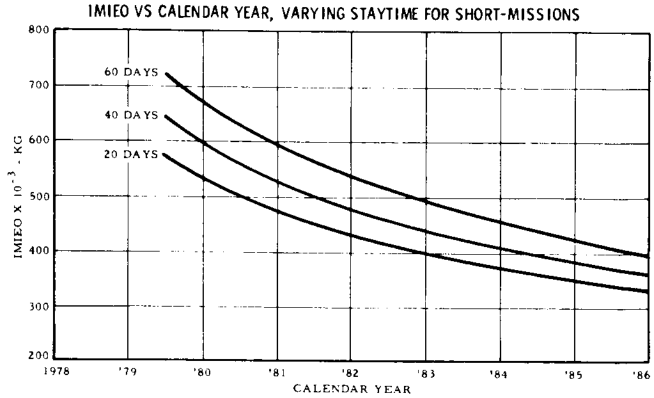


FIGURE 9.3-2

The IMIEO requirements for various Mars payload and sample return packages for the 1980 mission are obtainable from the nomograph in Figure 9.3-3. The Nominal Vehicle configuration is shown by the lower line while the upper line illustrates an alternate arrangement with a subsequent increase in IMIEO. Although no attempt was made in the present analysis to define an optimum scientific "package" consistent with various mission objectives, a knowledge of the options which are afforded is essential for planning purposes.

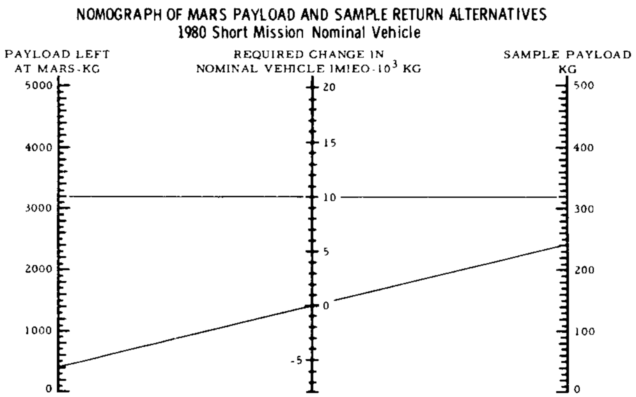


FIGURE 9.3-3

9.4 MULTI-NOMINAL VEHICLE

9.4.1 General. One rather obvious way of increasing the probability of mission success would be to provide 2 (or more) completely redundant vehicles for the mission. This type of mission was referred to in the study as the Multi-Nominal.

Two types of Multi-Nominal vehicles were considered, one incorporating provisions for a normal 3-man crew and an emergency 6-man

crew and the other incorporating provisions for a normal 6-man crew and a 12-man crew in an emergency. Comparisons were made with the Nominal vehicle.

9.4.2 Six-Man Vehicle. The broad band seen in Figure 9.4-1 emphasizes the influence of design in the Multi-Nominal system. The upper limit marks IMIEO for 2 of the Nominal-type vehicles with complete provisions for 6 men although each is carrying a crew complement of 3. The lower limit represents a minimum-crew, transferable configuration; in this configuration, stored life support provisions and the Earth reentry vehicle would be salvaged in the event of a disaster. The complete redundancies available in the former vehicle significantly enhance Pms.

9.4.3 Twelve-Man Vehicle. The greatest mission enhancement of the Multi-Nominal system is its maintenance of a high level probability of success when the system is threatened by the risk of an onboard disaster. Figure 9.4-2 contains a comparison of the 12-man Multi-Nominal system and a single 12-man vehicle. For example, if the results of the study indicate that the probability of disaster is 0.1 and the Pms is to be above 90 percent, then a Multi-Nominal system is required.

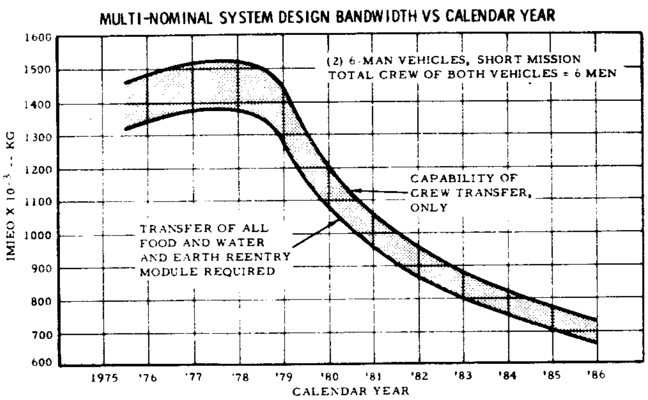


FIGURE 9.4-1

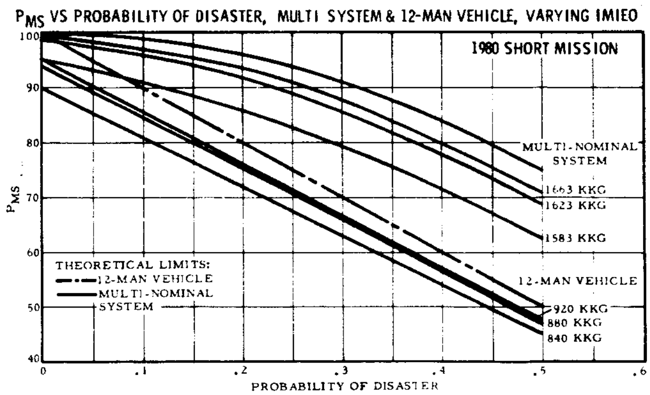


FIGURE 9.4-2

9.5 IMIEO SENSITIVITY TO PRIMARY PROPULSION

IMIEO sensitivity to the state of development of the primary propulsion system is shown in Figure 9.5-1. This data reflects the change in IMIEO which would result from arresting nuclear Isp at 720 seconds rather than adhering to the Isp growth expected in a continued graphite core engine development program. More significant is the increase in IMIEO, if all-chemical propulsion is used with an Isp of 500 seconds.

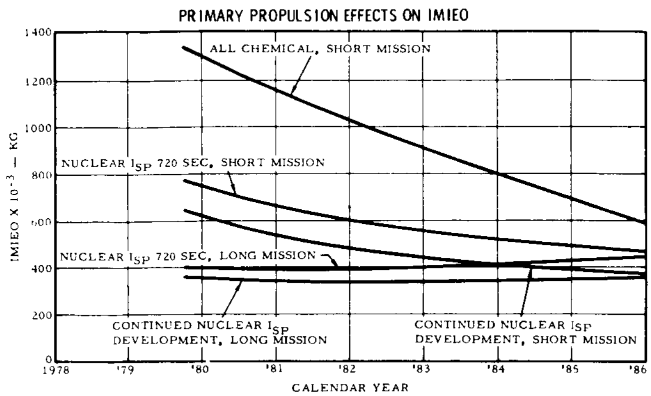


FIGURE 9.5-1

10.0 ANALYSIS OF TECHNOLOGICAL RISK

10.1 GENERAL

The success of a program is predicated upon meeting design objectives and schedules. There is a technological risk associated with the fulfillment of these goals. The objective of the risk analysis, as developed in this study, was to place a measure on the possibility of failure for a development program to meet the goals of performance, reliability, and time (delivery schedule) with respect to the original program estimates.

There are 3 primary sources of risk: (1) the risk involved in meeting performance, Ψ_P, (2) the risk involved in obtaining the required reliability, Ψ_R, and (3) risk associated with meeting schedule, Ψ_S. These values of risk are determined from the technological advance sought and the original cost estimate and proposed schedule of the development program. Risk may be reduced by increasing costs and schedules.

Risk values are established as decimal fractions, and for the purpose of clarity, they may be referred to on an odds or chance basis. A risk of 0.5 would indicate that the program stood a 50-50 chance of success; whereas, a value of 0.1 risk would indicate that there were 9 chances in 10 that the program would succeed without an increase in the allocated funds and estimated schedule. The approach is presented in Figure 10.1-1. Within the framework of these relationships, trades were performed to minimize risk without increasing specified funds and time allocations.

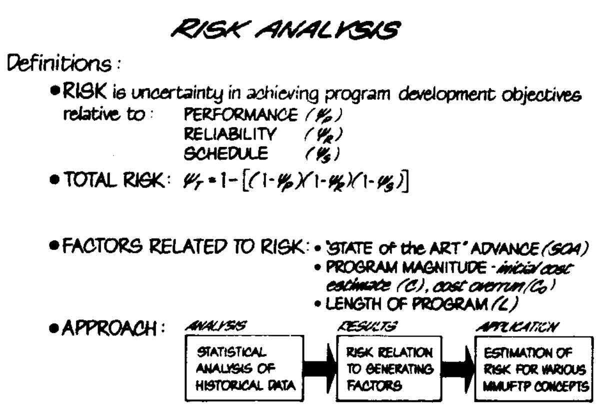


FIGURE 10.1-1

10.2 DATA SOURCES

The statistical data were collected from aircraft subsystem programs performed for GD/FW, Project Mercury and other space programs, and from similar analytical studies. The most significant of these are Marshall and Meckling's Rand report "Predictability of the Costs, Time, and Success of Development"; Peck and Scherer's <u>The Weapons Acquisition Process: An Economic Analysis</u> (Harvard University); Summers' Rand Report, "Cost Estimates as Predictions of Actual Weapons Costs: A Study of Major Hardware Articles"; and Bagley et al, "A Feasibility Study of Techniques for Measuring and Predicting the State of the Art," Batelle Memorial Institute.

10.3 DEVELOPMENT OF RISK

The following historical data were obtained for each development program used in the risk derivation:

SOA - State-of-the-art advance, objective/existing state-of-the-art, adjusted for position on development scale

C - Original cost estimate

L - Original delivery estimate

CO - Cost overrun, actual/estimated

SO - Schedule overrun, actual/estimated

PA - Performance attainment, actual/specified

RA - Reliability attainment, actual/specified

SA - Schedule attainment, estimated/actual

PA', RA', SA' - Predicted attainment values.

The attainment factors were plotted as a function of SOA, C, L, and CO (see Figure 10.3-1). A curvilinear regression analysis was used to fit the data. Risk is the variance associated with these prediction curves. The area under the risk distribution at any point along the curve, and limited by the 1.0 attainment line, is the risk associated with meeting the original objectives. These values of risk were plotted for various values of cost overrun, as well as the case of meeting the original cost estimate.

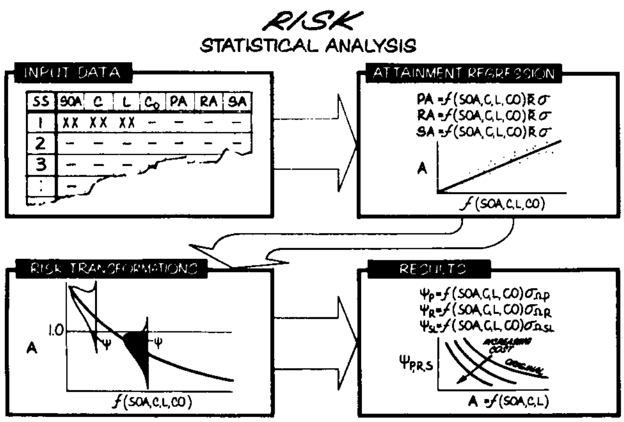


FIGURE 10.3-1

10.4 APPLICATION

In order to estimate risk, an estimate of the state-of-the-art advance (SOA), program magnitude (C), and length of the development program must be available. The values for SOA, L, and C are entered on the plots in Figure 10.4-1. The values attained from the plots are entered in the expressions contained in this figure to obtain values for PA', RA', and SA'. The attainment factors are entered on

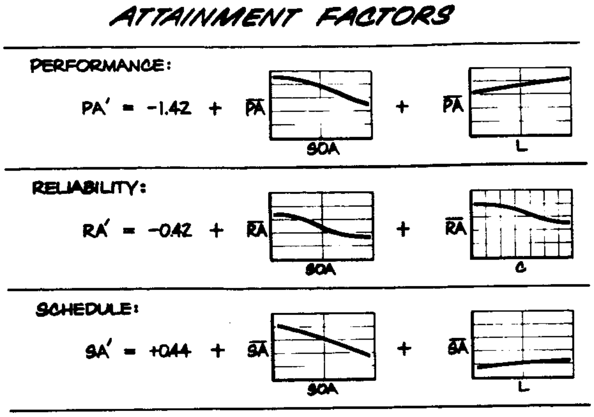


FIGURE 10.4-1

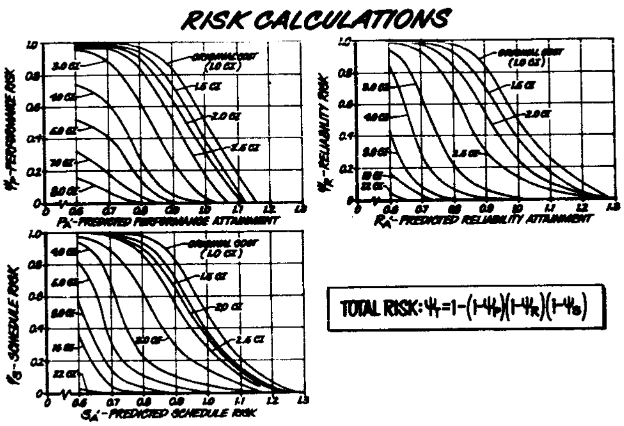


FIGURE 10.4-2

their respective risk curves (Figure 10.4-2). Along an iso-attainment line, risk may be attained as a function of multiples of the original cost estimate. A plot of risk versus cost overrun for the 3 types of risk is contained in Figure 10.4-3.

The nuclear engine with an Isp of 842 seconds is used as an example in this figure. The probability of success is one minus the total system risk and is defined as the likelihood of meeting all objectives. It is seen from this illustration that the risk associated with reliability achievement is the highest. At any level of cost overrun, the system risk (Ψ_T) is determined by $[1 - (1 - \Psi_P)(1 - \Psi_R)(1 - \Psi_S)]$. This procedure is followed for all of the development systems within a program.

The process used in determining total risk from systems risk is shown in Figure 10.4-4. Total risk was minimized with respect to program costs by the allocation of funding between systems. The systems risks were then summed by use of the expression shown in this figure. The final outcome is the total risk versus program cost for each concept.

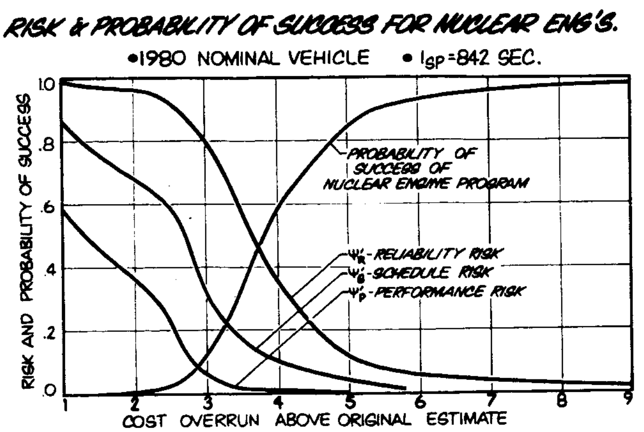


FIGURE 10.4-3

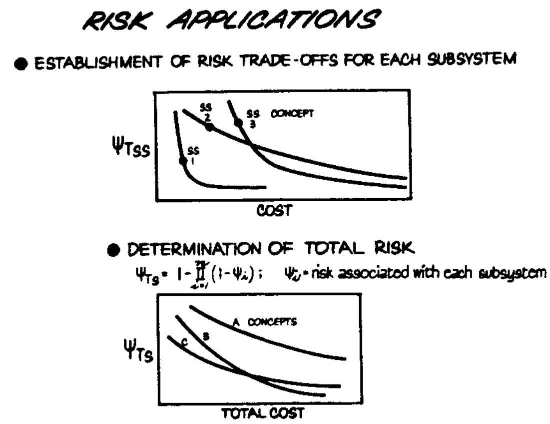


FIGURE 10.4-4

11.0 CONCEPT SELECTION AND RECOMMENDATIONS

11.1 GENERAL

The final phase of this study consisted of a critical evaluation and assessment of alternatives in order to establish the most promising mission concepts. A systematic review of concepts was made with respect to (1) probability of mission success, (2) program costs, (3) detailed sensitivity, and (4) development risks, as well as IMIEO. The most promising mission-class/vehicle-type was selected by systematically investigating the following on a basis as common to all of them as possible: (1) calendar-year and mission-duration effects, (2) propulsion system variations, (3) multiple vehicle types, (4) additional mission/vehicle modes, and (5) technological risk.

The probability of mission success (Pms) estimates consisted of the overall systems reliability, including spares, and the reliability of operations. The majority of the values assumed for operational reliability were obtained from the Ling-Temco-Vought Orbital Launch Operations study.

Program costs consisted of the total spacecraft cost, discussed in a previous section, and the lofting and orbital operations cost. At the present, it is not possible to define the precise technique and method which will be used for orbital assembly. However, enough detail is available to permit a meaningful cost analysis. Anticipating the availability at no cost of an orbital space station, the principal costs of orbital assembly are those of vehicles and launch operations used in transporting assembly and test personnel and their related supplies and equipment to the proximity of the space station. The number of lofting vehicles required was estimated as a function of Mars vehicle packaging and mission crew size. Cost estimates were made for 2 launch vehicles: (1) Saturn V, with an orbital payload capability of 110,000 kg, and (2) a post-Saturn-type vehicle, with a capability of 450,000 kg. Each Mars mission configuration was broken into packaging units suitable to the lofting capabilities of these vehicles. The General Dynamics/Astronautics Nova Cost Model was employed to estimate lofting and assembly costs. This model computed non-recurring and recurring costs from mission payload objectives, vehicle design parameters, reliability, and the number of prior launches. The post-Saturn cost estimate does not include the outlay of approximately $8.8 billion for the facilities and development required for this vehicle.

In the detailed sensitivity analysis, the effects of adding crew members, changing Mars stay-time, and other mission variables were factored in. Risks associated with selected concepts were evaluated by the procedures discussed in Section 10.0.

11.2 CALENDAR-YEAR AND MISSION-DURATION EFFECTS

In considering the effect of calendar year on cost for the minimum time/Nominal, minimum ΔV/Nominal, and minimum ΔV/Chemical classes of missions investigated, it was concluded that overall program costs for the same class of mission and vehicle-type showed a relatively small variation with calendar year. The maximum variation reflected in Figure 11.2-1 is on the order of 14%. The short missions were consistently lower in cost than the long missions, primarily because of the considerably lower life-support system costs.

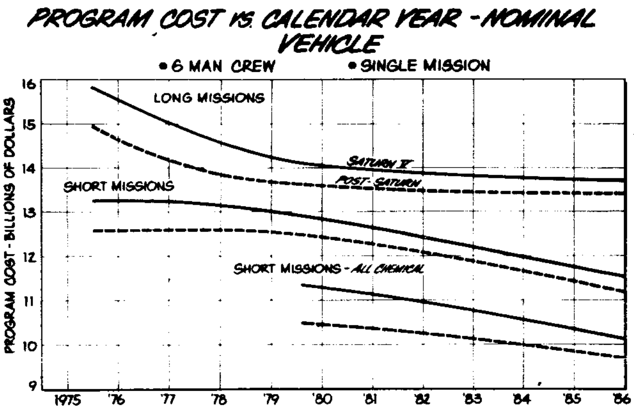


FIGURE 11.2-1

An apparent improvement in program costs of approximately \$2 billion would be realized by the use of an all-chemical propulsion system. This lower cost is a result of the less costly development program required for the Chemical system based on a single mission. On a more realistic multi-mission basis, the breakeven point for the Nuclear system with the Chemical system would occur at approximately 6 missions.

The savings in launch and assembly costs by utilizing the post-Saturn launch vehicle rather than the Saturn V varied with Mars vehicle configuration, but averaged approximately 40%.

Considered only on the basis of overall program costs, the short or minimum-time class missions appear to be the most attractive types and were selected for further evaluation. It was decided in the initial effectiveness analysis to eliminate all classes of missions prior to 1980 from detailed analysis because of the IMIEO penalties imposed to achieve a reasonably high Pms.

11.3 PROPULSION SYSTEM VARIATIONS

The effects of variations in the primary propulsion stages from the Nominal 3 nuclear and 1 chemical stages are indicated in Figure 11.3-1. The 1980 short-mission, minimum-IMIEO Nominal vehicle was used for comparison; this overall vehicle has a probability of mission success of 0.878. The various engine differences are presented in Table 7.5-1.

The Engine-Out vehicle has an additional engine as well as associated tankage and propellant on the first stage which would permit the mission to continue if one engine fails to operate, and this additional capability increases Pms. Nuclear Restart offers no

significant cost advantage and has a lower Pms than the Nominal vehicle. This effect is due to the decrease in engine reliability as a result of the increase in operating times imposed on the engines.

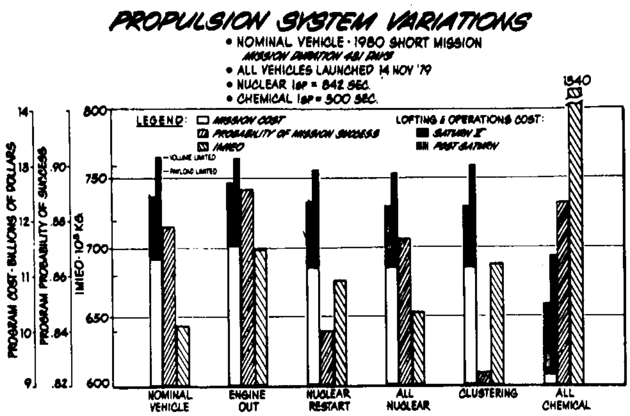


FIGURE 11.3-1

The All-Nuclear vehicle has a higher IMIEO than the Nominal, and a lower Pms because the inherent reliability of the nuclear fourth stage is lower than that of a corresponding chemical stage. The requirement that all engines must operate simultaneously decreased Pms for the clustered engine vehicle. The All-Chemical vehicle has a considerably lower overall program cost because of the relatively inexpensive engine development cost. It also has a higher Pms than the Nominal vehicle as a result of inherently higher engine reliability.

Propulsion system relative effectiveness, defined as Pms, divided by the overall program cost indicates that the Nominal, the Engine-Out, and the Chemical vehicles are superior to the All-Nuclear, Nuclear Restart, and Clustered engines concepts. Thus, it is tentatively concluded that the former are attractive for the Mars mission. The All-Chemical system is the most effective, based on this criteria. Again, the nuclear systems would appear to be the more effective on a multi-mission basis.

11.4 MULTIPLE VEHICLE TYPES

Figure 11.4-1 contains a comparison of the 6-man Nominal, the 6-man Convoy (2 vehicles), the 12-man Nominal, and the 12-man Multi-Nominal (2 vehicles) missions. These missions are defined in Table 7.5-1. The 2-vehicle Convoy has a low Pms of 0.825 which resulted primarily from the additional mating operations required in Mars orbit. Additional development costs also contributed to the rejection of the concept from further consideration.

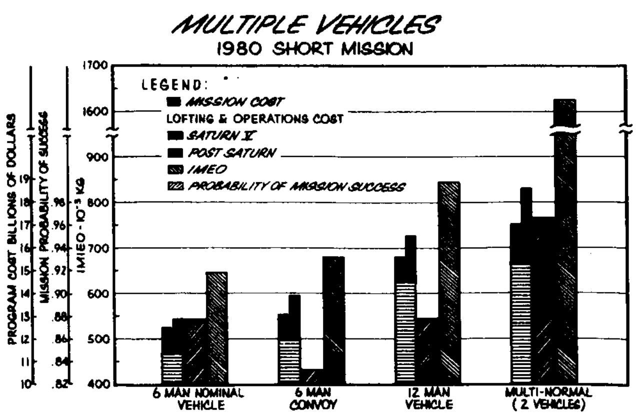


FIGURE 11.4-1

The Multi-Nominal concept consisted of 2 vehicles with a crew of 6 in each, with emergency provisions for a crew of 12. The high Pms (0.966) of this concept

is attributable to the provision for transferring the crew of one vehicle to the other in order to continue the mission if one vehicle becomes disabled. The second vehicle increases program costs approximately $2 billion.

The effective cost-per-man-day on Mars and the effective cost-per-unit-payload for the Multi-Nominal vehicle is 57% of that for the 12-man Nominal vehicles. The Multi-Nominal concept was selected as a promising one because of high Pms, favorable cost effectiveness, and flexibility.

11.5 ADDITIONAL MISSION/VEHICLE MODES

The additional mission/vehicle modes indicated in Figure 11.5-1 were investigated and compared with the 1980 nominal minimum IMIEO vehicle. The limited scope of these investigations did not allow Pms to be determined for each concept.

Of the 3 variations in entry velocity shown, only the all-atmospheric-braking-at-Earth mode appears to warrant additional investigation.

A significant reduction in IMIEO from the Nominal vehicle value was determined for the conical-shaped Mars atmospheric braking vehicle. The additional systems complexity, primarily in the structures system, resulted in a $1.3 billion increase in cost over that of the Nominal vehicle.

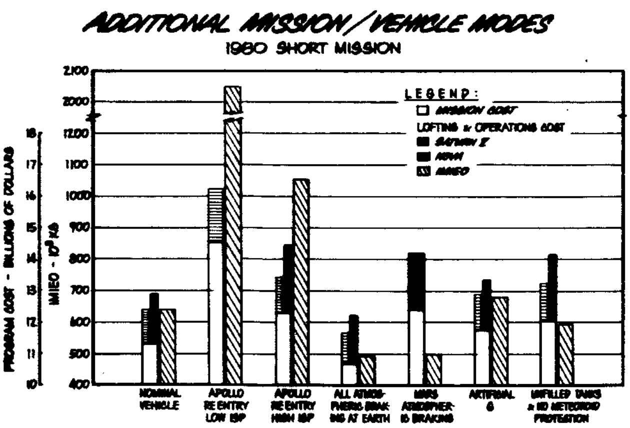


FIGURE 11.5-1

The inclusion of artificial gravity provisions on the Nominal vehicle increases overall program costs by less than 5.0%.

Although IMIEO of the Nominal vehicle was decreased by boosting the tanks unfueled and without meteoroid protection, the overall program costs were greater than the Nominal vehicle because of the large number of tankers required for orbital fueling.

It was concluded from this preliminary analysis that all aerodynamic braking at Earth and atmospheric braking at Mars warrant additional study.

11.6 CREW SIZE AND STAY-TIME

In the preceding discussion, crew sizes and stay-time were not varied in order to determine the sensitivity of other variables.

This section contains the results obtained by increasing the crew size or stay-time for the 1980 Nominal vehicle. The cost-per-crew-member ranges from \$3.5 billion for a crew of 3 to \$1.4 billion for a crew of 12. (See Figure 11.6-1.) Further reduction in cost-per-crew-member is expected to be gradual in a crew larger than 12 men. The cost-per-man-day on the Mars surface ranges from \$200 million to \$70 million as stay-time increases from 20 to 60 days. (See Figure 11.6-2.) If very long stay-times should become a primary consideration (approximately 300 days), the cost-per-man-day could be reduced to \$16 million by the use of the minimum ΔV class missions.

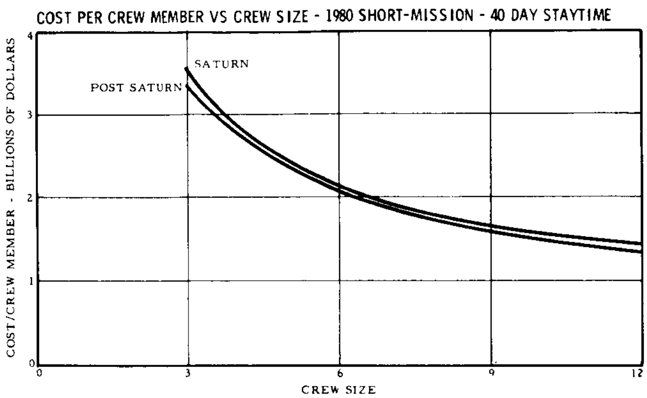


FIGURE 11.6-1

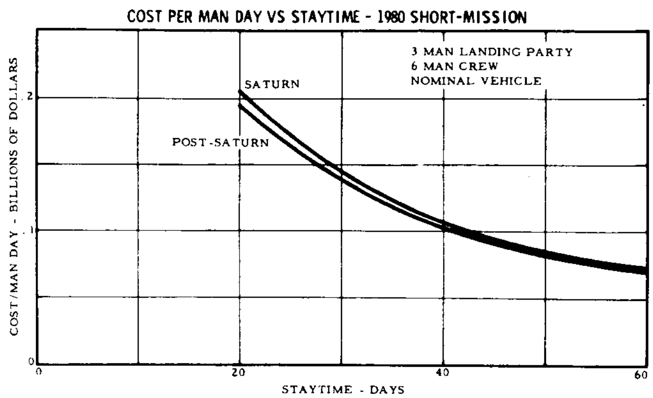


FIGURE 11.6-2

11.7 TECHNOLOGICAL RISK

The technological risk in the overall program is shown in Figure 11.7-1 as a function of program cost for the most promising mission class/vehicle types selected from the various concepts considered. This mission-vehicle combination was comprised of a post-1980 short mission and a Nominal or Multi-Nominal vehicle with either 4 chemical or 3 nuclear and 1 chemical main propulsion stages. In the provision for Engine-Out capability, approximately the same risk is involved as that in the Nominal system.

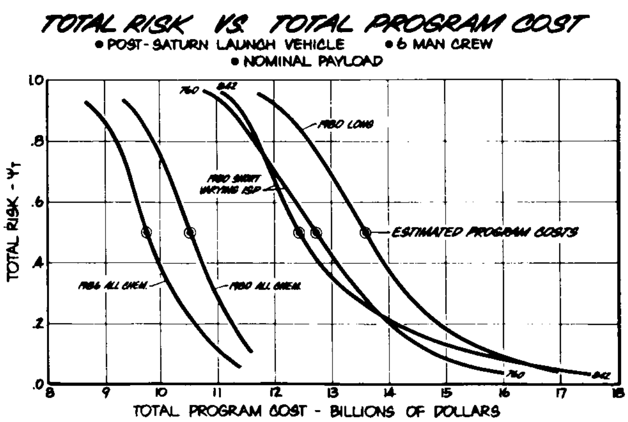


FIGURE 11.7-1

The cost relationships used in this study were based upon completed programs and adjusted current program estimates. Consequently, the costing relationships used were not as sensitive to the various system parameters and calendar-year effects as was originally anticipated. However, the resulting costs are reasonable, preliminary

estimates of overall program costs and were used as a basis for determining overall development risk. Since these total costs represent mean values based on past program experience, they are assumed to represent a 50% level of risk (i.e., chance of success is equal to chance of failure), since risk is a measurement of the uncertainty about these original estimates. If we define as successful a program which has an 80% chance of meeting its design objectives, then entering the risk curve at a level of 0.20 for the All-Chemical vehicle results in an overall probability of program success of 0.713 for a cost of $11.25 billion. On the same level of funding for the Nominal vehicle, a risk of 0.94 and a corresponding Pps of only 0.053 was obtained. To achieve the same risk for the Nominal vehicle as that for the All-Chemical (0.20), a program cost of $14 billion and a Pps of 0.703 would be incurred. Although the selection of an acceptable level of Pps is arbitrary, because of the limited amount of similar data available, a level of 0.70 would offer a reasonable likelihood of meeting planning, design, and mission objectives.

It will be noted in Figure 11.7-1 that the variation in nuclear Isp has an apparent negligible effect on program costs for the same level of risk. This is due in part to the compensating effect that tankage development costs have on the overall program costs.

In summary, it should again be emphasized that the data shown are applicable only to the selected class of missions and vehicles, and that they were generated on the basis of a single mission.

12.0 C O N C L U S I O N S

The following conclusions were drawn on the basis of the results obtained from mission, vehicle, and operations research studies:

PART I. MISSION AND VEHICLE STUDIES

1. The approximate 16-year cycle of repeated opposition locations, combined with the large eccentricity of the Mars'orbit, causes a significant variation of ΔV and IMIEO requirements for short, round-trip Mars stopover missions. An unfavorable time period definitely exists.

2. Although total propulsive ΔV is a significant mission evaluation parameter, it is not, in itself, a sufficient criterion for trajectory selection. IMIEO differences of as much as 67% were found between ΔV-optimized and IMIEO-optimized short missions.

3. For the nominal mission/vehicle definition selected, IMIEO for IMIEO-optimized short missions ranged from a peak value of approximately 800 metric tons in 1978 to a value of approximately 300 metric tons in 1986. Minimum IMIEO for long missions was relatively constant for all opposition periods; values were between 300 and 400 metric tons.

4. Principal characteristics for both ΔV-optimized and IMIEO-optimized missions were determined as follows:
   For short missions, optimum stay-time is zero. With stay-time constrained to 40 days, optimum mission duration ranged from 450 to 570 days, and minimum perihelion distance varied from approximately 0.4 A.U. in unfavorable years to approximately 0.7 A.U. in favorable years. Constraining perihelion distance to values in excess of 0.5 A.U. accentuates the penalties incurred from operation during unfavorable years.
   For long missions, optimum stay-time ranged from 300 to 560 days, and optimum duration from 900 to 1030 days. Perihelion distances were greater than 0.9 A.U.
5. In a parametric sensitivity analysis of IMIEO-optimized 1980 and 1986 missions, IMIEO was found to be more sensitive (on a percent IMIEO/percent variable basis) to specific impulse of the nuclear stages (Earth escape, Mars capture, and Mars escape) than to any other performance parameter of the nominal vehicle. Crew size was second in importance.

6. Intermediate-length missions (approximately 800 days), resulting from constraining long missions to shorter-than-optimum durations, do not appear desirable. Application of a heliocentric plane-change maneuver did not produce lower mission total ΔV values;

however, it may be useful for launch window improvement.

7. From the standpoint of heliocentric trajectory selection, the best method of alleviating the cyclic variation of IMIEO (and the large values of IMIEO required in unfavorable years) is to employ long missions. Alternately, the most profitable area of study for the alleviation of short-mission calendar effects appears to be the Earth-capture maneuver.

PART II. OPERATIONS RESEARCH STUDIES

1. On the basis of current estimates of systems performance, P_{ms} can be increased with relative ease up to 90%. Advances above this point for missions prior to 1980 become expensive in terms of IMIEO since duplications of entire systems or high mass redundancies are required.

2. For development risks associated with performance, reliability, and schedule, the risk associated with meeting the reliability objectives is, in general, the greatest.

3. On a <u>one</u>-mission basis, the following conclusions can be drawn concerning overall program costs:
   (a) For minimum time (short) class missions, overall program costs are lower than those of minimum energy (long) class missions.
   (b) The variation in overall program costs is relatively low for the same mission class/vehicle type.
   (c) Systems research and development cost accounts for approximately half of the overall program costs.

4. On the basis of IMIEO, P_{ms}, costs, and development risk for a total program involving only one mission, the most attractive mission class/vehicle type appears to be
   (a) A post-1980, minimum time class (short) mission
   (b) A nominal or multi-nominal vehicle
   (c) An all-chemical propulsion system or a nominal propulsion system with engine-out capability.

5. A 4% decrease in total program cost can be achieved by using a post-Saturn Earth launch vehicle rather than Saturn V.

6. Artificial gravity can be provided on the Nominal vehicle for an increase in program costs of less than 5% and an increase in IMIEO of only 6%.

7. Although a vehicle suitable for atmospheric braking at Mars would cost approximately $1.3 billion dollars more than the Nominal vehicle, the concept warrants additional investigation because of the possible substantial decrease in IMIEO.

26987

PART 10

MANNED MARS EXPLORATION IN THE UNFAVORABLE (1975-1985) TIME PERIOD

by

R. L. Gervais
Douglas Aircraft Company
Contract No. NAS8-11005

PREFACE

This document was prepared by the Douglas Aircraft Company, Inc., for the National Aeronautics and Space Administration, Marshall Space Flight Center, Huntsville, Alabama, under Contract No. NAS8-11005. It is part of the final report for the study, "Manned Mars Explorations in the Unfavorable (1975-1985) Time Period."

The entire final report is made up of the following 12 volumes:

Douglas Aircraft Company Report No.	Title
SM-45575	Volume I – Condensed Summary
SM-45576	Volume II – Summary
SM-45577	Volume III – Systems Integration
SM-45578	Volume IV – Astrodynamics and Vehicle Performance
SM-45579	Volume V – Planet and Space Environment
SM-45580	Volume VI – Entry Analysis
SM-45581	Volume VII – Life Science and Human Factors
SM-45582	Volume VIII – Life Support and Environmental Control
SM-45583	Volume IX – Auxiliary Power and Electronics
SM-45584	Volume X – Vehicle and Structural Design
SM-45585	Volume XI – Propulsion and Propellant Management
SM-45586	Volume XII – Program Development

Section 1
INTRODUCTION

This report summarizes the study, "Manned Mars Explorations in the Unfavorable (1975-1985) Time Period," which was performed by Douglas Aircraft Company, Inc. for the National Aeronautics and Space Administration. The work was done under a seven-month contract, NAS8-11005, from July 1963 through January 1964, for the Future Projects Office of the NASA Marshall Space Flight Center at Huntsville, Alabama. The purpose of the study was to develop data for advanced mission planning and preliminary design criteria of possible future space vehicle systems.

The launch period from 1975 to 1985 constitutes a time of increased difficulty for completing missions to Mars because of the particular planetary configurations available and the increased solar activity expected during these years. Therefore, one of the primary study goals was to determine the magnitudes and kinds of difficulties to be experienced during this period and any possible means of alleviating them.

1.1 STUDY OBJECTIVES

The primary objective of the study, as outlined in the request for quotation, was to "survey all attractive mission profiles for manned Mars missions during the 1975/85 time period, and to select mission profiles of primary interest." The criterion for determining profiles of interest, or optimization in any part of the study, was indicated to be the minimization of vehicle gross weight at the time of launch from earth orbit.

A secondary objective was to "define problem areas which appear to be capable of solution and which, if solved, can markedly improve mission capability during the 1975/85 time period."

An important third objective was the need for a multimission vehicle in which all subsystems are compatible with mission profiles and mission requirements in several synodic periods.

The fourth objective was to determine the nuclear thrust level which should be developed from the standpoint of interplanetary system requirements.

The missions were integrated with payload requirements and vehicle design for evaluation on a total concept basis. Included in this evaluation were the costs and schedules associated with development of any advanced ideas resulting from the secondary objective.

1.2 ASSUMPTION AND GUIDELINES

Instructions in the request for quotation stipulated that all missions would start from an earth orbit at an altitude of 325 km (175 n. mi), would assume Mars orbital operations, and would generally consider only high-acceleration (greater than 0.1g) propulsion systems. Additional guidelines were specified in the proposal or agreed upon at the orientation briefing, prior to initiation of the study. These considerations included:

1. Study of the Mars Excursion Module (MEM) was not to be included
2. A Mars circular capture orbit altitude of 555 km (300 n. mi) was initially assumed
3. Retro propulsion during earth return was for braking to 12.2 km/sec (40,000 ft/sec); aerodynamic braking from 12.2 km/sec to landing was assumed to be current state of the art
4. Retro propulsion braking for attainment of Martian orbit was initially assumed
5. Nuclear electric and nuclear pulse propulsion systems were not to be considered
6. Vehicle designs were assumed compatible with an earth launch vehicle of 70-ft diameter
7. The meteoroid environment established by Whipple's 1963A and D'Aiutolo's data.

1.3 REPORTING APPROACH

Final documentation of the study consists of this summary, a condensed summary, a systems integration volume, eight volumes covering the technical areas, and a volume on program development.

Data from each of the eight technical volumes and the program development volume are summarized in this volume to indicate the general study approach taken, and the information available, in each area. Because of emphasis on weight reduction in this study, a separate section on weight-saving techniques is included. Descriptions of 15 potential mission and system combinations are followed by a description of the adopted mission and system. Finally, the advanced concept development programs are outlined, including schedules, costs, and descriptions of the requirements for the adopted system.

Section 2 SUMMARY OF TECHNICAL DISCIPLINES

A summary of the technical disciplines contributing to the integrated missions and systems concept is presented in this section. Advanced technology concepts and weight-saving techniques are discussed in Section 3. The following technical disciplines are summarized: astrodynamics, planetary and space environment, aerothermodynamics, life sciences, life support, auxiliary power and electronics, structural and material design, and propulsion.

2.1 ASTRODYNAMICS AND VEHICLE PERFORMANCE

The terminology "Unfavorable Period" associated with the launch years of 1975 to 1985 is only partially valid. The late 1970's generally require more energy to accomplish missions, while the early 1980's require relatively high shielding requirements because of increased solar activity. To determine the promising missions during this period the assumptions listed in Section 1 were used to establish a baseline mission profile.

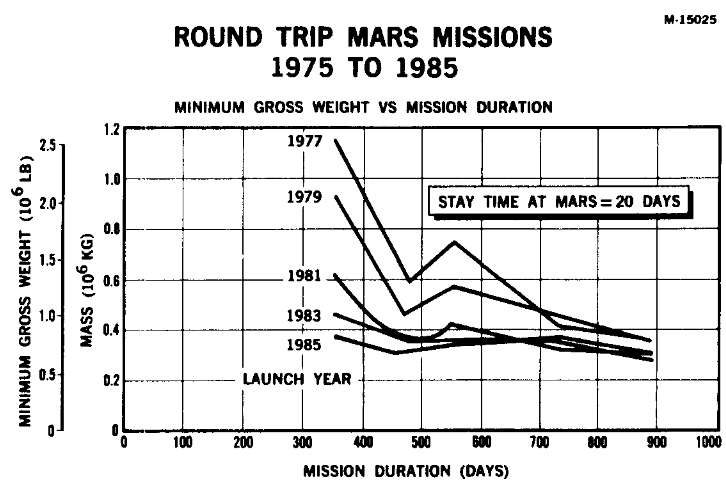
Using this profile, mission durations of 360, 460, 560, 720, and 880 days were examined, including stay times at Mars of 0 to 60 days. Stay times of up to 400 days were considered for missions of longer duration. This range of mission durations and stay times allowed the effects of solar flare activity, biotechnological problems, environmental hazards, and energy requirements to be integrated for proper mission selection.

The energy requirements for the 1975-1985 period can be described by the total velocity required for the round-trip mission, where the total velocity is defined as the sum of the velocity increments required at earth departure, Mars capture, Mars departure, and earth entry. This quantity, as will be shown in detail in Volume IV, is dependent on earth departure date, mission duration, stay time, and the perihelion distance allowed for either segment of the round-trip mission. To obtain the minimum total velocity for any given set of parameters, combinations of earth-Mars and Mars-earth trajectories were examined, where transfer time was used as the coordinating parameter.

Considering regions of minimum total velocity to be of primary interest, vehicle performance analyses were conducted, assuming a stage propellant mass fraction of 0.85 and a specific impulse value of 850 sec for all stages. The trends of gross weight in earth orbit as a function of mission duration for the various synodic periods are shown in Figure 1.

Adjustments were made to the propellant mass fraction and were incorporated in the vehicle design analysis as more data became available and the vehicle systems became better defined. As previously indicated, the term "Unfavorable Period" is justified from the energy requirements associated with short-duration missions in the late 1970's. By comparison, the energy and weight requirements for the 1980's appear desirable. However,

Figure 1



the long-duration missions have low gross weight requirements in all years.

The mission and vehicle performance analyses conducted for the 1975-to-1985 period established the following:

1. The vehicle gross weight in earth orbit is strongly affected by the velocity distribution for any given mission. A launch date having a minimum total velocity requirement does not necessarily yield minimum gross weight. The magnitude of the earth departure velocity is most influential in determining the minimum gross weight, with the general result that minimum first-stage weight corresponds to minimum gross weight.
2. In general, the 1977 synodic period required the greatest weight, and 1985 the least gross weight in earth orbit.
3. Long-duration missions (800-day class) correspond to low-energy transfers and consequently to low gross weights. The synodic period has little effect on the gross weight requirements, as illustrated in Figure 2.
4. Long-duration missions can be composed of either long transfer times and short (0-to-60 day) stays or short transfer times and long (300-to-400 day) stay times. Short stay times require the return flight to be made during the same synodic period as the arrival orbit. Long stay times are the most attractive because they allow the return transfer to be made during the following synodic period at more favorable astronomical configurations.
5. Short-duration missions such as that illustrated in Figure 2 are highly dependent on synodic period since departure and return are accomplished in the same period. The astronomical configuration decidedly affects the energy requirements, resulting in appreciable weight variations throughout the years.
6. Gross weights of short-duration missions are also strongly dependent on stay time. Stay-time effects of two synodic periods are summarized in Figures 3 and 4 for various mission durations. Weight decreases are experienced with small stay times for most short-duration missions.

Figure 2

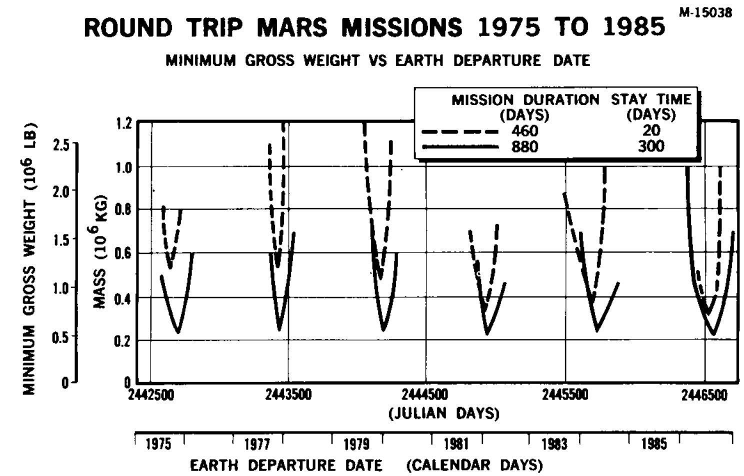


Figure 3

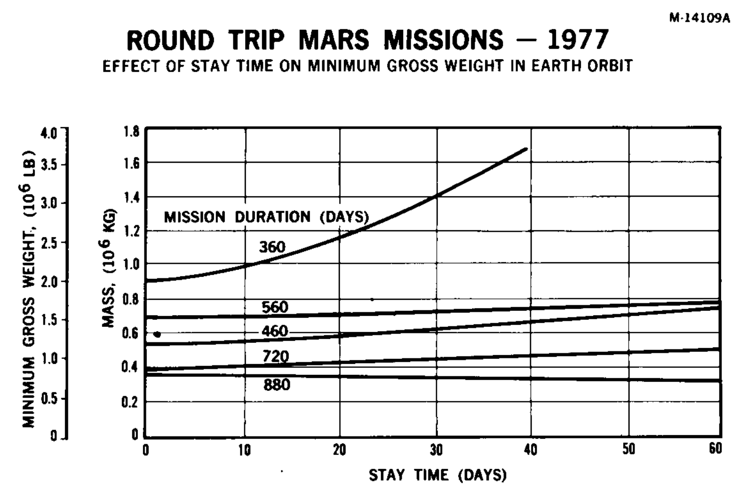


Figure 4

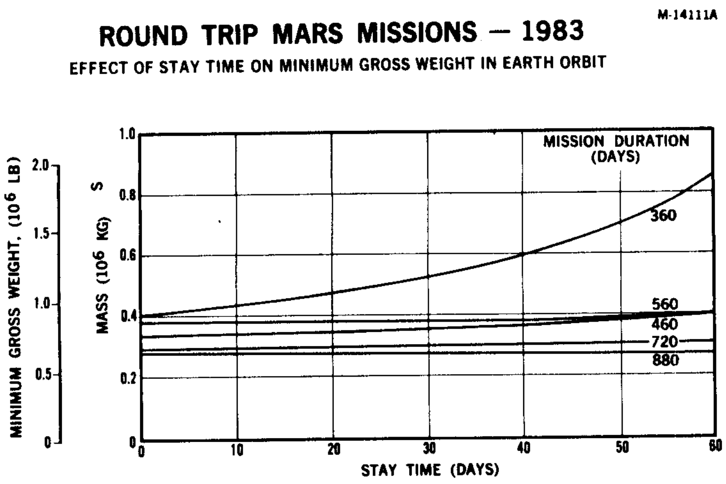


Figure 5

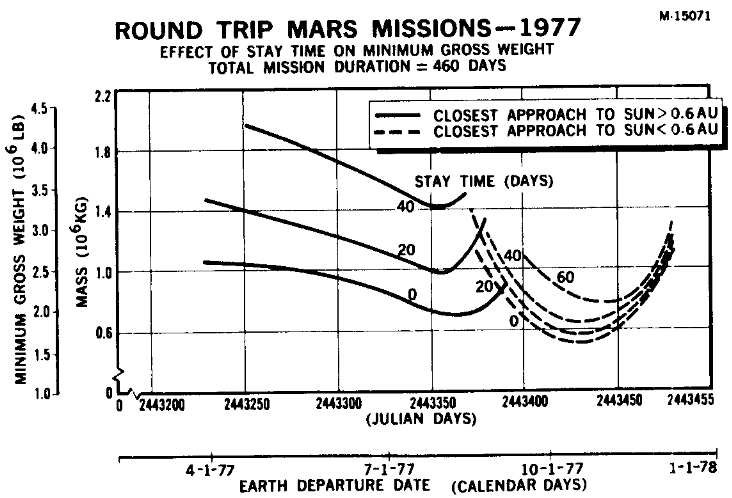
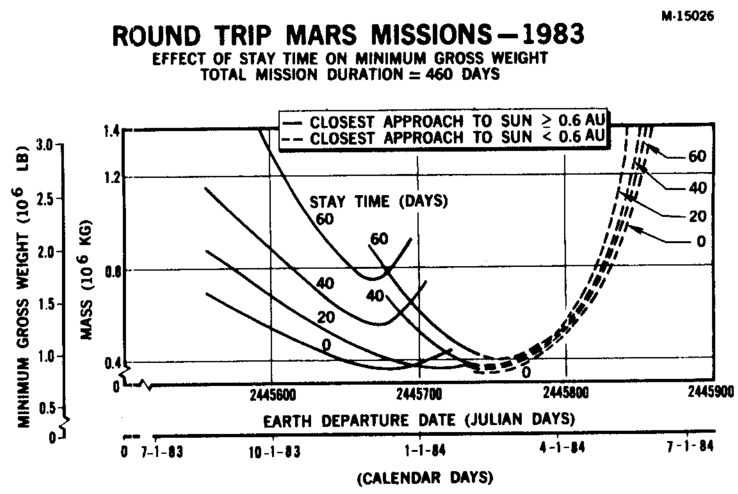


Figure 6



7. A close perihelion passage distance can result in large energy savings for short mission durations (360 to 560 days) particularly during the late 1970's as shown in Figures 5 and 6. Long missions are virtually unaffected throughout all periods.

8. For true minimum weight, the most promising missions are those of 800 to 900 days. These missions have low weight requirements in every synodic period, are not affected by perihelion distance restrictions, and have a large range of stay times that yield low weight requirements. Long stay times at the planet allow ample time for descent to the surface, observation of the planet, or for data gathering and transmission.

9. If vehicle systems or man's ability limit the total mission duration to less than 800 days, missions of shorter duration may be effectively used by incorporating short stay times and allowing perihelion distances of the order of 0.4 astronomical units in the mission profile. When compared with long-duration

missions, missions of the 460-day class as much as double the weight required in earth orbit during the late 1970's. However, the weight penalty is sharply reduced during the 1981 to 1986 period.

2.2 ENVIRONMENT

Four model atmospheres for the planet Mars were selected for use in aerodynamic entry studies. Three are for a molecular weight of 28.0 with surface pressures of 37, 85, and 162 millibars. They are representative of minimum, mean, and maximum conditions. The fourth model is for a molecular weight of 32.8 with a surface pressure of 37 millibars, representing extreme minimum conditions. These models indicate the expected numerical range of pressure, temperature, and density to be encountered in the Martian atmosphere from 0 to 400 km (1,312,300 ft) during the 11-year solar cycle of 1975 to 1985.

A space vehicle will be exposed to three major sources of charged-particle radiation, namely galactic cosmic rays, geomagnetically trapped radiation, and solar cosmic rays. Galactic cosmic rays give a dose of only a few rads per year, and geomagnetically trapped radiation can be effectively bypassed by choosing a low-altitude parking orbit. Passage through the Van Allen zone results in a dose of only a few rads and since the atmosphere on Mars has an appreciable thickness, the radiation dose on the surface is only a few rads per year. The most hazardous radiative source is solar cosmic rays.

Although the number of large solar cosmic ray events so far observed is too small to accurately calculate a frequency of occurrence, a tentative estimate varies from one to two per two-year period, depending on the solar cycle. These events are of the size that would produce a dose of a few thousand rads behind a 1 gm/sq cm shield.

The meteoroid flux models and particle characteristics described separately by D'Aiutolo and Whipple were selected as being representative of the current flux estimates. The near-Mars particle flux is based on a combination of Whipple's near-earth flux curve and an average particle density of 3 gm/cu cm. Erosion rates have been shown to be small.

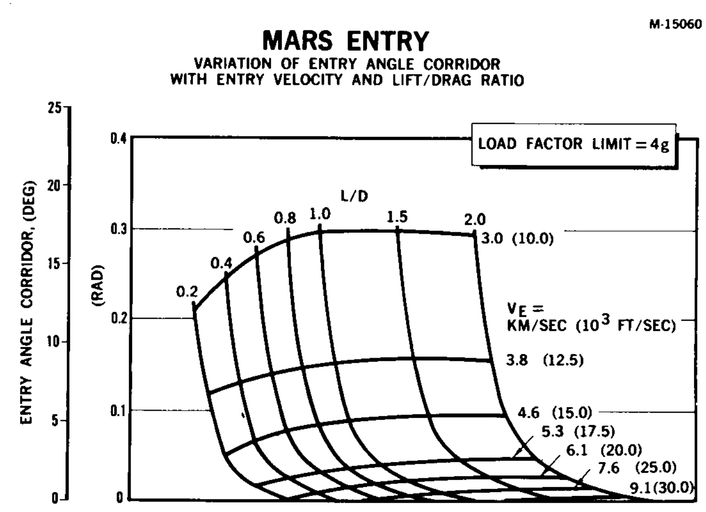
2.3 AERQTHERMODYNAMICS AND GUIDANCE AND CONTROL

2.3.1 Mars Entry

The effects of atmospheric variation on entry corridors were determined for the "nominal," severe, and extreme atmospheres discussed in Section 2.2. The principal effect of the different atmospheric models was a variation in the entry corridor location; the entry corridor magnitude was virtually unaffected.

Figure 7 indicates the variation of entry corridor magnitude with entry velocity and vehicle lift-drag ratio for a deceleration limit of 4 earth g's. It can be seen that a lift-drag ratio of slightly more than 0.5 provides an entry corridor at velocities up to 7 km/sec (22,960 ft/sec.). For most practical missions the vehicle lift-drag ratio required to meet corridor constraints is relatively low, hence the actual vehicle lift-drag ratio may well be governed by other considerations.

Figure 7

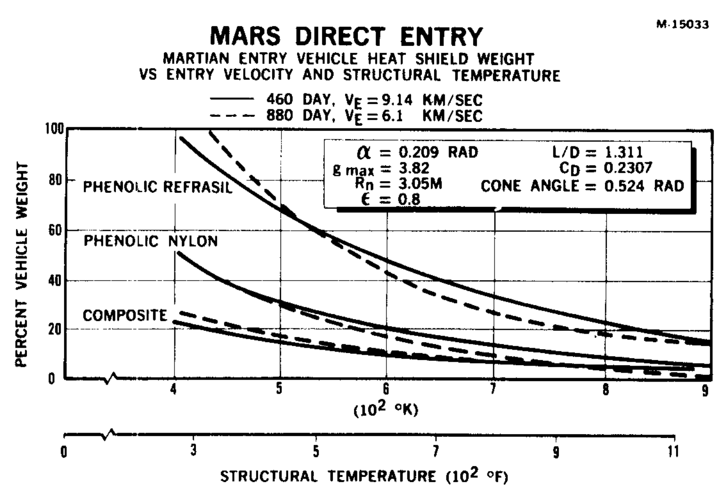


The uncertainties in estimating entry conditions just prior to entry were determined for a navigation system which repeatedly measures the angle between the target planet and a star in the orbit plane. The navigation system has an initial estimate of the uncertainties in the orbit-defining parameters: entry velocity, entry flight path angle, and entry range angle. The difference between the predicted and observed planet angle is used in an optimal estimation procedure to revise the estimated errors and uncertainties in these entry parameter errors each time a measurement is made. A reference trajectory, having an entry velocity of 7.62 km/sec (25,000 ft/sec) with a corresponding entry corridor of 0.00218 radians (0.125 degrees), was used to evaluate navigational accuracy. Although the analysis determined only the navigation accuracy, the results may be considered representative of an ideal guidance system because the navigation errors may be corrected by a guidance system to the same degree of accuracy as the known errors. In all cases, the uncertainty in entry flight path angle at a range of 8000 km was less than 0.00025 radians (0.014 degrees), which indicates a reasonable margin for the degradation of a real guidance system while maintaining a high probability of arriving within the entry corridor.

A thermodynamic analysis of direct Mars entry was made using entry trajectories for velocities of 6.10 and 9.15 km/sec (20,000 and 30,000 ft/sec). The entry vehicle was assumed to be a 0.26 radian (15 degrees) half-angle cone with spherical nose blunting and a lift-drag ratio of 1.31. Trajectories were simulated at a constant angle of attack using bank angle modulation where necessary to prevent skip-out or excessive atmospheric penetration. A peak load factor of 4 earth g's was achieved on both trajectories. The entry thermal design for all vehicles was based on accepted or proven heat-transfer prediction methods in a point-by-point analysis to obtain maximum heat fluxes, integrated heat, and maximum equilibrium temperatures for the preliminary selection of heat shield materials. These materials were analyzed in Douglas aerodynamic heating programs to obtain unit weight requirements (as a function of structural temperature and geometry) for specified points on the vehicle. The emissivity values used for the prediction of equilibrium heating rates in the stagnation region were obtained by modifying the Kivel and Bailey data generated for an earth atmosphere by the molar concentration of the species behind the shock. Since the data of Kivel and Bailey are suspected of being high by a factor of two, the effect of this reduction in the predicted radiation rates on total vehicle heat shield weight was also determined; the reduction was approximately 0.5% (heat shield to total vehicle weight).

An analysis of ablative materials (phenolic nylon and refrasil) revealed that, of the total material required to maintain a given structural temperature, less than 50% was ablated. This nonablated, virgin material was then replaced by an insulation material of high temperature, low density, and low conductivity, resulting in heat shield weight reductions as high as 80% for phenolic refrasil and 25% for phenolic nylon. Figure 8 illustrates the variation of heat shield requirements with material, structural temperature, and entry velocity for an 880-day and a 460-day mission. The composite (phenolic refrasil and low-density, high-temperature insulation) structure is most attractive throughout the temperature range with phenolic nylon becoming competitive at the high temperature.

Figure 8



The use of aerodynamic braking to achieve a Mars orbit was examined by means of an aerodynamic braking trajectory which consisted of the following: an initial plunge to pullout at maximum positive lift, followed by an altitude hold using lift modulation until velocity decreased to slightly above satellite velocity. At this point, maximum positive lift was applied and the vehicle exited the atmosphere on an ellipse. Impulsive velocities required to circularize the exit ellipse were found to vary between 0.1 and 0.3 km/sec (0.33 and 0.11 ft/sec), depending on entry conditions and orbit altitude. The thermal protection system for aerodynamic orbit attainment was found similar in weight to that required for direct entry.

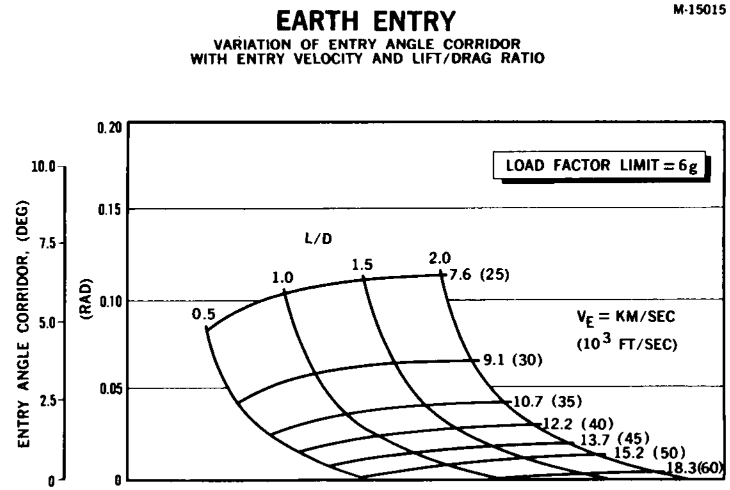
2.3.2 Earth Entry

Corridors were determined for entry velocities up to 21.4 km/sec (70,000 ft/sec). Data associated with a load factor limit of 6 g's are presented in Figure 9, which indicates that, for this limit load factor, an entry corridor exists up to an entry velocity of approximately 18 km/sec (59,000 ft/sec). Various techniques have been suggested to extend this upper limit on entry velocity (e.g., modulation prior to pullout); however, these techniques do not permit entry at appreciably higher velocities but merely widen the corridor at velocities where a corridor already exists.

Entry trajectories were simulated for a blunt-body configuration having a lift-drag ratio of 0.576 and a modified half-cone having a lift-drag ratio of 1.2. Maximum load factors of 3, 6, and 10 g's were used with the blunt-body configuration; 6- and 10-g trajectories were generated for the half-cone vehicle. All trajectories used bank-angle modulation to fly at constant altitude from pullout to an equilibrium glide. Thermal analyses of both the blunt-body vehicle and the half-cone configuration were conducted in the same manner as for the Martian entry vehicle with the exception of the thermal radiation.

The stagnation-point convective heating rates were predicted using the method of Hoshizaki. Equilibrium radiation rates were determined using the data of Kivel and Bailey modified to account for the changes in f numbers. Nonequilibrium radiation rates were assumed to be independent of density and nose radius. Again, a point-by-point determination of heating rates, integrated heat, and equilibrium temperatures led to the selection of a range of ablative materials. Analysis of these materials as heat shields again favors the composite (phenolic refrasil and a low-density, high-temperature insulation). Figure 10 shows the variation of heat shield requirements with entry velocity, structural temperature, and material. Heat shield weight varies from approximately 22 to 37% of the total vehicle weight for a structural temperature of 700°K (800°F). Also indicated is the advantage gained by using the composite system for the blunt-body vehicle which amounts to 13.5% of the vehicle weight for a phenolic nylon heat shield and a structural temperature of 700°K.

Figure 9



2.4 LIFE SCIENCES

Life science studies were initiated to determine vehicle-manned payload interactions and to investigate the effects of payload requirement variations on mission success. The payload factors of concern were crew size and composition, metabolic and volumetric requirements, and

Figure 10

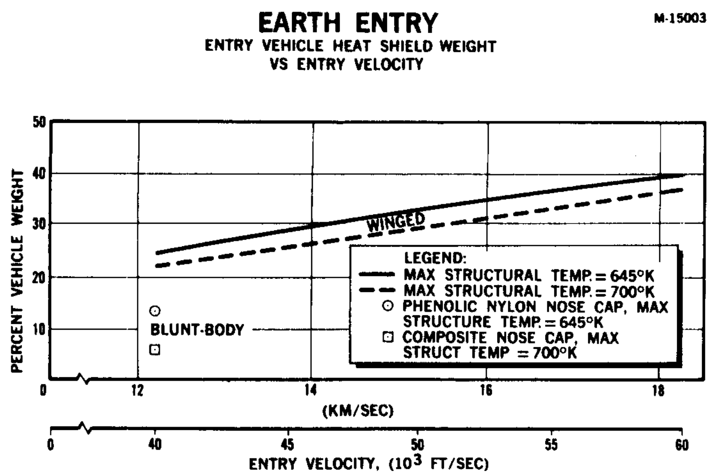
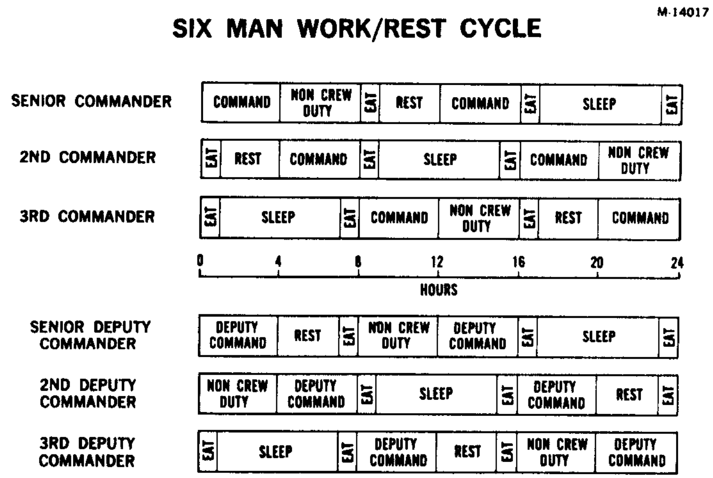


Figure 11



tolerance to acceleration, zero gravity, radiation, and long confinement.

A preliminary operations analysis arrived at a minimum desirable crew size of six men based on time-line and duty cycle requirements and crew requirements for the Martian exploration phase. Duty and professional requirements were then used to describe crew skills, distribution of these skills among the crew, and provisions for backup capability in a minimum crew.

A desirable work-rest cycle is shown in Figure 11. The time relations in this cycle essentially describe conditions commensurate with man's accustomed schedule in a terrestrial environment. An emergency work-rest cycle was established for three men to describe an activity schedule that might be observed under an emergency condition such as return from Mars with reduced crew.

Crew activities defined by the work-rest cycle were analyzed to determine metabolic energy requirements. These requirements, shown in Figure 12, indicate an energy need of 11,816 K Joules/man/day. The material exchanges involved are described in Figure 13.

Minimum volume requirements were initially determined by studying habitable spaces which can accommodate a 95-percentile man. Psychophysiological factors are extremely important in this analysis and, because of limited information, only preliminary estimates could be made. A net volume of 31.15 M^3 per man (1100 cu ft/man) was used for vehicle-sizing purposes. Opinions and predictions in the scientific community differ widely on the effects of long exposure to zero gravity. Therefore the effectiveness of several techniques proposed to prevent adverse effects was evaluated and centrifugation selected as the most promising. Two methods of providing centrifugal force were presented. One method provides a continuous-force field by rotating the vehicle; the other provides an onboard centrifuge for intermittent conditioning. Both methods were studied with regard to mission planning and vehicle habitability design, and although both are considered equally applicable from the life science standpoint, the zero-gravity concept was used in conceptual design of compartment interiors because of other considerations.

Figure 12

M-14018

MAN-DAY ENERGY EXPENDITURES

FUNCTION	TIME (HOUR)	RATE PER HOUR BTU	RATE PER HOUR K JOULES	T BTU	T K JOULES
COMMAND	8.0	500	528	4000	4220
NON-CREW DUTY					
MAINTENANCE	1.0	780	823	780	823
MOVEMENT	.6	1400	1477	840	886
MENTAL WORK	1.0	420	443	420	443
MISCELLANEOUS ACTIVITIES	.5	550	580	275	290
EXERCISE	.4	2400	2532	960	1013
REST AND RELAXATION					
STANDING	.5	400	422	200	211
STUDYING	1.0	400	422	400	422
PLAYING CARDS	1.0	450	475	450	475
SITTING AT REST	1.0	400	422	400	422
EATING	1.5	400	422	600	633
SLEEPING	7.5	250	264	1875	1978
TOTAL	24.0			11,200	11,816

Figure 13

M-14019

MATERIALS EXCHANGES

FUNCTION	MATERIAL	1 MAN/DAY KG.	1 MAN/DAY LBS.	6 MAN/DAYS KG	6 MAN/DAYS LBS.	6 MAN 460 DAYS KG	6 MAN 460 DAYS LBS.	6 MAN 880 DAYS KG	6 MAN 880 DAYS LBS.
INPUT	OXYGEN	.816	1.8	4.90	10.8	2253	4968	4311	9504
	WATER	2.586	5.7	15.51	34.2	7136	15732	13651	30096
	FOOD	.816	1.8	4.90	10.8	2253	4968	4311	9504
	TOTAL INPUT	4.218	9.3	25.31	55.8	11642	25668	22273	49104
OUTPUT	CARBON DIOXIDE	.953	2.1	5.71	12.6	2629	5796	5029	11088
	WATER								
	INSENSIBLE	1.197	2.64						
	URINE	1.601	3.53						
	FECES	.15	.33						
	TOTAL WATER	2.948	6.5	17.69	39.0	8137	17940	15567	34320
	SOLID	.317	.7	1.91	4.2	876	1932	1677	3696
	TOTAL OUTPUT	4.218	9.3	25.31	55.8	11642	25668	22273	49104

Human acceleration tolerance limits for various mission phases were established from analysis of information on proposed techniques which permit increased acceleration exposure. In addition, dose and dose rate criteria were established from available information on radiation risks and their implications on crew operations.

2.5 LIFE SUPPORT

A life support analysis was conducted to determine an integrated life support and environmental control system for various crew sizes and mission durations. In the study, the crew size was varied from three to ten men and mission durations were varied from 300 to 1,000 days. A six-man crew was then selected as a basis for comparing the various techniques of the individual subsystems.

The following main subsystems were considered as part of the integrated life support and environmental control system: thermo-conditioning, atmospheric supply, water management, and atmospheric conditioning. Additional subsystems were included in the weight summary and power requirement summary. These included food management, waste management, contaminant detection, fire protection, and cabin conditioning. The atmospheric supply subsystem will be used as an example to summarize the effects of mission duration on system weight. Figure 14 shows the systems weight breakdown for a six-man crew for missions of varying duration.

Figure 15 shows the weight breakdown of one of the techniques for supplying the required atmosphere. A reliability analysis was conducted to obtain a more complete picture on the more promising techniques and, as a result, the spare parts requirement has been added to the total weight curve of Figure 15. In addition, this curve indicates (Point A) the weight saving that can be obtained by repair rather than replacement. Figure 16 shows the individual subsystems integrated into a single environmental control/life support system. The integrated system makes maximum use of all subsystem byproducts as well as heating and cooling sources. Figures 17 through 21 present brief schematics of the main life support subsystems.

Figure 14

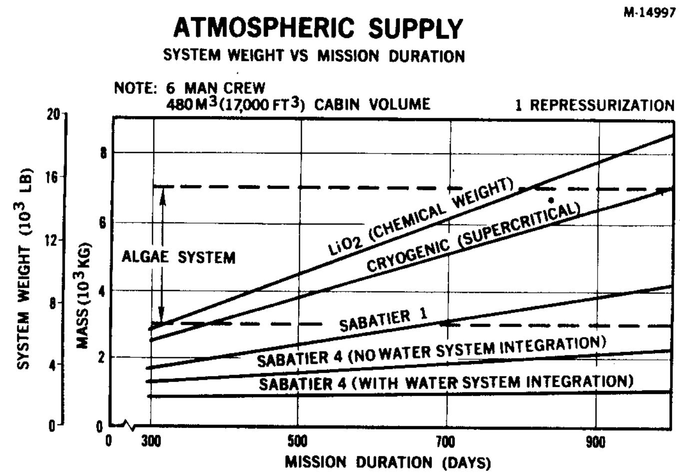


Figure 15

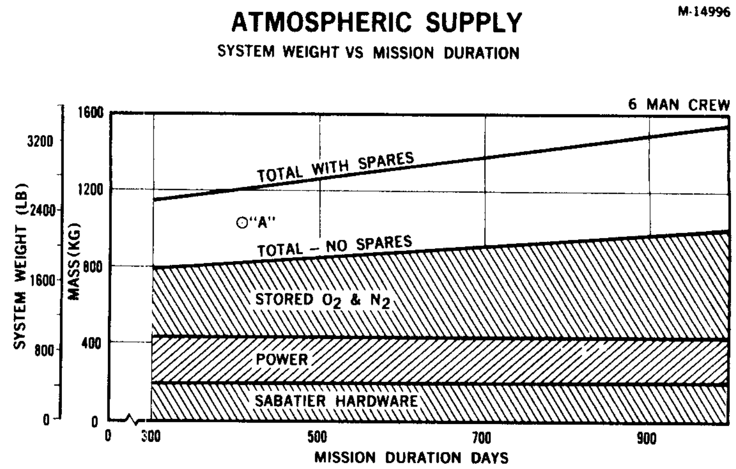
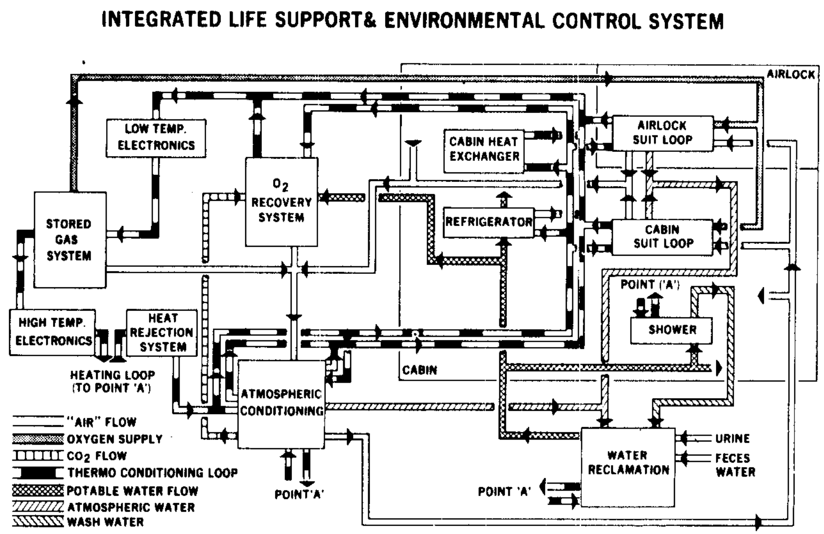


Figure 16



2.5.1 Thermo-Conditioning

As a result of comparing passive, semipassive, and active heat rejection systems, a semipassive system was selected for thermo-conditioning because of its low weight, low power requirement, high reliability, ability to operate in zero gravity, and high rate of heat rejection. The system was sized for the maximum "effective space temperature" of 213° K (383° R) which occurs during the earth-orbit phase of a mission. During interplanetary flight, the vehicle is "sun oriented" with the radiators facing opposite the sun. The system shown in Figure 17 consists of a circulating coolant, pumps, radiators, equipment cold plates, controls, and necessary plumbing. The system weighs approximately 240 kg (529 lb) and requires 240 watts of power to dissipate a continuous heat load of approximately 20 kw. Thermal control is accomplished using bypass valves out four segmented radiators, each with an area of 17 M^2 (184 sq ft) and each radiating from both sides. As the heat load or effective space temperature changes, the number of radiator segments in use is adjusted to maintain the proper system thermal balance.

2.5.2 Atmospheric Supply

Methods considered for oxygen supply include stored chemicals, cryogenic storage, regeneration of O_2 from CO_2 by algae, and regeneration of O_2 by chemical reaction of CO_2 with hydrogen. The Sabatier reaction of CO_2 with hydrogen was selected because of its low weight and high reliability. In this reaction ($CO_2 + 4H_2 \rightarrow 2H_2 + CH_4$), the methane is decomposed and the resulting hydrogen returned to the Sabatier reaction. Water from the Sabatier reaction is electrolyzed, and the resulting hydrogen returned to the Sabatier reaction and the oxygen to the atmosphere. Makeup oxygen is provided by electrolyzing water from the water-management system, which is shown in Figure 18. Nitrogen and emergency oxygen are stored as cryogens with boiloff eliminated, using cryostats to remove heat from the storage vessels. The atmospheric supply system mass is 832 kg (1,835 lb) for six men on a 460-day mission and 945 kg (2,080 lb) for an 880-day mission. The power requirements are 3.233 kw.

Figure 17

THERMO-CONDITIONING SYSTEM

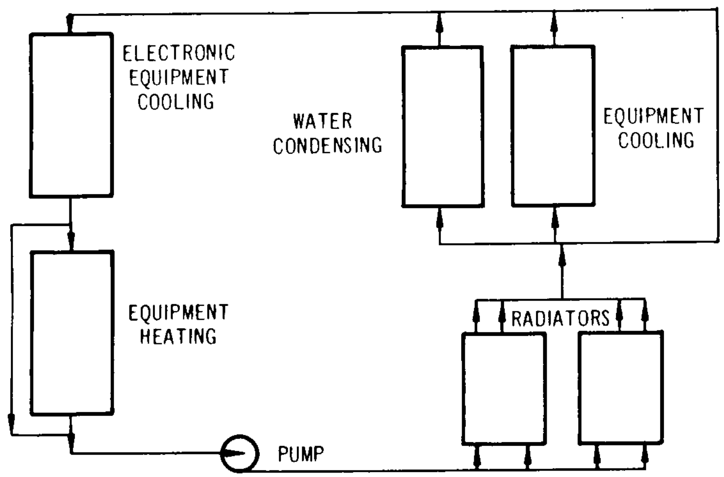


Figure 18

ATMOSPHERIC SUPPLY SYSTEM

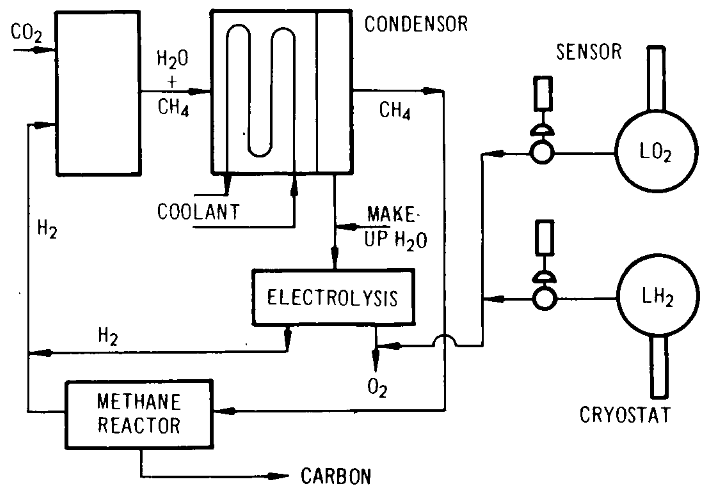
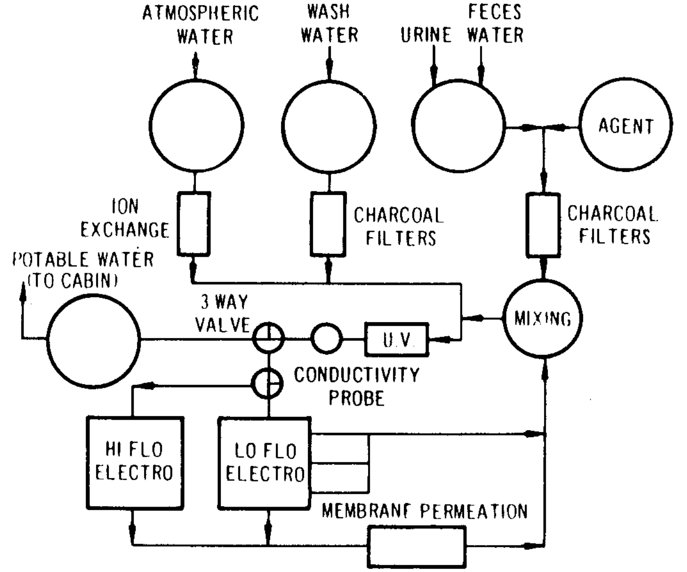
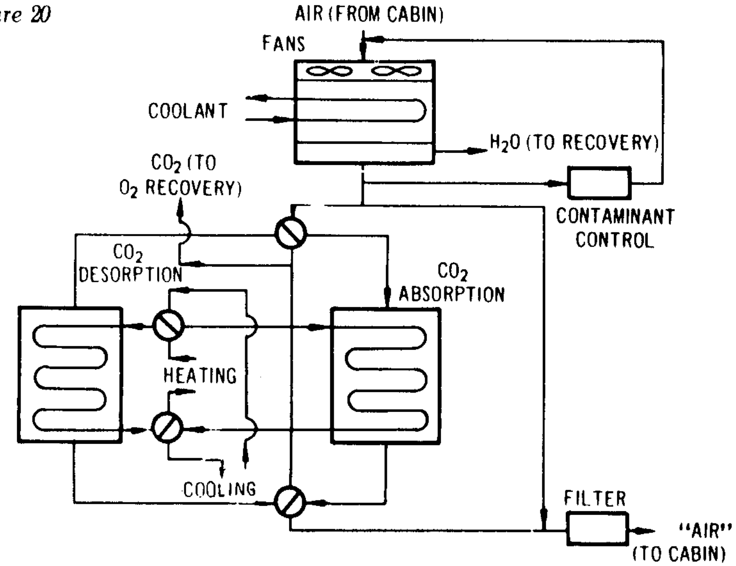


Figure 19 WATER RECLAMATION SYSTEM



ATMOSPHERIC CONDITIONING SYSTEM

Figure 20



2.5.3 Water Management

Techniques considered for reclamation of water were vapor-compression distillation, vacuum distillation with pyrolysis, and electrodialysis. Electrodialysis, as shown schematically in Figure 19, was selected because of its low weight and high reliability. For purification of urine, urea complexing agent is added and the urine filtered through charcoal before entering the electrodialysis unit. This unit removes ions by transferring them through cation and anion permeable membranes. An ultraviolet lamp in the system assists with sterilization. Wash water is also filtered before passage through an electrodialysis unit. Water condensed in the atmospheric conditioning system is passed through a mixed bed of charcoal and ion exchange resin for purification. Charcoal filters are regenerated to save weight. An additional water supply for drinking and hydration of food is stored for emergency purposes. The system mass is 210 kg (460 lb) for six men on a 460-day mission, and 230 kg (508 lb) for an 880-day mission. The power requirements are 95 watts.

2.5.4 Atmospheric Conditioning

A number of methods of removing CO_2 from the ventilating air stream of a space vehicle were considered for atmospheric conditioning and comparisons made with regard to simplicity, reliability, and power requirements. The comparison included regenerative techniques such as absorption with ion transfer through membrane, semi-permeable membrane, mono-ethanolamine (amine solution), pellet form of amine, magnesium oxide (metallic oxide), molecular sieve, and freeze-out. The three lightweight processes (pellet form of amine, magnesium oxide, and molecular sieve) were selected for further study on the basis of simplicity, reliability, and low power requirements. The pellet form of amine was finally selected because it has the added advantage of being unaffected by the moisture content of the air stream and can be regenerated at 80° C (175° F). This system, shown in Figure 20, operates on the two-bed principal, one bed absorbing while the other bed is being desorbed. It also includes a condenser for controlling the moisture content of the air stream and a catalytic burner for contaminant control. Collected CO_2 is transferred to the atmospheric supply system to recover the oxygen. This system weighs approximately 30 kg (66 lb) and uses 0.82 kw of power.

2.5.5 Waste Management

The waste management system uses air flow through a permeable bag for pneumatic collection of feces. The feces are partially dried and sufficient water recovered to balance the water and oxygen cycle. The feces are then stored to provide radiation protection. The feces collection and drying system mass is 25 kg (55 lb) and requires 110 watts of power.

2.5.6 Food Management

The life science material exchange and energy-balance requires storage of 0.816 kg (1.8 lb) of dry food per man per day. Dry foods were compared with partially hydrated Air Force IF-10 rations and were selected because of their lower weight. Container weights are approximately 10% of food weight.

2.5.7 Contaminant Detection

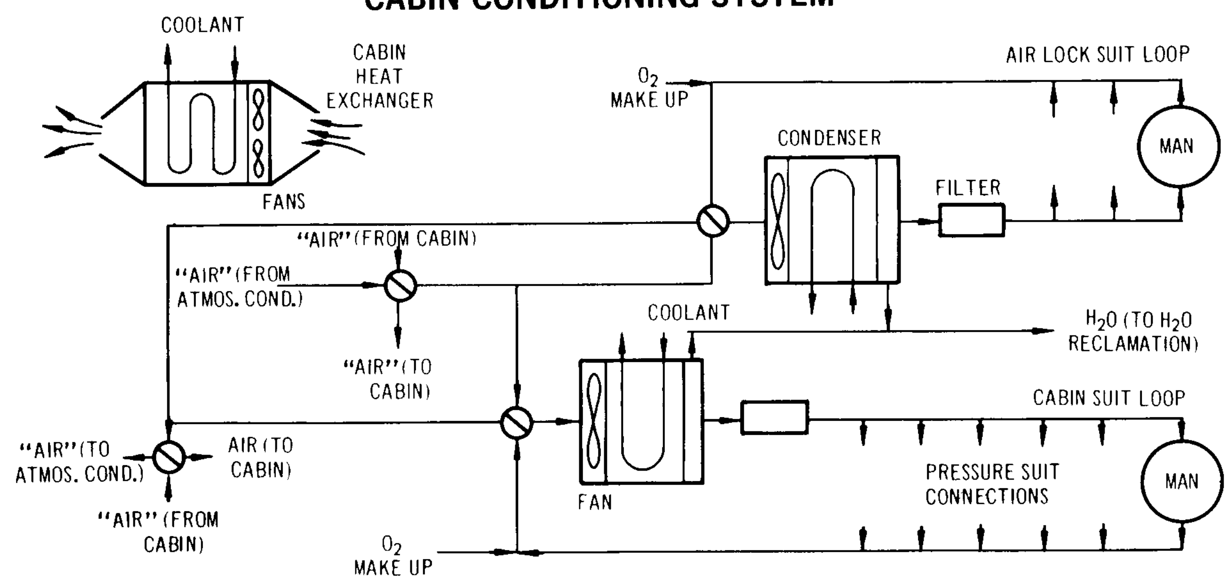
A mass spectrometer and a gas chromatograph will be used for detection of trace contaminant gasses. The mass of the spectrometer is 4.54 kg (10 lb) with a power requirement of 15 watts. The chromatograph mass is 6.8 kg (15 lb) and requires 5 watts.

2.5.8 Cabin Conditioning

This system, shown in Figure 21, provides cabin temperature control, cabin ventilation, air filtering, and supply gas mixing, as well as coolant to the refrigerator and heat for food preparation and washing. The suit-loop system provides ventilation, cooling, filtration, gas supply, and pressure control. Each loop can operate with cabin "air," with the cabin atmospheric control and purification subsystem, or independently. The suit loop operates independently by recirculating 100% oxygen and leaking a controlled amount to hold the partial pressure of CO_2 at a safe level. The suit-loop system selected was based on

Figure 21

CABIN CONDITIONING SYSTEM



flexibility and "graceful degradation." However, the selected system also provides backup support to the cabin environmental control and life support system during emergencies, thus allowing the back-pack system with the expendables to be saved for emergency measures only. Both the airlock and cabin are provided with suit-loop systems.

2.5.9 Reliability

A spare provision list has been generated for the life support system and its associated subsystem by a Douglas-designed computer program using component weights and data from the U.S. Navy Bureau of Weapons FARADA Program on component failure rates. This list provides an order of preference in which spare parts should be chosen in order to realize the greatest increase in potential reliability per pound of spares. In addition, the program provides weight and reliability figures which show attainable system reliability with the addition of each spare. In this way, the exact weight where reliability meets its allocated goals is easily determined.

The total weight of the completely integrated system for a 460-day mission would be 422 kg (9305 lb) for equipment and expendables. Spare parts for replacement would add another 818 kg (1,826 lb) to meet the reliability goal. The total weight would therefore be 5,048 kg (11,131 lb), requiring approximately 6.3 kw maximum power and 5 kw of continuous power.

2.6 AUXILIARY POWER AND ELECTRONICS

2.6.1 Auxiliary Power

Nuclear, solar, and isotopic power sources were considered for the auxiliary power system in the vehicle. The power-load profile used in evaluation is presented in Figure 22, which indicates that the major load contribution results from the hydrogen reliquefaction system.

The Nuclear-Rankine System, shown schematically in Figure 23, is recommended for the 460-day mission. This system employs two 300-kw thermal nuclear reactor heat sources of the SNAP-8 type (capable of 600 kw thermal each) and five 120-kw Mercury-Rankine [980°K (1,300°F)] conversion systems. The system includes one redundant reactor and two redundant conversion units. Mission durations in excess of 460 days require spare units to accommodate the wearout cycles in addition to those required for random failures. The nuclear system was selected as the result of an evaluation of development and mission risks in addition to a tradeoff of systems weight and performance as shown in Figure 24. Radioisotope source systems are not recommended for this mission because of the limited availability and high cost of the source material.

Solar dynamic systems are competitive with nuclear systems and are, in fact, lighter than nuclear systems up to 20 kw, as demonstrated in Figure 24. However, the uncertainty associated with the life and performance of a large concentrator during prolonged exposure to the space environment negates the recommendation of the solar dynamic system. The stringent orientation requirements are also undesirable restraints on the vehicle.

Figure 22

M-15001

SUBSYSTEM POWER REQUIREMENTS

SUBSYSTEM	POWER LEVEL, KW — EMERGENCY MODE, CREW 3	6	10	EARTH ORBIT LAUNCH, CREW 3	6	10	CONTINUOUS, CREW 3	6	10	INTERMITTENT, CREW 3	6	10
1. LIFE SUPPORT & ENVIRONMENTAL CONTROL SYSTEM	3	5	6	4	6	8	4	6	8			
2. VEHICLE MAINTENANCE & OPERATION	2	2	2	2	2	2	3	3	3			
3. COMMUNICATIONS	.5	.5	.5	1	1	1	3	3	3			
4. PROPULSION				2	2	2					2	
5. GUIDANCE & CONTROL				2	2	2	1	1	1		2	
6. EXPERIMENTS							3	3	3		2	
7. PROGRAMMED INTERMITTENT LOADS							4	4	4			
8. CENTRIFUGE											2	
9. HYDROGEN RELIQUEFACTION							31.5	31.5	31.5			
10. DATA MANAGEMENT				1	1	1	1	1	1			
11. TELEVISION											3–5	
12. ATTITUDE CONTROL (ELECTRICAL)				5	5	5	5	5	5		10	
13. RADAR MAPPING & OPTICAL SYSTEMS											3	
TOTALS	5.5	7.5	8.5	17	19	21	55.5	57.5	59.5		26	

Figure 23

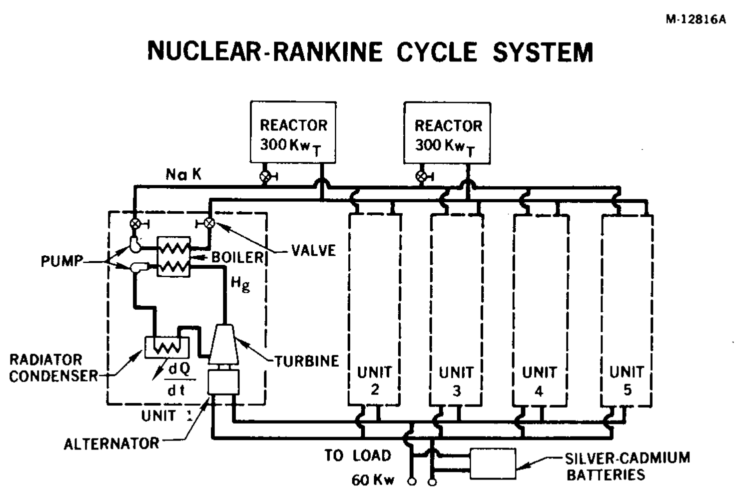
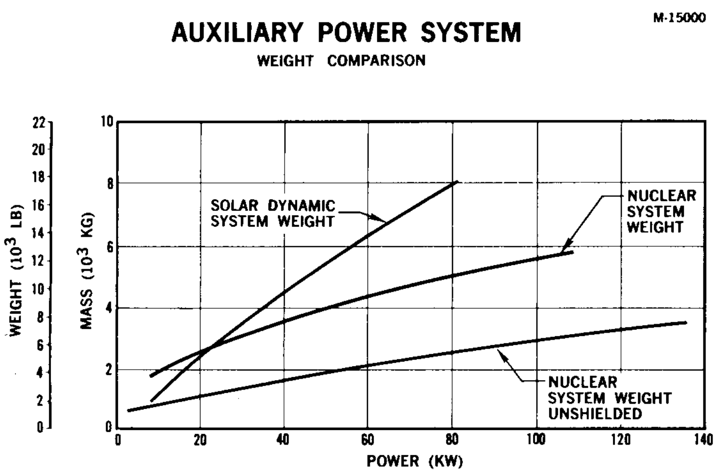


Figure 24



Brayton cycle and thermionic conversion systems were also considered for use with the nuclear source. Although the Brayton cycle system has the advantage of using a single-phase working fluid, it has lower cycle efficiency and requires more radiator area than the Rankine cycle. Also, high-temperature Brayton cycle gas bearings present a development problem comparable to that of Rankine cycle boiling and condensing in "zero-g." The thermionic converter offers very high power density energy conversion. However, development of this system is not anticipated until considerably beyond 1985 and, therefore, not compatible with the time period considered in this study.

2.6.2 Electronics

Two Mars landing missions were selected as a guide for electronic system considerations: a 460-day mission with Mars encountered at opposition and an 880-day mission with Mars encountered at conjunction. Both missions include a 40-day stay in orbit about Mars by the trans-Martial vehicle and approximately a 40-day surface stay for the vehicle. The study areas of primary concern are: (1) data handling, compaction, and transmission; (2) scientific experiments and instrumentation; (3) navigation, guidance, and control, and (4) power distribution.

Analysis indicates that communications from the Mars vehicle to earth are limited by the availability of transmission power. Figure 25 indicates 20kw is required to transmit (at maximum earth-Mars distance of 250 million n. mi) a minimal TV presentation (approximately 1 mc/sec information bandwidth) between a 690-meter (210-ft) diameter ground antenna and a 65.6-meter (200-ft) diameter spacecraft antenna. The major restriction on communications between planets, as shown in Figure 26, is the presence of the sun along the radio path. One method of overcoming the total radio silence would be through the use of one or more solar satellites with an orbital path that permits both earth and Mars to view the repeater satellite without receiving the radio energy of the sun. Alternate solutions would be to schedule critical vehicle maneuvers when they can be viewed from earth.

Figure 25

M-15016

TRANSMITTER POWER REQUIREMENTS

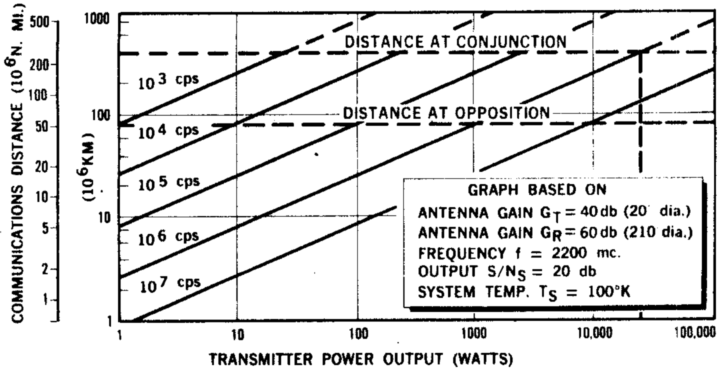
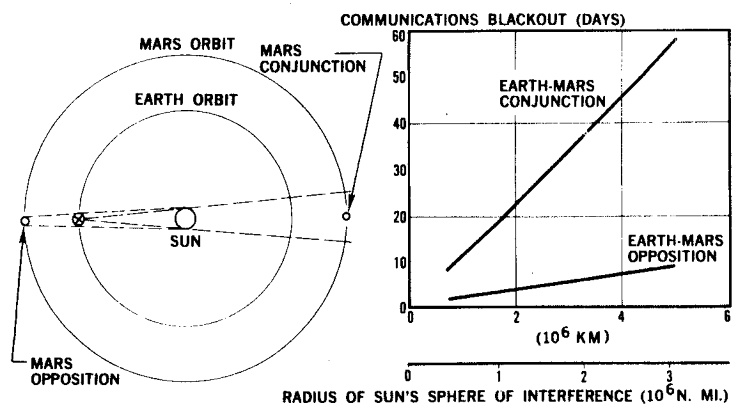


Figure 26

M-15017

COMMUNICATIONS BLACKOUT



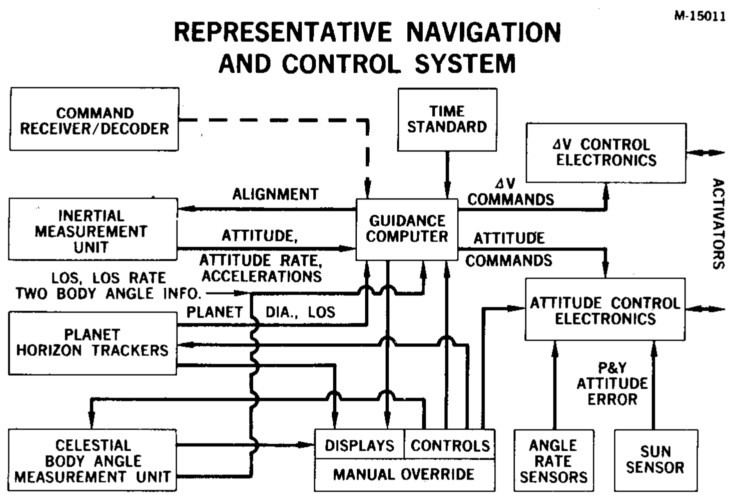
A study of the available data concerning the Mars atmosphere, the nature of magnetic fields, and the extent of ionzied layers above the Mars surface indicates that suitable transmission frequencies are available during the stay on the surface and in orbit about Mars.

The data-handling system required for this vehicle will be more complex than any such system previously included in a space vehicle. Onboard data processing will vary between no processing to complete reduction and analysis for some experiments. Both personnel and computers will, therefore, be used as part of the data-handling process.

The experiments and instrumentation requirements were investigated only to the extent to permit estimates for sizing the power system communications system. The following is a weight breakdown of the electronics subsystem: communications, 360 kg; test systems, 100 kg; instrumentation, 400 kg; data processing, 200 kg; navigation equipment, 240 kg; miscellaneous, 200 kg; spare parts, 1500 kg; scientific payload, 2600 kg; total electronic subsystems, 5600 kg (12, 300 lb).

The Mars vehicle navigation and control system were analyzed on the basis of the general requirements of the three mission phases: transfer orbit, mid-course, and planet homing. Conservative extensions of present techniques and equipment performance were used in the generation of representative systems. The system shown in Figure 27 uses conventional inertial-guidance techniques during all thrusting periods of the mission and is considered a self-contained system. The first mission phase, the transfer orbit, uses this technique to arrive at the heliocentric trajectory from earth or earth-orbit launch. Optical triangulation techniques are employed for the bulk of the mid-course measurements to determine spacecraft position and velocity for trajectory calculations. During the homing phase, triangulation measurements are made as during mid-course. Planet range is determined optically from the angle subtended by the planet disc, although radar techniques may be employed, depending upon the orbital requirement and altitude. Position measurement and trajectory calculations are made enroute to provide thrust impulses to reduce the calculated miss distance.

Figure 27



The guidance computer is the system element that will undergo the greatest technological improvement in the next decade. The primary system effects are those of increased emphasis on component reliability and navigational clock accuracy.

Provisions have been made for interconnecting the primary power generators. Computer-operated switching and analysis of protection and control systems are provided, as well as a manual over-ride to permit rebuilding the system if failures occur. The electrical power distribution system delivers power to all electrically operated equipment on the vehicle by taking power from the nuclear generator primary source or the emergency power source and converting it to a useful power with the proper characteristics for each load item.

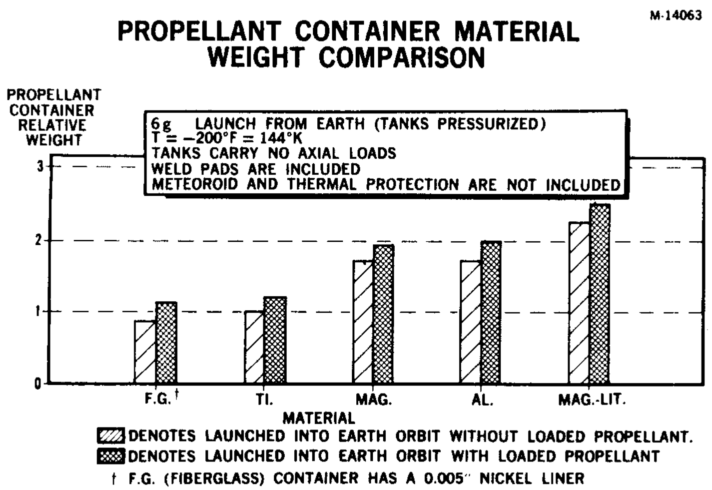
2.7 STRUCTURAL AND MATERIAL DESIGN

Some of the broad design concepts incorporated into the Mars vehicle configuration are as follows: (1) all stages are to form an integral unit when launched, with no orbital assembly; (2) LH_2 in the tanks during launch will be sufficient only for cooldown and pressurization, thereby achieving a near-minimum gage tank design; stages will be fueled in orbit by transferral of LH_2 from tankers; (3) fuel tanks with or without engines are to be designed as jettisonable units, singly or in pairs, to capitalize on maximum mass fraction improvement; (4) tank domes, valves, fittings, and support methods are to be identical whenever possible to save fabrication costs and/or complexity; (5) the meteoroid shield design must be integrated with the vehicle structural design, thermal insulation, and propellant tank design wherever possible. All must be designed for easy jettison.

A relative weight comparison of the five propellant container materials studied is shown in Figure 28. Of all the tank materials investigated, titanium or fiberglass would produce the lightest propellant tanks. Although titanium tanks appear to be slightly heavier than fiberglass, titanium tanks were used in all vehicle configurations. This decision was based on investigations of the complexities and attendant problems of fabricating (filament-winding techniques) extremely large fiberglass tanks with satisfactorily bonded liner materials such as nickel. However, fiberglass tanks should not be completely disregarded because improvements in technology in the next 15 years could make this material competitive.

The design of the meteoroid shields for various mission durations and vehicle sizes was based on Whipple's 1963A and D'Aiutolo's flux curves and a modified Bjork penetration formula. In this analysis three discrete meteoroid environments were considered; near earth for earth-orbiting vehicles, near Mars for Mars-orbiting vehicles, and deep space for vehicles moving between planetary bodies. Figure 29 shows meteoroid shield thickness of aluminum armor for the three environmental regions

Figure 28



mentioned, as a function of mission duration and vehicle area. An equivalent Whipple-type shield was developed from these studies. Relative K factors (ratio of armor plate thickness to Whipple-type shield), with K=1 for single armor aluminum sheet, were derived from available data. The study indicated that a trussgrid core (K=0.20) was more effective than standard honeycomb core (K=0.27) with open channels. Figure 30 shows Whipple-type meteoroid shields of aluminum and titanium for the same parameters as are illustrated in Figure 29.

Design of the meteoroid shield was integrated with the structural design of the vehicle, thermal insulation, and propellant tank design, so that the shield formed a multipurpose structure.

Figure 29

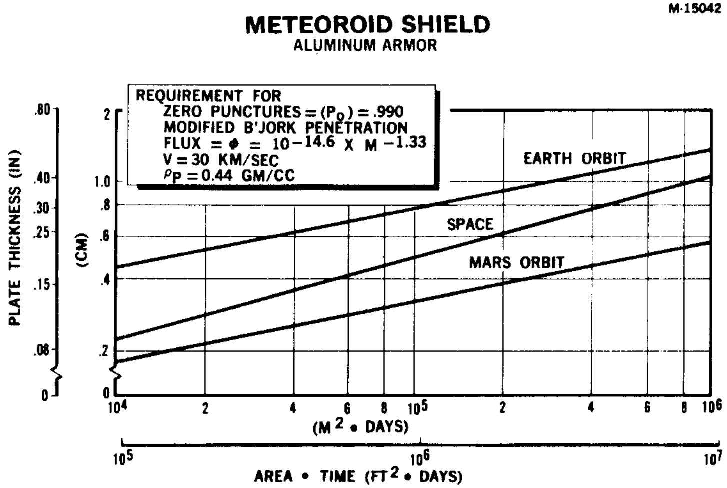
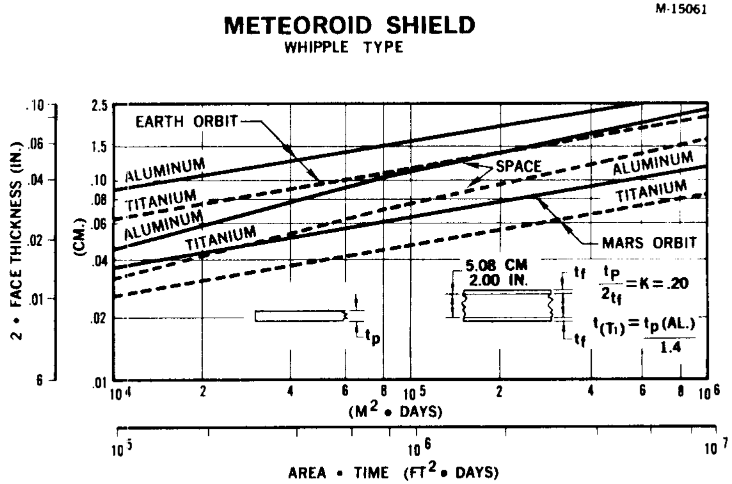


Figure 30



A study of the meteoroid shield requirements, as a function of time, established that the 120-day earth orbits were the most critical period of the entire operation and that any shield designed for this condition would furnish adequate protection during the entire mission. Several configurations of tank and shield combinations studied are shown in Figure 31. The final design selected for the lower stage of the vehicle was a sandwich shell enclosing a multicellular monocoque toroidal tank. This shield also serves as the load-carrying structure for launch from earth-to-earth orbit. A check of the upper stages indicated that the titanium sandwich used for the tank wall and interstage structure provided sufficient protection for outer space and Mars orbit.

Figures 32 and 33 present a typical structural support of thermal protection systems for reentry vehicles, based on calculations previously discussed in Section 2.2.3. The composite material (phenolic refrasil and high-temperature, low-density insulation) is common to all vehicles because it maintains the load-carrying structure at a temperature not in excess of 700°K (800°F). This temperature level allows the thermal protection system to be bonded to the supporting structure and also permits the use of titanium instead of high-density refractory metals.

Figure 34 presents a synopsis of instantaneous and time-integrated propellant boiloff rates. The rates were determined from numerous parametric studies of heat loads imposed on the Mars vehicle surface by solar, planetary, and atmospheric heating. The results are for a typical tank insulation thickness of 80 sheets of NRC-2 high performance insulation. The boiloff and boiloff rates presented are for the following mission phases where the area terms refer to the area subjected to heat transfer only:

1. Earth and Mars Orbit: The vehicle, in earth and Mars circular orbits of 325 km (175 n. mi) and 555 km (300 n. mi), respectively, has its nose oriented toward the sun at all times. During both orbits, constant surface properties of 0.2 solar absorptivity and 0.8 thermal emissivity were assumed. Heat leaks, through the insulation, were assumed to constitute 3% of the propellant tankage heat-transfer area.

2. Interplanetary Transfer: During interplanetary transfer, the vehicle was assumed to be oriented with its base toward the sun. A solar absorptivity of 0.94 and a thermal emissivity of 0.84 were assumed for the base. The sides were assumed to have the same properties as stated in Item 1. During the transfer to and return from Mars, 93% and 97%, respectively, of the heat loss was through the base.

3. Direct Entry to Mars: The propellant loss presented is for the 460-day trajectory, an insulation thickness of 2.54 cm (1 in.) and maximum temperature of 755°K (900°F).

4. Martian Stay: The ground-hold conditions assumed were a constant Mars temperature of 286°K (55°F), a heat-transfer coefficient of 14.42 Joules/sq cm-hr-°K (0.75 BTU/sq ft-hr-°F) and a solar constant of 216 Joules/sq cm-hr (190 Btu/sq ft-hr).

2.8 PROPULSION

The chemical engines considered for use in this program are presented in Figure 35. They represent a modest extension to currently available H_2/O_2 engines, and an advanced beryllium hydride combination, BeH_2/H_2O_2. Such engines are applicable to mid-course connections and fourth-stage propulsion. The two large chemical engines, applicable for launch from earth orbit or retro into Mars orbit, are the high-pressure H_2/O_2 type.

Four nuclear engines were considered: a first-generation reactor, capable of 2.2×10^5N (50,000 lb) thrust; two second-generation engines of the high-thrust graphite-core type and the low-thrust refractory metal-core type, respectively; a fourth, gas-core type, which may not be available until the end of the 1975-85 period.

Figure 31

M-15050

ALTERNATE FORMS OF TANK METEOROID SHIELD STRUCTURAL CONFIGURATION

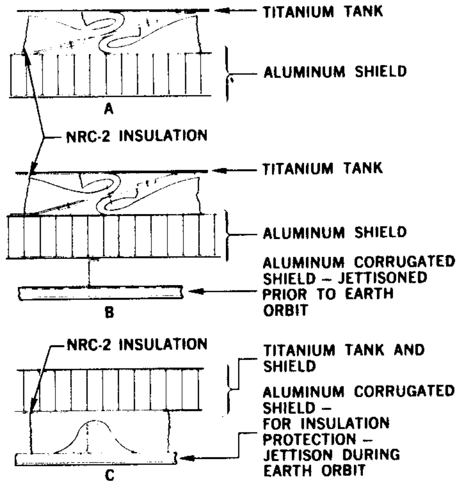


Figure 32

M-15010

MARTIAN ENTRY

THERMAL PROTECTION SYSTEMS

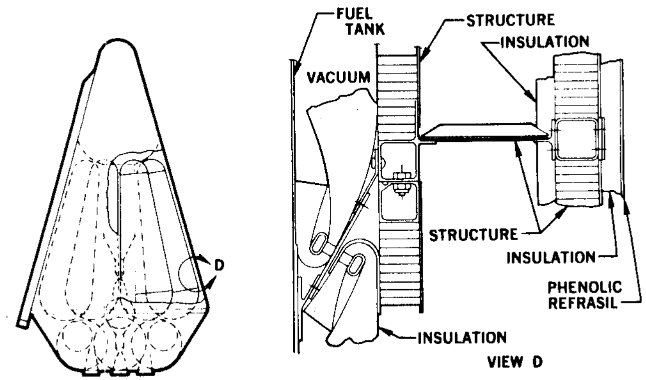


Figure 33

M-15009

EARTH ENTRY

THERMAL PROTECTION SYSTEMS

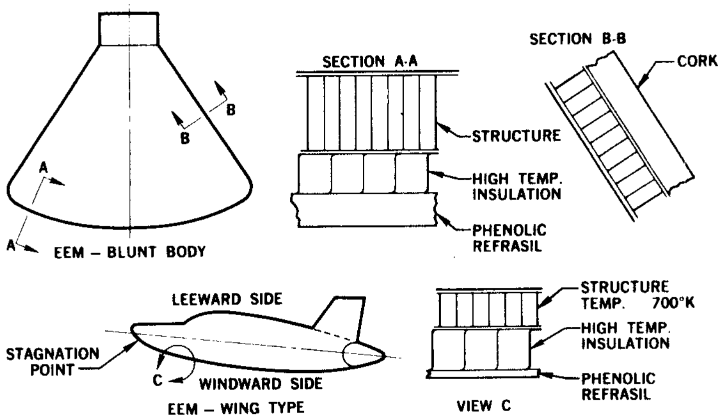


Figure 34

M-14999

PROPELLANT MANAGEMENT

HYDROGEN BOILOFF HISTORY

MISSION PHASE	BOILOFF RATE (GM/CM^2 – HR)	INTEGRATED BOILOFF (GM/CM^2)
EARTH ORBIT		
NO HEAT LEAKS	1.03×10^{-4}	–
3% HEAT LEAK AREA	6.35×10^{-2}	–
INTERPLANETARY TRANSFER		
EARTH TO MARS	–	3.92×10^{0}
MARS TO EARTH	–	1.28×10^{1}
MARS ORBIT		
NO HEAT LEAKS	6.15×10^{-5}	–
3% HEAT LEAK AREA	3.77×10^{-2}	–
MARS ENTRY	–	8.0×10^{-4}
MARS STAY		
SOLAR HEATING INCLUDED	1.29×10^{-3}	–
SOLAR HEATING NEGLECTED	1.0×10^{-3}	–

Figure 35

M-15027

ENGINE CHARACTERISTICS

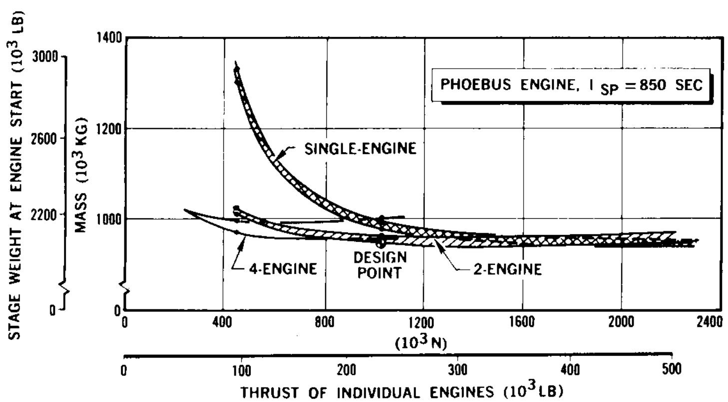
			THRUST 10³N	THRUST 10³LBF	THRUST/WT. LBF/LBM	ASSUMED I_{SP} LBF-SEC/LBM	PROPELLANT(S)	Pc ATM.	SIZE 5m
CHEMICAL ENGINES	SMALL ENGINES	CURRENT	67	15	50	440	H_2/O_2	20	
CHEMICAL ENGINES	SMALL ENGINES	ADVANCED	130	30	75	480	BeH_2/H_2O_2	40	
CHEMICAL ENGINES	LARGE ENGINES		890	200	75	440	H_2/O_2	200	
CHEMICAL ENGINES	LARGE ENGINES		6700	1500	120	440	H_2/O_2	200	
NUCLEAR ENGINES	FIRST GENERATION		220	50	3.5 (UNSHIELDED)	750	H2	35	
NUCLEAR ENGINES	PHOEBUS		1100	250	15 (UNSHIELDED)	850	H2	68	
NUCLEAR ENGINES	METALLIC CORE		130	30	5.5 (UNSHIELDED)	850	H2	68	
NUCLEAR ENGINES	GAS CORE		2200	500	2.3 (UNSHIELDED)	2000	H2	300	

Figure 36

M-15008

THRUST DETERMINATION

FIRST STAGE WEIGHT VS NUCLEAR ENGINE THRUST



The effect of engine thrust level on vehicle weight was determined using an early-configuration vehicle (nuclear propulsion all stages, standard profile, weight of 940 Mg), as a reference point and using appropriate engine thrust-to-weight ratios. Figure 36 presents the effect of first-stage engine thrust level on gross weight for single-engine and clustered-engine propulsion systems. The bandwidths indicate the effects of a factor-of-two uncertainty in reactor core power density. The design point (two 205K engines) is seen to be only 0.4% heavier than the optimum for two-engine propulsion systems and only 0.6% heavier than the lowest value shown (that for a single 500K-engine system). The general flatness of the curves, except for the single engine curve at low thrusts, indicates a considerable freedom of choice of first-stage engine size and number of engines in the cluster. However, when engine reliability is considered, the choice in number of engines becomes restricted.

Figure 37 shows the effect of thrust level on second-stage gross weight for two systems: a single graphite-core engine, and a cluster of four metallic core engines. Again, the design point (one 250K engine) is close to optimum, and the curves are "flat"; all of the values shown are less than 3% above the minimum. Results of the third-stage (design point of two 30K engines) and fourth-stage (design point of one 30K engine) thrust determinations are shown in Figure 38.

Analysis of nuclear engine shielding problems was extended to cover combined core-decoupling and propellant-heating requirements. For all vehicle configurations except one which used a cluster of ten 50K engines, the optimum engine support structure arrangement afforded sufficient separation distance between engines for reactivity decoupling. For the cluster of ten 50K engines, radial decoupling shields were needed for centerline-to-centerline separation distances less than 4 meters (13.2 ft) in the decagon arrangement. The effect of the additional shielding is shown in Figure 39. For propellant heating rates of interest (less than 10^6 watts), the additional shield mass required for decoupling is 1250 kg at a separation distance of 2.5 meters (8.2 ft).

Figure 37

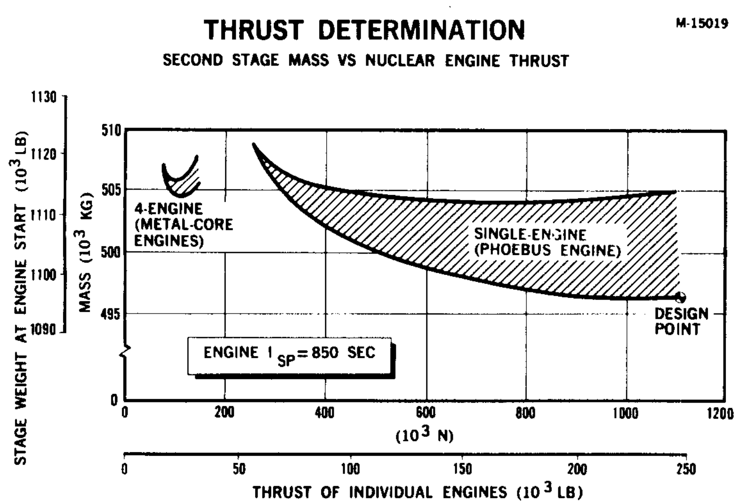


Figure 39

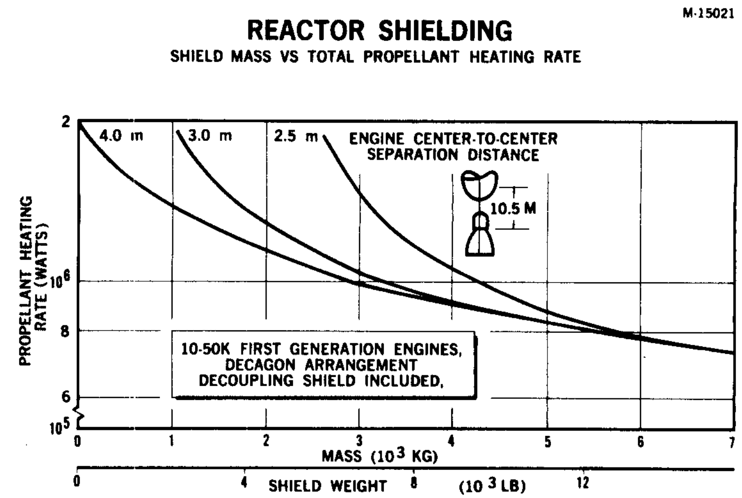
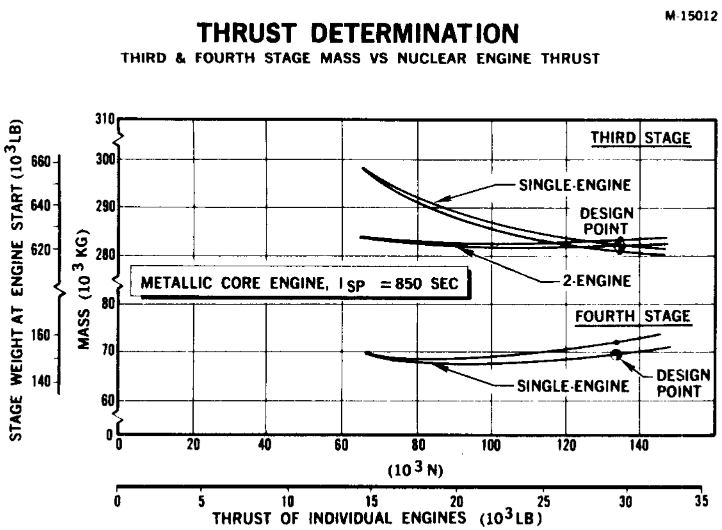


Figure 38



An advanced chemical propulsion system using BeH_2/H_2O_2 has been considered for fourth-stage propulsion. Primary reasons for consideration are: the use of both BeH_2 and H_2O_2 as space radiation shields, since both have excellent radiation-attenuation properties; the comparatively high specific impulse; the high bulk density; and storability. When BeH_2/H_2O_2 and nuclear systems are compared in fourth-stage operation, both are comparable with regards to space radiation shielding capability but the nuclear systems show an advantage from a specific impulse standpoint. The BeH_2/H_2O_2 system, however, does not require engine shielding and consequently exhibits a higher stage structural efficiency (λ'). System selection is based on the velocity requirements of the fourth stage. BeH_2/H_2O_2 is more desirable for lower velocity requirements while nuclear systems are compatible with the higher velocities.

In addition to the primary propulsion systems, small thrustor systems are required to orient the stages prior to maneuvers and to overcome roll, pitch, and yaw torques generated internally. The thrustors may be chemical or electrical and, for low-thrust levels (1 to 2 Newtons), an arc jet or resistojet of the type under development would be satisfactory. Chemical thrustors may also be required for propellant-settling prior to start of the main engines. Acceleration levels on the order of 10^{-3} g's are required for the second stage, for periods up to 100 sec, unless passive propellant orientation techniques (surface tension, dielectrophoresis) can be used.

Section 3
WEIGHT-SAVING TECHNIQUES

It is possible in some instances to increase the vehicle's performance by applying techniques that decrease the initial gross weight in earth orbit. Several such weight-saving techniques have been investigated in the areas of astrodynamics, propulsion, propellant management, structures, materials, and payload systems, and their effectiveness will be discussed.

3.1 ASTRODYNAMIC TECHNIQUES

From an astrodynamic standpoint, there are several interesting techniques that might provide additional vehicle performance or a saving in initial weight in earth orbit. Some techniques, such as lunar-based vehicles, use of the moon's gravitational field to decrease incremental velocity requirements, and orbit-altitude decay to alleviate orientation problems at Mars departure, are primarily of academic interest and will be discussed only in Volume IV. Several others, however, have real potential.

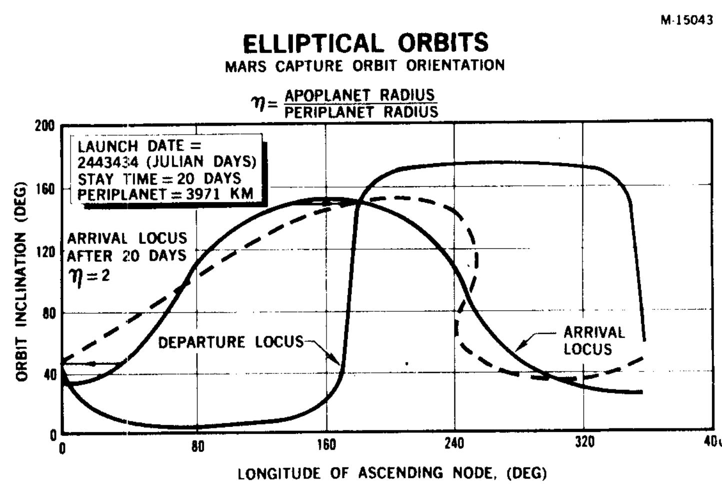
3.1.1 Elliptical Parking Orbits at Earth and Capture Orbits at Mars

The transfer from an elliptical parking orbit to a departure hyperbola, if accomplished at the perigee position of the parking orbit, minimizes the incremental velocity required to achieve the departure hyperbola. Although a sizable velocity saving can be obtained, this velocity is certainly not "free" because earth launch vehicles now require a greater velocity capability to place the same payload in the higher-energy elliptic orbit. Problems associated with space vehicle rendezvous in elliptical orbits, in-orbit vehicle maintenance, logistic support, etc., complicate the system but should be feasible in the time period under consideration.

The theory of using elliptical capture orbits at Mars logically follows that for earth, i.e., the larger the periplanet velocity of the capture orbit, the less incremental velocity required to transfer from the approach hyperbola to the planetary capture orbit. As with earth orbits, planetary oblateness complicates the situation as the vehicle's orbit inclination directly affects nodal regression rate. Figure 40 indicates the variation of orbit inclination with ascending node for the approach and departure orbits for a specific launch case. The nodal regression experienced by the capture orbit during the planet stay time is indicated by a dashed curve. A compatible inclination is obtained by the intersection of the dashed curve with the departure orbit requirement. To eliminate (or at least minimize) plane change and apsidal rotation requirements, which are costly from an energy standpoint, all three orbits (approach, capture, and departure) must have identical periplanet positions.

For some arrival and departure conditions this can be obtained by proper selection of periplanet altitudes and capture-orbit eccentricities so that the apsidal advance

Figure 40

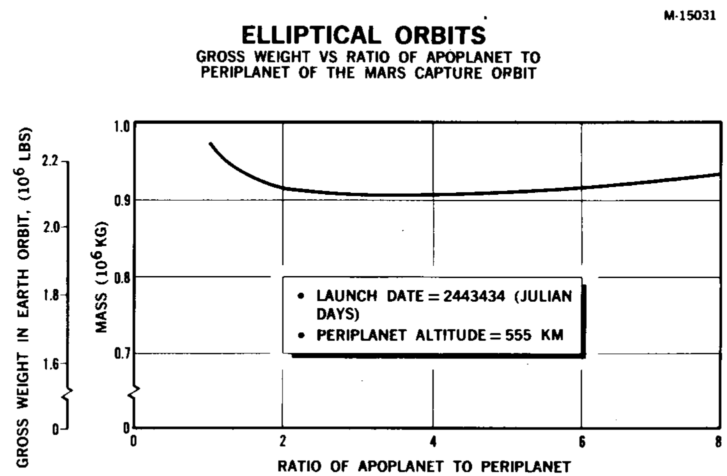


due to oblateness results in a compatible position. In other situations the only solution is to rotate the line of apsides to the proper position by means of an incremental velocity. A representative mission of 460-day duration and 20-day Martian stay time was used to indicate the variation of orbit inclination and incremental velocity required to make the capture orbit compatible with departure conditions. Although the inclination is nearly constant with apoplanet-to-periplanet radius ratio (η), the velocity required to rotate the apse experiences a large variation because nodal regression is greatly affected by orbit eccentricity.

The incremental velocity savings and penalties as a function of capture orbit eccentricity were obtained by re-evaluating the vehicle weight to account for the larger velocity requirements associated with elliptical orbits, and then recalculating the gross weights. Figure 41 shows this weight comparison and indicates that small weight savings (6%) can be obtained with both large and small values of η, the maximum saving occurring at (η) = 2 through 4. A decrease in periplanet altitude will result in somewhat larger weight savings. Since capture orbit inclination is a vital factor in velocity requirements, several missions were investigated for 1979, 1981, and 1983. Results indicate a wide variation of inclination; hence, the majority of missions require additional velocity to relocate the apse position of the capture orbit.

Weight reduction by use of elliptical capture orbits is relatively small when other system interactions are considered. Although no insurmountable problems exist, incorporation of these interactions into a baseline mission profile will cause increased costs, development problems, and increased risk, with a small gain in performance which can be appreciated only a portion of the time. It is therefore recommended that this technique be used only in isolated circumstances when additional vehicle capability is mandatory.

Figure 41



3.1.2 Aerodynamic Attainment of Martian Capture Orbits

A capture orbit at Mars can be obtained by employing aerodynamic braking, thus eliminating a large portion of the velocity requirements of a propulsive braking system. This is accomplished by a programmed maneuver during atmospheric entry; this results in atmospheric exit conditions that allow the attainment of circular or elliptical orbits with only a small additional velocity increment.

This method can result in a weight saving on the order of 20% for typical missions. The allowable entry corridor for this type of operation is the same as with a direct entry but requires a more complex and highly developed maneuver control program because pullout locations and initiation times are critical to the attainment of proper exit conditions.

3.1.3 Direct Mars Entry

The direct entry mode of operation differs from the orbit attainment mode in that the entire vehicle is landed on the surface of the planet. The aerodynamic braking maneuver is used to decrease the velocity to an acceptable level for a powered letdown to the surface, thus eliminating a separate Mars Excursion Module as well as the large propulsive requirements at planetary approach. However, two powered boost phases are now required, one to achieve soft landing and another to depart from the surface of the planet. These additional operations, combined with entry and exit aerothermodynamic requirements, result in a weight increase of approximately 100%.

3.1.4 Super-Escape Earth Entry

Earth entry at velocities in excess of 12.2 km/sec (40,000 ft/sec) appears desirable from a weight and vehicle performance standpoint. Earth entry speeds up to 18.3 km/sec (60,000 ft/sec) were investigated with regard to thermal requirements, structural integrity, aerodynamic loads, and applicable materials, with the result that a decrease in weight with increasing entry velocity is experienced (Figure 42). This trend is due to the desirable thermal protection properties of composite materials whose increase in weight with entry velocity is less than the propellant weight saved using these large entry velocities. This technique appears desirable from a weight standpoint but bears further investigation from the composite material standpoint.

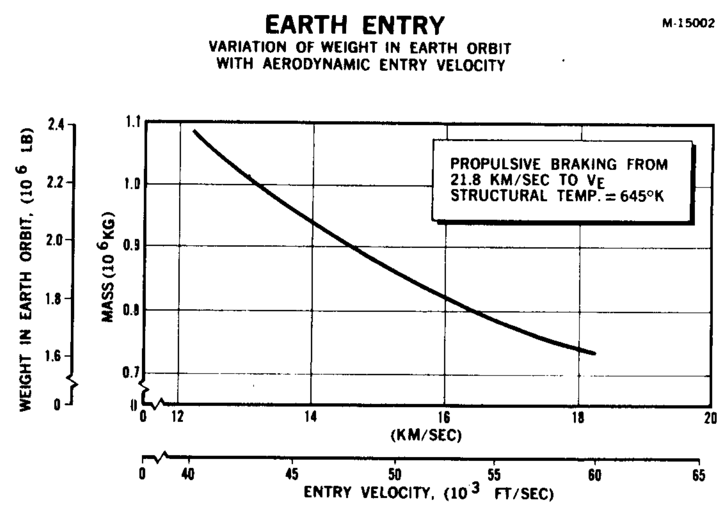
3.1.5 Mid-Course Plane Changes

Select combinations of launch date and mission duration result in large velocity requirements for both one-way and round-trip Mars missions in all synodic periods. Nearly all of this excess energy requirement can be eliminated by applying two plane changes, one at departure and the second in the region of the line of nodes. If a mission is scheduled for this sensitive area, the mid-course plane change maneuver can be employed to yield energy requirements commensurate with those of less sensitive launch date regions.

3.1.6 Long-Duration Missions

Minimum energy requirements for one way earth-to-Mars and Mars-to-earth transfer usually occur for transfer times of 200 to 220 days, indicating that 400-to-500-day total mission durations might be optimum from an energy standpoint. For most years these minimum-energy transfers are not compatible, i.e., the return launch must be made before the vehicle arrives at the planet. However, desirable low-energy transfers can be obtained for the earth-to-Mars and Mars-to-earth legs if long-duration missions of 800 to 900 days are allowed. This type of transfer still uses a one-way transit time of 200 to 220 days, but now the vehicle stays in the vicinity of the planet for 300 to 400 days and returns to earth during the next synodic period. Taking advantage of desirable energy requirements available at both the earth and Mars departures, a saving in weight of more than one-half can be obtained as compared to the short-duration missions. Unsolved problems still exist in the prediction of solar

Figure 42



events, necessary shielding requirements, human confinement, etc., but these problems are common to all interplanetary missions. Although the problems may be more severe for long missions, the low weight requirements still make such long missions attractive.

3.1.7 Vehicle Staging

Interplanetary missions are composed of four distinct phases: earth departure, Mars approach, Mars departure, and earth approach prior to re-entry. A vehicle system consisting of four stages becomes a natural first choice, particularly if a nuclear system is considered in which the engines are discarded after one use. Improvements in performance were sought through the use of staging within each phase of the mission profile. Two of the methods investigated for each phase incorporate series and parallel staging in which both tanks and engines are jettisoned at each staging point. If complete freedom in the choice of engines is available, performance gains of 5% to 7% can be realized. However, for specific engine systems, weight penalties may be experienced if engine weight is too large for the particular propellant loading. An alternate method of staging that may be advantageous consists of dividing the required propellant into several tanks and jettisoning just the propellant tanks as the propellant is depleted. This method yields a 5% performance gain.

Duplicate staging was investigated in an effort to reduce manufacturing and development costs of the vehicle system. In this approach, combinations of upper stages were used for the lower stages, e.g., the first stage consisted of two second stages fired in parallel, resulting in a slight weight penalty of approximately 1% to 2%. The beneficial aspects of this approach, such as lower development costs and higher reliability, necessitate further analysis to determine the true effectiveness of this approach.

The largest performance gain was obtained with a mass fraction improvement scheme which discards the meteorite protection and thermal insulation shield of each stage just prior to stage ignition. A performance gain of 38% is possible with this technique.

3.1.8 Additional Theories

A method requiring additional investigation is the reorientation of astronomical configurations to allow frequent recurrence of low-energy transfers. Since the position of earth and Mars cannot be changed, the vehicle launch point must be changed to a more suitable orientation by departing from the earth in an interim orbit, preferably in the ecliptic plane, and then changing the entire transfer orbit when the new energy requirements are lower. This method is most effective on the return orbits where the new transfer orbit can be initiated at aphelion, but it can also be used in particular classes of earth-departure orbits as well. Reducing vehicle velocity during perihelion passage is effective in reducing arrival velocities at the target planet. As expected, this method becomes more effective as the perihelion passage distance is reduced. Further detailed analysis is required to determine the usefulness of this technique because thermal protection requirements, additional staging, and environmental control requirements may impose severe penalties on the vehicle system.

3.2 PAYLOAD WEIGHT-SAVING TECHNIQUES

A number of weight-saving techniques relating to the life support systems and man-vehicle interfaces were investigated to determine their effect on vehicle weight reduction and to determine any increase in risk to the crew or mission. The techniques, with the resulting savings and associated increase in risk, are summarized in Figure 43. Column 2 shows the savings of the particular system; Column 3 indicates the change in vehicle gross weight.

Figure 44 indicates the caloric input requirements of 5- to 95-percentile men to perform the tasks of a Mars mission. A crew composed of 20-percentile men, rather than 90-percentile men would save 0.1 kg (0.22 lb) of dry food per man per day. Actually, a larger weight saving is experienced when one includes the requirement for less clothing weight, smaller personal equipment, smaller body mass, less structure to provide the same internal volume, less shielding, smaller biowell, and vehicle growth factor. A saving of approximately 5% of

Figure 43

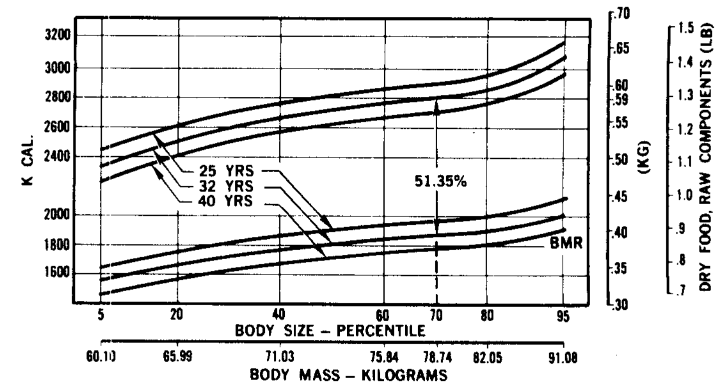
WEIGHT SAVING TECHNIQUES
PAYLOAD INTERGRATION SUBSYSTEMS

TECHNIQUE	WEIGHT REDUCTION (KG)		RISK
	TECHNIQUE	GROSS VEHICLE	
1 20TH PERCENTILE MAN (90TH TO 20TH)	2,279	31,600	NONE
2 COMPARTMENTATION	NONE	-	NONE - SOME ADDITIONAL SAFETY
3. REDUCED CREW SIZE (6 TO 5)	3,790	52,600	SMALL - ONLY DURING MARTIAN EXPLORATION AND MAINTENANCE
4. MARS N_2 COLLECTION	NONE	-	SMALL - RETURN ON 100% O_2
5. TANK SAFETY FACTOR (FROM 3 TO 1.5)	150	1,020	SMALL - POSSIBILITY OF FAILURE FAILURE WOULD BE CATASTROPHIC
6. HABITABLE VOLUME (11,000 FT^3 TO 4500 FT^3)	750	5,130	UNKNOWN PSYCHOPHYSIOLOGICAL EFFECTS
7. HEMETIC SEALING OF VEHICLE HATCHES	NEGLIGIBLE	-	DIFFICULT TO EGRESS
8. METABOLIC INTAKE (10% REDUCTION IN K. CAL.)	183	1,240	PRODUCE BEHAVIORAL DECREMENTS
9. SPACE SUITS (50% OF TIME IN SUIT)	56	380	INCONVENIENT - NOT SHIRT-SLEEVE ENVIRONMENT
10. 100% O_2 AT 3.5 PSIA	200	1,360	INCREASED FIRE HAZARD
11. SYSTEM RELIABILITY			
(.972 TO .958)	59	400	1.5 TIMES AS GREAT
(.972 TO .944)	150	1,020	TWICE AS GREAT
12. CONTINUOUS ZERO GRAVITY			
CENTRIFUGE	440	3,000	COMPARATIVE RISKS INDETERMINATE
ROTATING VEHICLE	9,200	62,600	
13. METABOLIC RATES (DRUG INDUCED)	UNKNOWN	-	CREW INCAPACITATION
14. ALLOWABLE ACCELERATION	UNKNOWN	-	INCREASED PHYSIOLOGICAL RISK
15. RADIATION SHIELDING	UNKNOWN	-	INCREASED PHYSIOLOGICAL RISK

Figure 44

M-14041A

PERCENTILE MAN'S CALORIC INPUT



gross vehicle weight is obtained with the 20-percentile crew; of equal importance is the fact that no additional risk is involved.

Compartmentation was considered in an effort to save the gas weight lost by meteoroid penetrations. Based on the frequency of penetrations anticipated for this vehicle and mission, this concept is not beneficial but would be if the frequency of penetration was much larger or if material weight of the compartments was reduced. Reducing the crew size from the desired six men to five would save food, personal equipment, smaller volume cabin, structure, and shielding weight. Although nitrogen gas collection from the Martian atmosphere eliminates transporting the gas to Mars, the MEM vehicle must now carry additional equipment to collect, liquefy, and store the required nitrogen for the return trip, resulting in no weight saving.

3.3 STRUCTURAL AND MATERIAL DESIGN WEIGHT-SAVING TECHNIQUES

Throughout the study many alternative designs were considered and studied in areas of artificial gravity, space-radiation protection, meteoroid shielding, and thermal insulation. The primary purpose was to achieve a satisfactory operating system, meeting all the design criteria, and yet be of minimum weight. The following discussion covers a few of these areas.

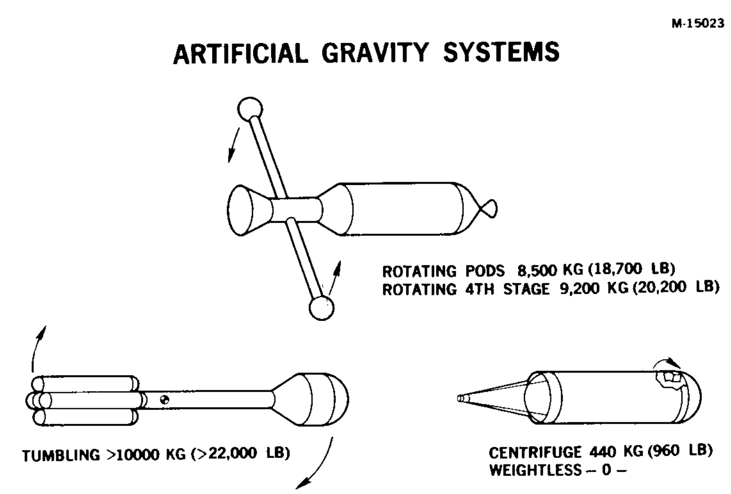
3.3.1 Artificial Gravity

Parametric studies were made of various methods for developing an artificial gravity environment. Figure 45 presents five basic methods that were analyzed, as follows:

1. Rotating Pods: Vehicle stationary; in brief, this concept has two spherical pods, 4 meters in diameter and approximately 50 meters apart, rotating at 4 rpm around the main body of the vehicle. The connecting tubes serve as structural support and access to the pods.
2. Rotating Entire Fourth Stage: Design is similar to (1) except that the pods are rigidly attached to the fourth stage.
3. Tumbling the Entire Vehicle: Artificial g was induced by rotating the vehicle about its center of mass.
4. Centrifuge: The centrifuge design consisted of a toroid tank enclosed within the vehicle as an integral unit of the payload.
5. Weightlessness.

Whereas a uniform constant g is maintained in the rotating g pod-system, its large weight and problems of maintaining an airtight seal make this concept less attractive than the others. Rotation of the fourth stage solves the sealing problem but creates high coriolis problems when the crew is in the command center in the fourth stage. The major drawback of the tumbling concept is that the distance from the vehicle center of gravity continuously decreases as vehicle stages are jettisoned; hence, additional structure in the fourth stage is required to maintain sufficient distance between the manned cabin and the center of gravity. The centrifuge system, being ~ 8000 kg lighter than the rotating pod system, has attractive features such as light weight, ease of design, good access, easy maintenance, and no antenna problems.

Figure 45



3.3.2 Biowell

Various space-radiation protection compartments (biowells) were investigated based on possible design concepts, material effectiveness, material weight, and material utility. Vehicles that require an earth retro stage contain either LH_2 for nuclear propulsion or chemical propellant for chemical propulsion. Incorporation of the fourth-stage propellant requirements into a biowell design, for solar flare protection, minimizes shield

weight. Considering the attendant fourth-step growth factors, substantial total vehicle weight savings can be realized. Figure 46 presents a typical biowell arrangement in which the LH_2 propellant is used as shielding. The system shows a blunt-body Earth Entry Module (EEM) as part of the integral unit. If a winged EEM system is used for entry, its size is not compatible with the biowell concept; hence, an alternative biowell design which shields minimum volume and uses propellant or some other solid shield (e.g., borated polyethylene) must be chosen. In either case, a minimum of 183 cm (6 ft.) of LH_2 or 23 cm (9 in.) of polyethylene was considered sufficient protection. This yields approximately 25 to 30 gm/sq cm of space radiation shielding when structural materials and ecological materials are considered in conjunction with the above shields. Shield thicknesses were based on shielding analysis of space and onboard nuclear radiation sources. The solar flare model was that of November 12, 1960 (3+). The number of such events for various mission times was calculated using a frequency of occurrence of one per year and Poisson distribution. Figure 47 presents the biowell configuration mass comparisons.

3.3.3 NAPU Location Weight Comparison

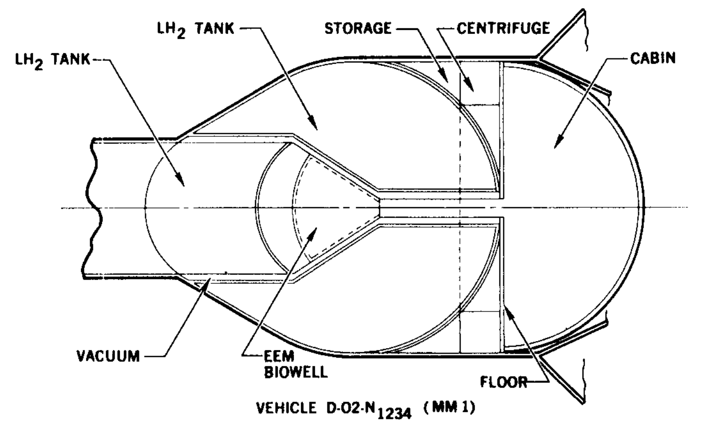
Proper placement of the nuclear auxiliary power units (NAPU) allows substantial weight savings to be made. In brief, the NAPU units, when positioned on a boom forward of the Mars vehicle, require less shielding than when nested among the fourth-stage propellant tanks. The type of artificial gravity system used also has a marked effect on the total shield weight. This is principally due to the conical shadow-shield design required for rotating pod type of artificial gravity.

An additional 3000 kg (6600 lb) must be added to the NAPU shield if the direct Martian entry mode is used, because of the scattering effects of the Martian atmosphere.

Figure 46

M-15022

BIOWELL AREA



VEHICLE D-02-N_{1234} (MM 1)

Figure 47

M-15013

BIO-WELL CONFIGURATION WEIGHT COMPARISON
(ASSUMES 4TH STEP RETRO)

ITEM/DESCRIPTION	EFFECTED MASS (KG)		EFFECTED WEIGHT (LB)	
EARTH ENTRY MODULE TYPE	APOLLO-TYPE		WING-TYPE	
SHIELD MATERIAL	LH_2	POLY-ETHELYNE	LH_2	POLY-ETHELYNE
RELATIVE LSS PENALTY	(890)	(4,350)	(1,950)	(9,600)
RELATIVE 4TH STEP PENALTY	(0)	(4,350)	0	(9,600)
TOTAL RELATIVE BIO-WELL PENALTY	(890)	(8,700)	(1,950)	(19,200)
Δ LSS PENALTY	0	3,460	0	7,650
Δ 4TH STEP PENALTY	0	4,350	0	9,600
Δ BIO-WELL PENALTY	0	7,810	0	17,250
Δ LSS PENALTY X GROWTH FACTOR	0	23,560	0	52,100
Δ 4TH STEP PENALTY X GROWTH FACTOR	0	259,260	0	572,160
TOTAL Δ BIO-WELL PENALTY	0	282,820	0	624,260

3.3.4 Garage Technique

Meteoroid protection weight savings may be obtained by protecting the entire vehicle with an independent, external shell (garage) while it is in earth orbit. Since the entire shell will be removed prior to earth orbit launch, the stage meteoroid shells will be designed only for space or Martian orbit operations. If a 120-day earth orbit is assumed, the total meteoroid protection requirements decrease by 31% on each stage using this method.

3.3.5 Mass Fraction Improvement

In order to attain maximum mass fraction improvement, the following design techniques were employed:

1. All stages of the vehicle are launched from earth unfueled except for a minimum amount of LH_2 fuel ($\sim$ 5%), sufficient only for cooldown and pressurization. By essentially eliminating propellant G loads on the tank walls and support structure, a near-minimum tank gage and support structure design was effected. This results in a significant weight saving.

2. Jettisoning of the dual-purpose, ground-launch, load-carrying structure of the first stage, prior to leaving earth orbit. The dual-purpose structure incorporates the vehicle load structure, meteoroid shield structure, and thermal insulation. It is, in fact, the previously mentioned garage.

3. Jettisoning of individual stage meteoroid shields prior to stage engine ignition also affords a weight reduction. Individual stage tankage is unprotected only during engine operation of that stage.

4. Fuel tanks are jettisoned as they become empty and the complete stage (including engine) is separated after each maneuver, thereby obviating cooldown propellant for nuclear engines.

3.4 PROPULSION AND PROPELLANT CONTROL WEIGHT-SAVING TECHNIQUES

The largest weight reduction effected in propulsion systems results from use of a gas-core nuclear reactor for first- and second-stage application. Of the three gas-core types considered (one- and two-cell vortex, and glow plug), the two-cell vortex type appeared the most promising with a specific impulse of 2000 sec and a thrust of 2.2×10^6N (500,000 lb).

Several propellant conservation techniques have been investigated. They include the use of advanced insulating materials and configurations, umbrella shadow shields, active (refrigeration/reliquefaction) thermal control systems, slush hydrogen, etc. Three methods of active thermal control of the liquid hydrogen were evaluated: direct recondensation using Joule-Thompson cycle, indirect recondensation using a helium Stirling cycle and a hydrogen Joule-Thompson cycle, and hydrogen refrigeration using a helium Stirling cycle only. These systems are compared in Figure 48. The Stirling cycle refrigeration system was chosen because of light weight, simplicity, and the relatively advanced state of development of Stirling engine cryogenerators.

Even with active thermal control, a high level of insulating efficiency is necessary if the refrigeration system power requirements are to be kept within reasonable bounds. As illustrated in Figure 49, a refrigeration system shaft power equivalent to the entire output of a SNAP-8 is required during the most thermally difficult phase of the mission (perihelion passage during Mars-earth transit) even with 100 layers of NRC-2 insulating material.

Umbrella-type shadow shields can be used to significantly reduce solar heating of the liquid hydrogen propellant, with consequent large reduction in insulation, boiloff, and refrigeration system mass (Figure 50).

Figure 48

M-15032

HYDROGEN LIQUEFACTION SYSTEM COMPARISON

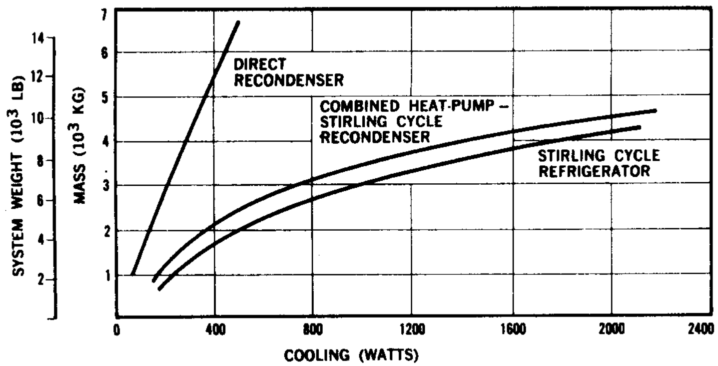


Figure 49

M-15014

PROPELLANT CONSERVATION
LIQUEFACTION SYSTEM WEIGHT VS POWER
(0.4 AU MARS TO EARTH TRANSFER)

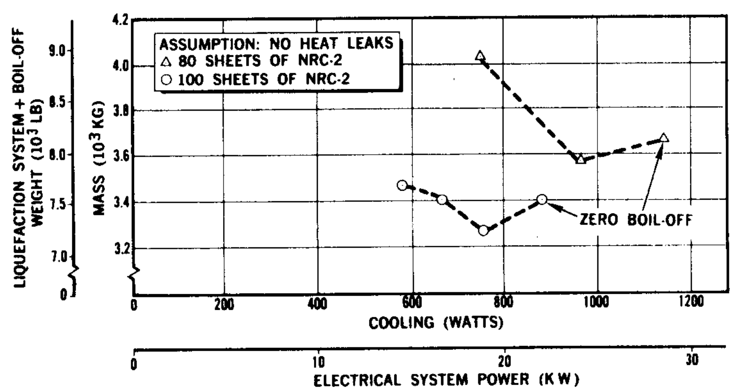
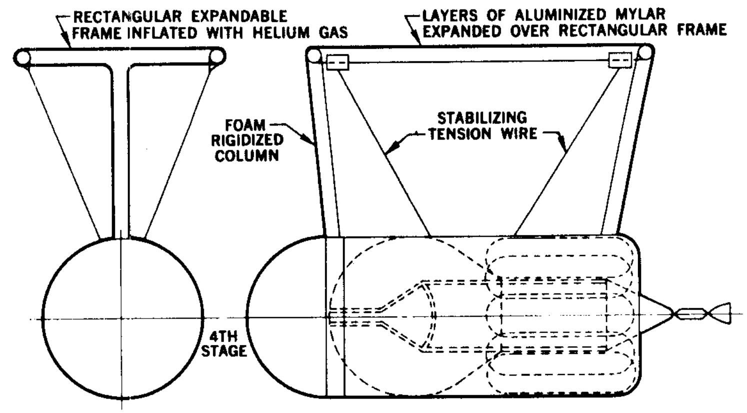


Figure 50

M-15065

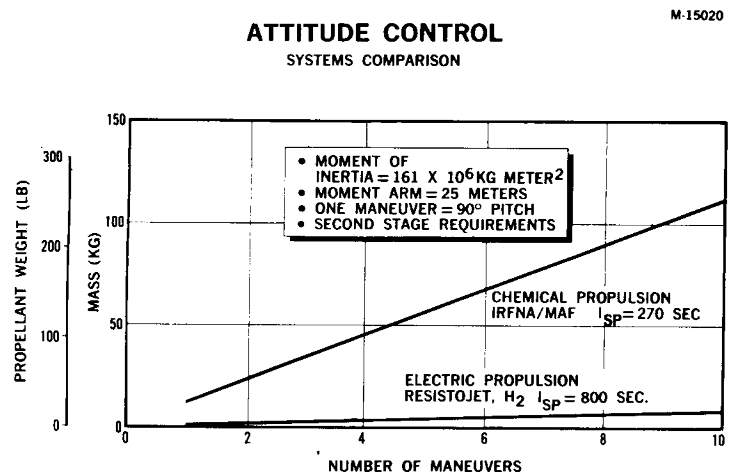
PROPELLANT CONSERVATION UMBRELLA



Hydrogen stored as a slush rather than a liquid also conserves weight. The advantages of a 50% slush over atmospheric liquid hydrogen are due to its higher heat capacity and its 13% smaller tankage requirements. Vented slush can offer a weight saving of 18.5% for long-term storage.

Small, but useful, mass reductions can be effected in the area of reaction control systems by the use of electric-propulsion devices (arc jets and resisto-jets using hydrogen as the working fluid) in place of conventional chemical-rocket systems. These reductions (Figure 51) must be evaluated in the light of the restricted thrust levels available due to electrical power system limitations.

Figure 51


M-15020
ATTITUDE CONTROL
SYSTEMS COMPARISON
• MOMENT OF INERTIA = 161 X 10^6 KG METER^2
• MOMENT ARM = 25 METERS
• ONE MANEUVER = 90° PITCH
• SECOND STAGE REQUIREMENTS
CHEMICAL PROPULSION
IRFNA/MAF I_{SP} = 270 SEC
ELECTRIC PROPULSION
RESISTOJET, H_2 I_{SP} = 800 SEC.
PROPELLANT WEIGHT (LB)
MASS (KG)
NUMBER OF MANEUVERS
0
100
200
300
0
50
100
150
0
2
4
6
8
10

Section 4 MISSION AND SYSTEM DESCRIPTIONS

4.1 PAYLOAD DESCRIPTION

The integrated payload, postulated for the adopted six-man crew, consists of three basic systems: the life support system, the Mars Excursion Module, and the Earth Entry Module. The life support system is composed of payload structure, electronic equipment, scientific payload, radiation protection, power supply, liquefaction system, centrifuge, oxygen supply, water management, atmosphere-thermal-cabin condition, and food and water management subsystem. The MEM was not subject to analysis, and hence, a fixed weight was assumed. Two EEM's were considered, one capable of entry at 12.2 km/sec (40,000 ft/sec) and another capable of 18.3 km/sec (60,000 ft/sec) entry. The former adheres to blunt-body geometry and was considered a baseline approach in all performance and vehicle design work; the second EEM is a winged vehicle and was treated as a perturbation to the baseline. Figure 52 presents the payload weight breakdown.

Figure 52

M-15006

PAYLOAD WEIGHT SUMMARY

ITEM	MASS (KG)		WEIGHT (LB)	
	BLUNT-BODY	WINGED	BLUNT-BODY	WINGED
EARTH ENTRY MODULE	(5,920)	(12,830)	(13,030)	(28,230)
LIFE SUPPORT SYSTEM	(35,320)	(36,800)	(77,910)	(81,170)
LIFE SUPPORT AND ECOLOGICAL SYST.	6,930	6,930	15,260	15,260
STRUCTURE	9,480	9,480	20,900	20,900
ASTRIONICS	3,000	3,000	6,620	6,620
SCIENTIFIC PAYLOAD	2,600	2,600	5,740	5,740
BIO-WELL PENALTY	890	2,370	1,950	5,210
POWER SUPPLY AND SHIELDING	11,060	11,060	24,430	24,430
LIQUEFACTION UNIT	950	950	2,100	2,100
CENTRIFUGE	340	340	760	760
UMBRELLA	70	70	150	150
MARTIAN EXCURSION MODULE	(25,000)	(25,000)	(55,000)	(55,000)
TOTAL PAYLOAD – 460 DAYS	(66,240)	(74,630)	(145,940)	(164,400)
TOTAL PAYLOAD – 880 DAYS	(71,860)	(80,250)	(158,390)	(176,850

4.2 VEHICLE DESCRIPTION

The baseline mission profile indicated in Section 2 is applied to all vehicle systems: the total mission time is 460 days and the Martian stay time is 20 days. This mission profile is: launch from a 325-km (175-n mi) earth orbit, retro into a 555-km (300-n mi) circular Mars orbit, use of a MEM for Mars landing and boost, launch from the 555-km Mars orbit, retro down to 12.2 km/sec (40,000 ft/sec) earth approach velocity, and then direct superescape entry into the earth's atmosphere using an EEM.

The design philosophy applied to all configurations in this discussion is composed of the following criteria:

1. A vehicle diameter of 21.3 meters (70 ft) is never exceeded. This restriction was imposed so that during boost into orbit the trans-Martian vehicle would be compatible with a post-Saturn launch vehicle.
2. The post-Saturn payload capability is 362,000 kg (800,000 lb).
3. Vehicle height is kept at a minimum out of consideration to possible thrust vectoring control limitations of the launch vehicle.
4. During boost into orbit, propellant tanks on all interplanetary vehicle stages are empty except for a minimum amount of LH_2 for cooldown and pressurization.
5. The vehicle design will not require orbital mating. Propellant transfer in orbit is required.

6. Earth boost acceleration effects on structural weight will be minimized.
7. The weight penalty for propellant tank meteoroid protection will be minimized by separating the meteoroid structure by mission phase and then jettisoning the structure required for a particular phase after phase completion.
8. Propellant tanks on all stages are divided into jettisonable units which are pair-jettisoned as they are emptied.
9. Propellant tanks are constructed of titanium.
10. Nuclear propulsion systems will be jettisoned immediately after reactor shutdown, thereby obviating reactor cooldown requirements.
11. Prior to fourth-stage startup, all payload structure and equipment will be jettisoned, leaving only the EEM and fourth-stage tankage.

Further weight-reduction approaches in the design include incorporating propellant tank insulation and meteoroid protection into one structure and using the meteoroid structure as the earth-boost load-carrying structure, since both can be jettisoned in earth orbit.

Four types of vehicle systems are considered and a description given of their pertinent characteristics. One type is concerned with propulsion system variations while the remaining three represent various mission modes, i.e., Martian operations, earth return operations, and long-duration missions. Vehicle sizing is accomplished by considering each system capable of flights in each synodic period over the entire 1975-1985 launch period. Identification of each vehicle system is based upon the following coding reference:

D	P_2	N_{1234}	(MM1)
Douglas	Configuration Type and Number	Engine Type and Stage Number	Mission Selection
	P - Parallel Staging	B - H_2O_2/BeH_2	MM1 - Multi-mission 460 days
			MM2 - Multi-mission 880 days
	S - Series Staging	C - O_2/H_2	DA - Direct Aero
		F - First Nuclear	OA - Orbital Aero
		G - Gas Core	E - Elliptical Orbit
		N - Phoebus or Metallic	

All vehicles were designed to have multimission capability. No configuration sketches are shown for Modes 7, 9, and 12.

Propulsion System Variation - Vehicle systems and their associated performance and mission characteristics are presented in order of increasing propulsion development requirements (Figures 53 through 55).

1. D-C_{1234} (MM1) (H_2O_2) - This all-chemical configuration is representative of the most conservative propulsion system approach to the problem. Three distinct engines are required on the four stages. As indicated in Figure 55, the dry weight exceeds the post-Saturn payload capability; hence, orbital mating is required.

Figure 53

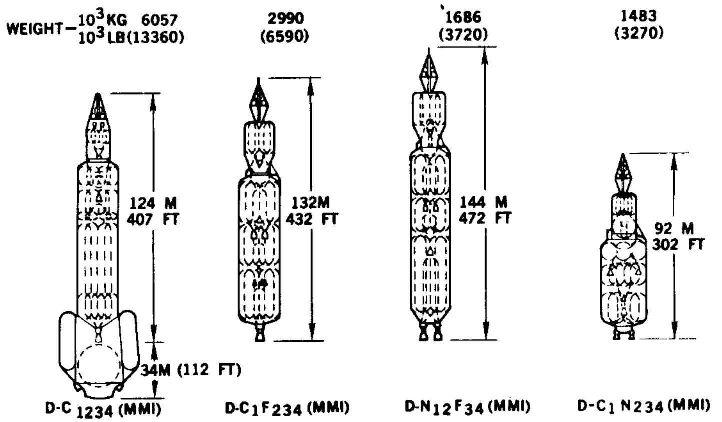
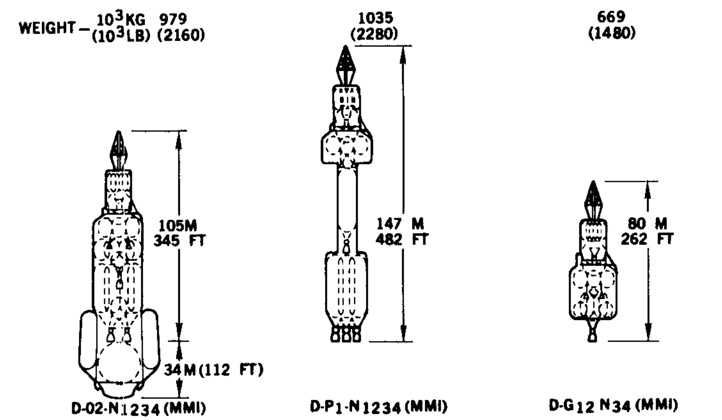


Figure 54



2. D-C_1F_{234} (MM1) - A first-generation nuclear engine, 2.2 x 10^5N (50K), is introduced on the three upper stages of this configuration. Use of this nuclear engine for first-stage application was not considered because of the nuclear clustering problems that arise due to its limited thrust rating. The same 6.66 x 10^6N (1,500K) H_2/O_2 first-stage engine that was used in the all-chemical vehicle, D-C_{1234} (MM1), is also used as first-stage propulsion for this vehicle. As indicated in Figure 55, this vehicle cannot be boosted by a single post-Saturn booster; therefore, it requires orbital mating.

3. D-$N_{12}F_{34}$ (MM1) - With this configuration a large, Phoebus-class nuclear engine is introduced, i.e., a 1.11 x 10^6N (250 K) thrust engine. Use of this large engine is restricted to use on the lower two stages. Propulsion system for the upper stages is the previously introduced first-generation nuclear engines. Again, orbital mating is required because system dry weight exceeds launch vehicle payload capability.

4. D-C_1N_{234} (MM1) - A third nuclear engine is introduced with this configuration. It is a 1.33 x 10^5N (30 K) metallic-core nuclear engine that uses the fast-spectrum principle of operation. Although used on only the two upper stages, inclusion on the second stage will be possible when solutions are obtained to nuclear decoupling problems associated with clustering eight of these engines. No orbital mating is required.

5. D-N_{1234} (MM1) - This configuration is the nominal or standard all-nuclear vehicle. It is composed of a mixture of the Phoebus and metallic-core nuclear engines. No orbital mating is required.

6. D-P_1-N_{1234} (MM1) - The same engines used on the previous system are incorporated into this configuration. However, three Phoebus-class engines are required on the first stage since parallel staging was used with a resulting increase in first-stage weight that required additional propellant. Parallel

Figure 55

MISSION & SYSTEM DESCRIPTION

MODE	SYSTEM	CAPTURE/ENTRY OR ORBIT (MARS)	CAPTURE/ENTRY OR ORBIT (EARTH)	STAGE	THRUST/ ENGINE	NO. ENGINE	TYPE	DRY MASS 10^3 KG (10^3 LB)	GROSS MASS 10^3 KG (10^3 LB)
1	D-C_{1234} (MM_1^*)	RETRO	RETRO-40K	1	1500K	1	H_2/O_2	444	6057
		CIRCULAR	DIRECT	2	200K	2	H_2/O_2	(979)	(13,360)
				3	200K	1	H_2/O_2		
				4	15K	3	H_2/O_2		
2	D-C_1F_{234} (MM_1)	RETRO	RETRO-40K	1	1500K	1	H_2/O_2	462	2990
		CIRCULAR	DIRECT	2	50K	5	NUCLEAR	(1020)	(6590)
				3	50K	2	NUCLEAR		
				4	50K	1	NUCLEAR		
3	D-$N_{12}F_{34}$ (MM_1)	RETRO	RETRO-40K	1	250K	2	PHOEBUS	405	1686
		CIRCULAR	DIRECT	2	250K	1	PHOEBUS	(893)	(3720)
				3	50K	2	NUCLEAR		
				4	50K	1	NUCLEAR		
4	D-C_1N_{234} (MM_1)	RETRO	RETRO-40K	1	200K	3	H_2/O_2	253	1483
		CIRCULAR	DIRECT	2	250K	1	PHOEBUS	(558)	(3270)
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
5	D-N_{1234} (MM_1)	RETRO	RETRO-40K	1	250K	2	PHOEBUS	278	979
		CIRCULAR	DIRECT	2	250K	1	PHOEBUS	(613)	(2160)
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
6	D-P_1-N_{1234} (MM_1)	RETRO	RETRO-40K	1	250K	3	PHOEBUS	319	1035
		CIRCULAR	DIRECT	2	250K	1	PHOEBUS	(703)	(2280)
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
7	D-$N_{123}B_4$ (MM_1)	RETRO	RETRO-40K	1	250K	2	PHOEBUS	269	1028
		CIRCULAR	DIRECT	2	250K	1	PHOEBUS	(593)	(2270)
				3	30K	2	METALLIC		
				4	30K	1	B_eH_2/H_2O_2		
8	D-$G_{12}N_{34}$ (MM_1)	RETRO	RETRO-40K	1	500K	1	GAS	287	669
		CIRCULAR	DIRECT	2				(633)	(1480)
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		

*MM_1 – 460 DAY MISSION 20, DAY STAY

**MM_2 – 880 DAY MISSION, 300 DAY STAY

Figure 55 (Continued)

MISSION & SYSTEM DESCRIPTION

MODE	SYSTEM	CAPTURE/ENTRY OR ORBIT (MARS)	CAPTURE/ENTRY OR ORBIT (EARTH)	STAGE	THRUST ENGINE	NO ENGINE	TYPE	DRY MASS 10^3 KG (10^3 LB)	GROSS MASS 10^3 KG (10^3 LB)
9	D-N_{1234} (MM_1) E_M	RETRO ELLIPTIC	RETRO-40K DIRECT	1	250K	2	PHOEBUS	278 (613)	920 (2030)
				2	250K	1	PHOEBUS		
				3	30K	2	METALLIC		
				4	30K	1			
10	D-$N_1C_2N_{34}$-DA_M	AERO DIRECT	RETRO-40K DIRECT	1	250K	4	PHOEBUS	609 (1340)	1951 (4300)
				2	200	3	PHOEBUS		
				3	30	4	METALLIC		
				4	30	1	METALLIC		
11	D-$N_1C_2N_{34}$-$0A_M$	AERO CIRCULAR	RETRO-40K DIRECT	1	250K	2	PHOEBUS	298 (657)	784 (1730)
				2	200K	1	PHOEBUS		
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
12	D-N_{1234} (MM_1) E_E	RETRO CIRCULAR	RETRO ELLIPTIC	1	250K	2	PHOEBUS	287 (634)	1050 (2320)
				2	250K	1	PHOEBUS		
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
13	D-N_{1234}-DA_E	RETRO CIRCULAR	RETRO-60K DIRECT	1	250K	2	PHOEBUS	245 (541)	738 (1630)
				2	250K	1	PHOEBUS		
				3	30K	2	METALLIC		
				4	30K	1	METALLIC		
14	D-C_{1234} (MM_2^{**})	RETRO CIRCULAR	RETRO-40K DIRECT	1	200K	4	H_2/O_2	175 (386)	1113 (2460)
				2	200K	1	H_2/O_2		
				3	200K	1	H_2/O_2		
				4	15K	1	H_2/O_2		
15	D-$N_{123}C_4$ (MM_2)	RETRO CIRCULAR	RETRO-40K DIRECT	1	250K	1	PHOEBUS	177 (390)	452 (1000)
				2	30K	1	METALLIC		
				3	30K	1	METALLIC		
				4	15K	1	H_2/O_2		

* MM_1 – 460 DAY MISSION, 20 DAY STAY ** MM_2 – 880 DAY MISSION, 300 DAY STAY

and series staging on all stages was attempted in an effort to reduce system weight, increase reliability, and decrease cost. Series staging was attempted with nuclear engines but large weight penalties resulted in all cases. When parallel staging is used with this configuration, the first stage is composed of three identical tanks with the same type of engine, a 1.11×10^6N (250 K) Phoebus class, attached to each tank. Duplicate staging is done by using this same tank and engine (one) in the second stage. The duplicate parallel staging philosophy was also used in the second and third stages and subsequently in the third and fourth stages. Unfortunately, this attempt at uniformity resulted in nonoptimum stage weights in the upper stages. The configuration shown, D-P_1-N_{1234} (MM1), results in the least weight penalty for the parallel-staging vehicles analyzed, since the duplicate-parallel staging is accomplished in the lower two stages. No orbital mating is required in this configuration.

7. D-$N_{123}B_4$ (MM1) - A propulsion combination similar to the two previous configurations with the exception that a beryllium-hydride, hydrogen peroxide (BeH_2/H_2O_2) chemical engine for the fourth stage was analyzed. An BeH_2/H_2O_2 propulsion system was used because the fourth-stage propellant doubles as a space-radiation shield around the biowell, and BeH_2/H_2O_2 possesses attractive proton attenuation capability while serving as a shield; in addition, it has a high specific impulse. When the BeH_2/H_2O_2 system and a nuclear system (metallic) are compared, the nuclear system is advantageous when high velocity is required of the fourth stage, as is the case in MM1. But the situation reverses when there are lower velocity requirements because of the decreased hydrogen available for space shielding in the nuclear system. No orbital mating is required.

8. D-$G_{12}N_{34}$ (MM1) - A twin-cell vortex, gas-core nuclear propulsion system is now introduced, representing the most advanced engine to be considered. Because of gas-core nucleonic limitations, smaller gas-core reactors cannot be considered for upper-stage application. Due to the relatively low gas-core system thrust/weight ratio, use of separate gas-core reactors on both the first and second stage resulted in weight penalties when compared to a single reactor for both stages. It should be mentioned that this two-stage use of one reactor is only applicable to gas core reactors because of the almost complete lack of residual radiation apparent after shutdown in gas-core reactors. No orbital mating is required.

Martian Operations - Four approaches to landing on Mars have been analyzed. The first was specified in the preceding section, i.e., retro into a 555-km circular Mars orbit and then rely on the MEM for actual landing. A second approach uses propulsive braking to achieve an elliptic rather than circular orbit. The two remaining methods use aerodynamic braking; one accomplishes a direct landing on the Martian surface and the other aerodynamically attains a Martian orbit. The mission selected for these systems is the same as for all the previous vehicles.

The vehicle design philosophy for the elliptical orbit mode is exactly the same as that presented for propulsion system variations (see the preceding Propulsion System Variation paragraph) except that larger propellant weight is required in the MEM because of the greater deorbit and boost requirements associated with the elliptical orbits. All upper-stage design is influenced by the aerothermodynamics of Martian aerodynamic braking and launch and not strictly by structural optimization. Martian operations configurations are shown in Figure 56 and their associated performance and mission characteristics in Figure 55.

9. D-N_{1234}(MM1)E_m - This all-nuclear vehicle resembles the previously described nominal or standard all-nuclear vehicle, D-N_{1234}(MM1), with the exception that the velocity capability of the second and third stages is perturbed to accommo-

date a Martian elliptical capture orbit with an eccentricity of 0.6 and periplanet altitude of 555-km (300 n. mi). The MEM size is increased to accommodate the elliptical orbit. No orbital mating is required.

10. $D\text{-}N_1C_2N_{34}(MM1)DA_m$ - The mission profile for direct Martian entry is one where the complete vehicle enters the Martian atmosphere, aerodynamically brakes to terminal velocity, then propulsively lands on the Martian surface. The landing tankage is jettisoned prior to launch from the surface, the vehicle is launched into the specified Martian orbit, and then returns to earth. Since there is no actual propulsive stage required in attaining the Mars surface (excluding terminal conditions), the Mars-boost stage now becomes the second stage and the Mars-orbit departure stage the third stage. Use of a Mars parking orbit prior to Mars escape was selected rather than direct escape because of the larger launch windows associated with the parking-orbit concept.

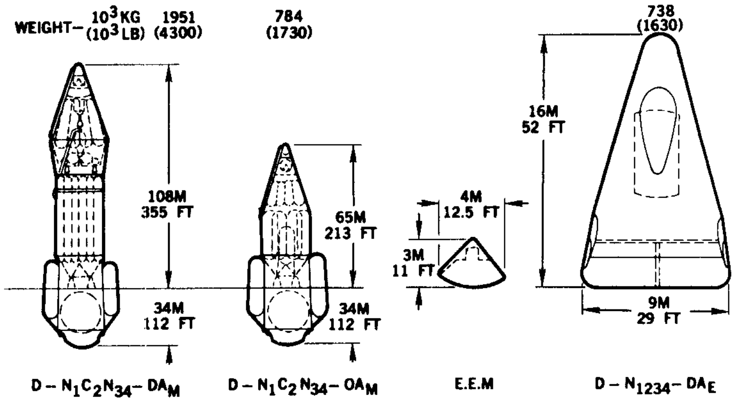
The Mars entry velocity of 9.14 km/sec (30,000 fps) influences vehicle geometry since the radiation mode of heat transfer is predominant at greater than escape velocity in Mars atmosphere. This predominance dictates the use of a slim nose body; therefore, a cone shape was adopted because it also afforded symmetry for tankage design. The entry thermal protection system is a phenolic refrasil covering a high-temperature, low-density insulation and is jettisoned prior to boost. The most critical design problem for this operation is not the entry itself but that of propellant conservation while on the surface. Conservation was accomplished by vacuum-jacketed super-insulation about the individual tanks. Nuclear engines were not used in the Mars atmosphere because of radiation scattering and contamination. Orbital mating is required due to high system dry weight.

Figure 56

CONFIGURATIONS

MARTIAN OPERATIONS & EARTH APPROACH

M-15049



11. $D\text{-}N_1C_2N_{34}(MM1)OA_m$ - The mission profile for aerodynamic attainment of Mars orbit requires a launch from earth orbit; direct entry into the Mars atmosphere, followed by aerodynamic deceleration down to a velocity slightly greater than satellite velocity; aerodynamic pullup out of the atmosphere; and coast to apocenter. At apocenter 555 km (300 n. mi) the small velocity increment required for a circular orbit is achieved propulsively. A MEM is then used for landing operations. The return-to-earth portion of the mission is the same as that of the baseline profile.

Since the design philosophy for this system again depends upon the entry aerothermodynamic environment, a cone shape was adopted for the entry configuration. Since the entry trajectory for aerodynamic orbit attainment and direct entry is essentially the same down to satellite velocity, the maximum heat rates of each will be the same, and the only thermodynamic difference is that of total

integrated heat. As a result, the entry thermal protection material is the same in both cases. Its thickness, however, varies as the total integrated heat; consequently, a phenolic refrasil covering a high-temperature, low-density insulation is used for this system. No orbital mating is required.

Earth Return Operations - Three methods of earth return are considered. The nominal or standard mode previously described consists of propulsive braking to an earth entry velocity of 12.2 km/sec (40,000 fps) and aerodynamic entry to the surface. The second method is totally reliant on propulsive braking, in that the vehicle does not return to the earth surface, but rather retros into an elliptical orbit and waits for rendezvous by an earth-launched rescue vehicle. In this mode, the EEM need not be carried throughout the entire mission, thereby reducing payload weight. The third mode, which relies on aerodynamic braking at velocities greater than 12.2 km/sec, was analyzed with entry velocities up to 18.3 km/sec (60,000 ft/sec), which is the velocity presented for comparison purposes. The propulsion systems are the same as the standard all-nuclear vehicle. Earth-return operation configurations are shown in Figure 56, and associated performance and mission characteristics in Figure 55.

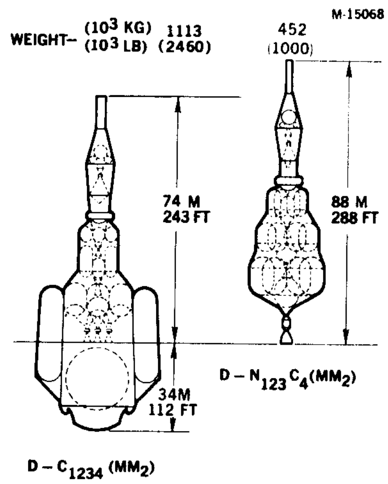
12. $D\text{-}N_{1234}(MM1)E_E$ - The mission profile for this system differs only from the nominal during the earth-return retro maneuver. Complete propulsive braking is used to obtain an earth elliptical parking orbit whose perigee is 555 km (300 n. mi) and eccentricity is 0.6. The earth-return module decreases in weight by 54% because aerodynamic braking is not required. Since the nominal mission selected exhibits a high earth-approach velocity, the velocity requirement for this retro operation is now 12.4 km/sec (40,600 ft/sec), as compared with 10.01 km/sec (32,800 ft/sec) for the standard mode. When the standard all-nuclear propulsion systems were used to size the vehicle, it was found that the gross weight requirement for this velocity increased by 7% when compared to the nominal profile. However, weight savings can be obtained with this approach if the fourth stage is series-staged into a fourth and fifth stage or when missions are selected with a lower fourth-stage velocity requirement.

13. $D\text{-}N_{1234}(MM1)DA_E$ - Retro propulsion is used to reduce the earth approach velocity of 21.65 km/sec (71,000 ft/sec) to 18.3 km/sec (60,000 ft/sec) and an aerodynamic entry is accomplished from 18.3 km/sec (60,000 ft/sec) to the surface. The vehicle selected for the high-speed entry is a winged body that utilizes a blunt half-cone modified to improve aerothermodynamic characteristics. The modifications included: (1) rounding of the sides of the half-cone to increase aerodynamic stability, (2) addition of a base fairing (boat-tail) to reduce base drag, (3) use of full span elevons which are deflected as a single unit at extremely high velocity and separately in the low hypersonic-supersonic regime, (4) swept leading and trailing edge vertical control surfaces which are toed in 5°, and (5) a sharpened nose to account for radiative heat transfer. The entry thermal protection system is a phenolic refrasil, covering a high-temperature, low-density insulation. No orbital mating is required when this winged body is incorporated into vehicle design.

Long-Duration Missions - The long-duration missions (MM2) use the same baseline mission profile previously described except that the mission duration is now 880 days, 300 of which are spent on the Mars surface. Payload, meteoroid shield, and shielding weights are increased because of the increased duration. Design philosophy is the same as that for the 460-day mission with the exception that propellant is boosted into orbit with the interplanetary vehicle in special tanks below the vehicle. Long-duration mission configurations are shown in Figure 57 and their associated performance and mission characteristics in Figure 55.

Figure 57

CONFIGURATIONS
LONG DURATION MISSIONS



14. D-C_{1234}(MM_2) - This all-chemical (H_2/O_2) vehicle represents the conservative approach to this long-duration mission. Due to the absence of large quantities of hydrogen in the fourth stage, polyethene is used for shielding, with a resulting weight penalty. No orbital mating is required.

15. D-$N_{123}C_4$(MM_2) - The first three stages of this vehicle are nuclear, Phoebus first stage and metallic-core upper stages. The fourth stage is chemical, due to the low velocity requirement of the earth-retro maneuver. No orbital mating is required.

Section 5 MISSION AND SYSTEMS EVALUATION AND SELECTION

5.1 EVALUATING CRITERIA DESCRIPTION

Deriving and applying some evaluating criteria becomes important because of the large number of mission modes and vehicle systems that can be hypothesized for manned Mars exploration. Seven criteria were selected as suitable evaluators, both within the objectives of this study and for interplanetary exploration in general. Four of the seven criteria require a qualitative evaluation because no suitable analytic parameter can be easily obtained. However, with continued analysis it may be possible to relate them to some analytic function or parameter. First of the criteria is the probability of mission success. This includes the technical confidence in the mission mode plus the general reliability and maintainability of the vehicle system. Second is the expected useful life (obsolescence) and the growth potential of the complete vehicle. Together, these two concepts (obsolescence and growth) minimize total long-range development cost. Third is the degree to which advanced concept development programs required for this vehicle will be applicable to other space efforts. Fourth is the need for special or additional support functions as a result of unique logistics or operational requirements of the system. Since the other three criteria could be derived analytically, the mission and system combinations were ranked on the basis of analytic data. The fifth criteria, mass in earth orbit, was calculated first for the interplanetary vehicle. Then, on the basis of an assumed earth-launch vehicle payload capability and reliability, it was translated into the number of earth-launch vehicles required. This was used as the figure of merit. The cost of the advanced concept development was added to the production cost of a single vehicle for the sixth criteria. Advanced concept development costs were derived as part of the program development studies, and the production costs were related to the vehicle mass.

The last criteria intended to be used was the percentage of available resources (both men and dollars) for the advanced concept development. This had been relatively unimportant as a criteria and in some respects seemed to overlap the preceding criteria, so it was eliminated from the final evaluation. It is now felt that a better measure would have been to estimate the complete vehicle development program requirements.

5.2 DESCRIPTION OF EVALUATION TECHNIQUE

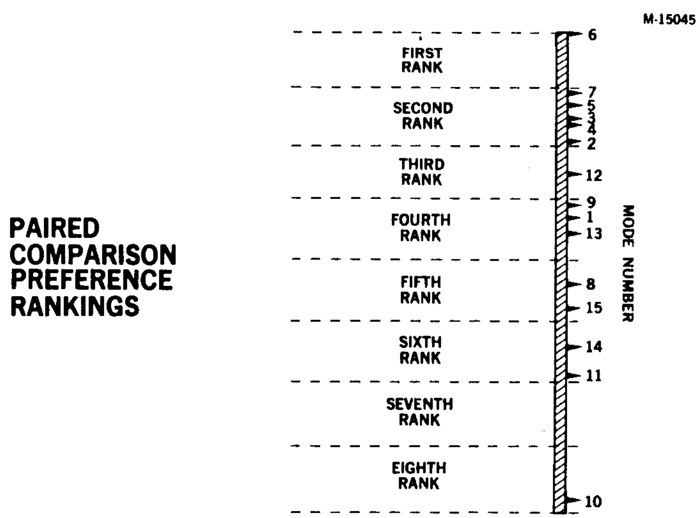
Pairwise comparison preference rating is a psychophysical method of rating items which has increasingly wide use for items such as these criteria. One of its basic advantages is the absence of any fixed rating scale; i.e., a judge does not have to fit a large number of items to an arbitrary scale. A preference comparison is made for each pair of items for all the evaluating criteria. From these comparisons, preference matrices are constructed. The result of the evaluation of these matrices is a final ranking of all the items considered, with the preferred one heading the list. Spacing between the items indicates the degree of preference.

The seven criteria were first evaluated in a paired comparison where the only criterion for rating was relative importance. The result of this comparison indicated that probability of success should have a weighting factor twice that of any of the other criteria.

The 15 mission systems described in Section 4 were then evaluated on the basis of the subjective criteria by 10 judges, each of whom had some systems knowledge in addition to being qualified in his specialized field. The quantitative criteria were numerically ranked and then combined with the results of the other criteria. After evaluating the rating forms, forming preference matrices and rankings for each criteria, the individual rankings (including the proper weighting factors) were combined to determine the final preference ranking for

the 15 mission systems. This ranking is shown in Figure 58. This ranking is not purported to be infallible or a mandate for any single mode over another. However, the top three modes do indicate a definite preference for a short mission with a vehicle using nuclear graphite propulsion in the first two stages, a nuclear metallic third stage, and either nuclear metallic or chemical propulsion in the fourth stage. From the three top-ranked modes, mode 5 was selected for more detailed description. Mission profile for mode 5 is launch out of earth orbit, retro into Mars orbit, use of MEM for landing and boost, launch from Martian orbit, and propulsion braking at earth to 12.2 km/sec (40,000 ft/sec), followed by aerodynamic braking.

Figure 58



Section 6 RECOMMENDED MISSION PROFILE AND VEHICLE DESIGN

6.1 MULTIMISSION PERFORMANCE

The weight-saving potential of several variations of the baseline mission profile were discussed briefly in sections 3 and 4 and are discussed in greater detail in Volumes III and IV. It is concluded that the original baseline profile is the most desirable mode of operation because it offers relative simplicity without large weight penalties. It is reasonable to conclude, purely from an economic standpoint, that vehicles must be developed which are capable of being launched at various dates on various missions, and during several synodic periods. A multimission launch vehicle capability was obtained for two mission durations, a 460-day mission with 20 days of stay time at the planet, and an 880-day duration with a 300-day stay. Individually refined estimates of stage propellant mass fraction, engine size, tankage, etc., were used in all weight comparisons.

For each synodic period, the baseline mission profile previously selected was used to size a vehicle configuration that accommodates a 40-day launch window. Figures 59 and 60 show the gross weight variation for the 40-day launch window in each synodic period for two typical mission durations. The variations in weight within each synodic period reflect the variation in velocity requirement as a function of launch date. The propellant tanks were sized for the maximum load within the 40-day window and were offloaded as required for any given launch date within this window. It may be seen that the 1977 synodic period is the most demanding when a short mission is considered.

Figure 59

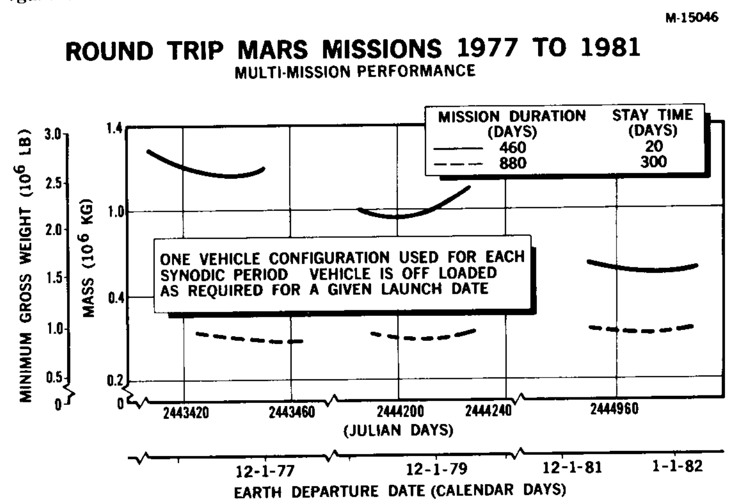
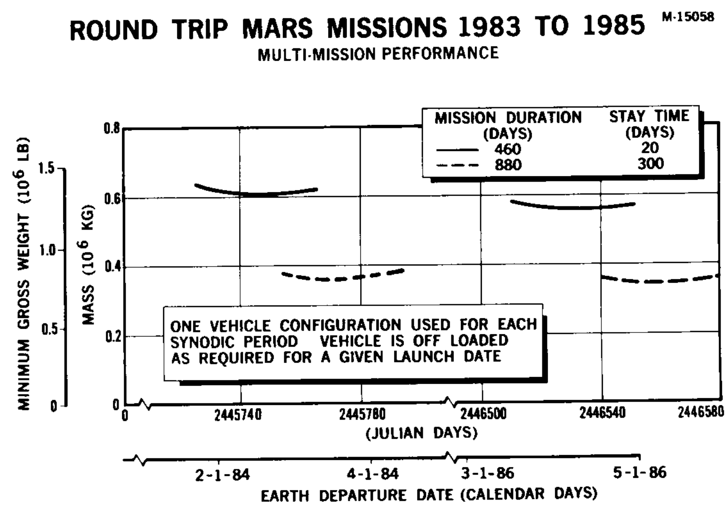


Figure 60



If a vehicle designed for a 460-day mission in 1977 were to be used for that mission in a later synodic period it would be greatly oversized but would probably offer much larger launch windows in the other years. The minimum gross weights for the 880-day vehicles show only slight variation with synodic period because the energy requirement is nearly constant.

The multimission launch vehicle size can be reduced if a small launch window is selected for the 1977 period rather than the full 40-day span. The vehicle size shown by a point in Figure 61 was obtained by using the minimum gross weight vehicle in 1977 with a 2-day launch window. The vehicle has a multimission launch capability, as shown by the available launch windows for other synodic periods, but has decreased in weight by 23% over that required for the 40-day launch window in 1977.

The same concept was used to determine multimission launch capability for the 880-day mission, with resulting minimum gross weights and available launch window shown in Figure 62 for several synodic periods. One vehicle configuration was used throughout all synodic periods and each vehicle was offloaded with propellant as required. Since energy requirements are nearly the same for all of these periods, only small variations in gross weight are required and multimission launch capability is more easily attained.

Additional weight savings can be obtained by eliminating 1977 as a launch period. The validity of such action is dependent upon development time, program cost, and the mission success probability associated with obtaining manned Mars landing missions by the 1977 period. These factors, combined with the state of development of various technologies, suggest that 1979 or 1981 is probably the earliest period of interest as far as manned Mars landings are concerned.

While the standard all-retro mission profile has been recommended, other profiles that have potential and should not be entirely dismissed are: the elliptic orbits at either Mars or earth, and the use of earth entry at 18.3 km/sec (60,000 ft/sec).

Figure 61

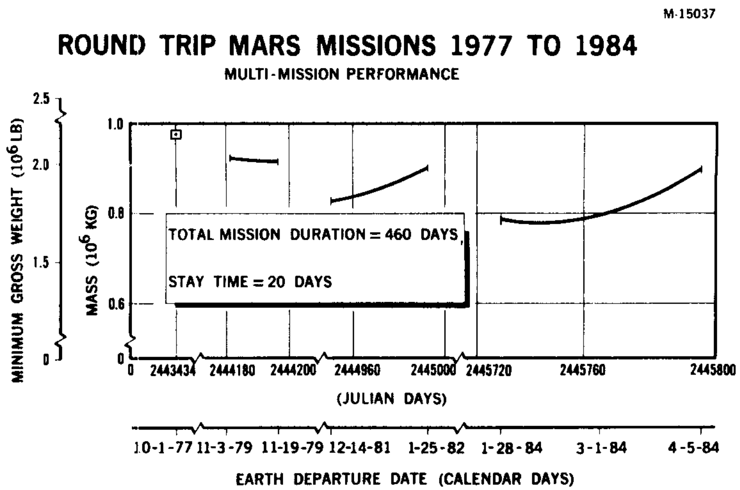
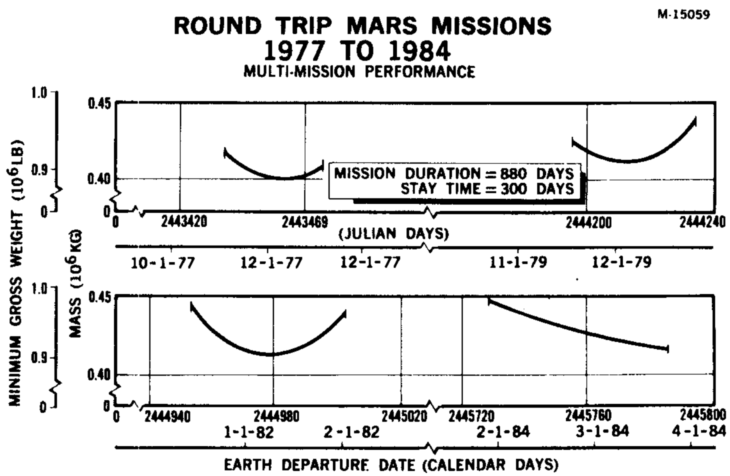


Figure 62



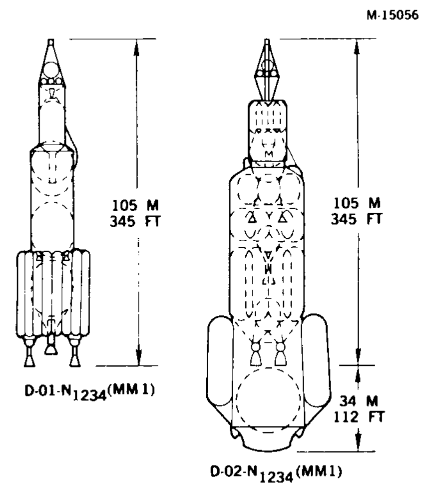
6.2 MULTIMISSION VEHICLE DESIGN

The multimission configurations shown in Figure 63 for the 460-day mission contain four stages, all of which are powered by nuclear engines: two clustered 250K graphite engines in the first stage, one 250K graphite in the second stage, two 30K fast-spectrum metallic core engines in the third stage, and one 30K in the fourth stage.

The chief aspects common to both vehicles are: (1) no orbital assembly, and (2) when launched from earth, each is unfueled except for a minimum amount of LH_2 fuel (5%) sufficient only for cooldown and pressurization. Once in orbit they are fueled by transferral of LH_2 from tankers. Multimission configurations for the 880-day mission have already been shown in Figure 57, and, except for size, possess the same design characteristics as the 460-day vehicles.

Figure 63

MULTI-MISSION VEHICLES



6.2.1 Configuration D-01-N_{1234} (MM1) Figure 63

This configuration has the first two stages arranged in a compact cluster, or basically a multishell structure, with the second stage (M2) propellant tank nested inside a multicellular toroidal first stage (M1) tank design. The top two stages have a manned cabin and biowell between the third (M3) and fourth (M4) stage tankage. The fourth-stage propellant is incorporated into a biowell design for solar flare protection. The EEM is centrally located in the biowell. The artificial-gravity centrifuge is forward of the hemispherical manned cabin command center. The MEM and nuclear auxiliary power units (two SNAP-8's) are at the upper end of the vehicle.

The dual-purpose earth-boost load structure incorporating the meteoroid shield and thermal insulation is jettisoned prior to leaving earth orbit. Empty tanks (stage 1) are retained for second-stage meteoroid protection and are jettisoned prior to Mars retro.

6.2.2 Configuration D-02-N_{1234} (MM1) Figure 63

Configuration 2 is similar to Configuration 1 in that both have four stages, EEM, biowell, MEM, and command center-living quarters.

Configuration 2 has clustered cylindrical tanks and is tandem staged. The M1, M2, and M3 stages are each composed of a central tank around which are located six cylindrical satellite tanks. The M4 stage has a central tank with a cluster of eight tanks surrounding it. All stages were designed for minimum weight at the time of engine firing. The combined earth-boost/acceleration-load structure, encompassing meteoroid protection, and super insulation (which is the outside shell of the vehicle) are jettisoned prior to engine ignition. All fuel tanks are divided into jettisonable units using identical domes, valves, fitting, and support methods wherever possible. Tanks were designed to be jettisoned in pairs as they are emptied. Sump tanks are used for the engine feed system.

6.2.3 Propellant Tank Design - Load Path - Meteoroid Shield

Propellant tank designs were based on studies of materials, pressures, and various configurations of large and small cylindrical and spherical tanks. As stated previously, titanium tanks were used in all configurations.

The majority of the tanks shown in Figure 63 are of conventional spherical or cylindrical shape. One particular tank design, used in Configuration 1, is a toroidal-type cylinder with a membrane web joining the intersection points of the cylinders. The tank design of the first two stages of Configuration 1 may be described as a multi-shell structure with the M2 propellant tank nested inside the segmented M1 tank.

An outer shell serves as meteoroid shield, load path, tank structure and fairing to protect the NRC-2 insulation. During earth-launch phase, the boost loads are transmitted through the outer shell to the upper structure. Figure 64 shows the outer shell earth boost structural support and trussgrid micrometeoroid shield. The outer shell structure is retained during earth orbit and jettisoned just prior to leaving earth orbit.

The basic structural difference between Configurations 1 and 2 occurs in the tank configurations of each stage and their load-transfer path. The first, second, and third stages of Configuration 2 are each composed of a central tank around which are located six cylindrical satellite tanks. The central tanks of each step are connected by interstages to form a cylindrical spine. At the top of this spine an inverted frustum of a cone provides a load path around the command center and intersects the outer shell structure. The outer shell structure is a jettisonable meteoroid shield which also provides the load path during the earth-launch phase. During launch from earth orbit, retro into Mars orbit, and launch from Mars orbit the cylindrical spine serves to transmit the engine thrust load to the structure above.

Figure 64

METEOROID & INSULATION STRUCTURE

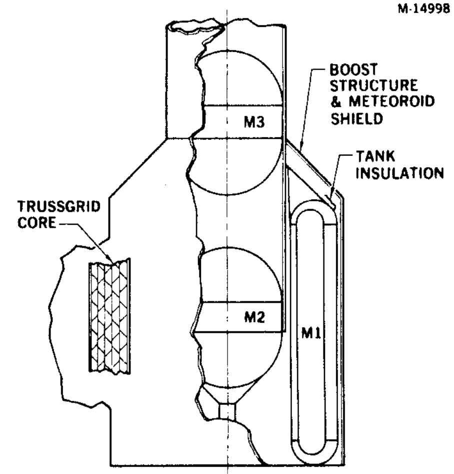
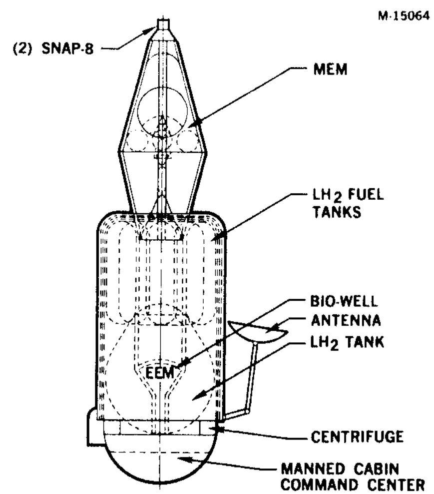


Figure 65

PAYLOAD ARRANGEMENT



6.2.4 Payload Arrangement

The payload of the multimission configuration as shown in Figure 65 incorporates the manned cabin command center, centrifuge, biowell-EEM, fourth-stage earth retro system, Mars entry module, and the nuclear auxiliary power units. The fourth-stage propellant, in conjunction with the EEM, was incorporated into a biowell design for solar flare protection. Approaching earth, the propellant forming the biowell is transferred to a central tank. In addition to the propellant shield, the M4 nuclear engine has a top gamma and neutron shadow shield, based on biological dose criteria and propellant heating considerations. Maximum protection is therefore afforded the crew from space and nuclear radiation.

6.2.5 Manned Cabin Description

Figures 66 and 67 present a cutaway view and equipment floor plan for the manned cabin compartment. The main floor deck of the manned cabin supports and positions all equipment for work and living volumes. The volumes are defined by equipment bays in 60° quadrants with access from the top and sides. The six quadrants define functional areas such as the command and control center, test and shop areas, biomedical centrifuge and dispensary, laundry and water closets, life support equipment, exercise and recreational areas, galley and food storage, and eating areas. The radial equipment bays contain deep open-truss structures formed by the floor beams with the top cap being the upper airlock structure. No equipment is attached to the curved dome. All loads are taken out by floor attachments at the tangent line of sphere to cylinder structure. Equipment is centrally located for short electrical line runs. Hence, the clear center core allows access from three sides and are no more than 25.4 cm (10 in.) deep for easy access and maintenance. Bays are enclosed for air-conditioning. The biowell is reached through the large airlock which also allows for system lines running to related equipment and controls in the biowell.

Figure 66

MANNED COMPARTMENT (CUTAWAY)

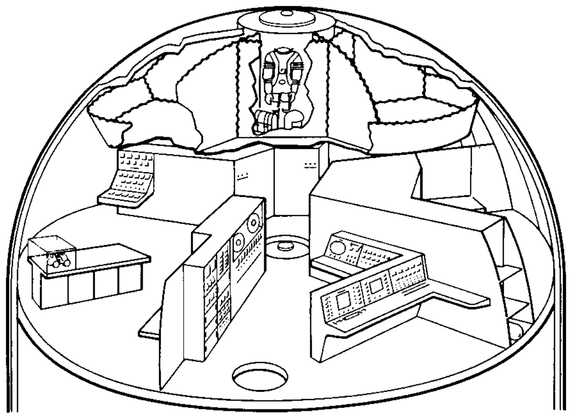
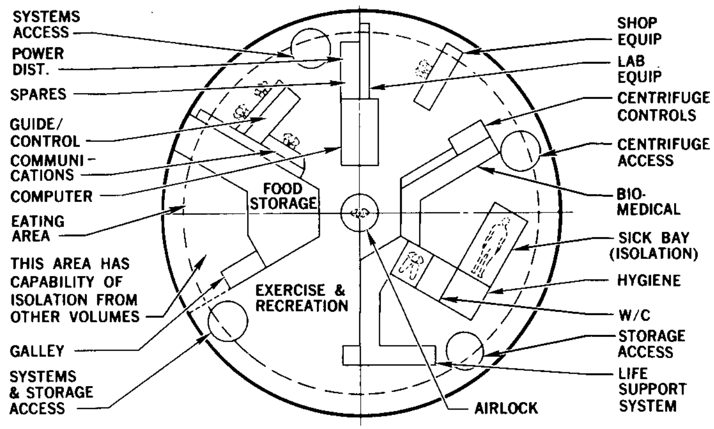


Figure 67

MANNED COMPARTMENT (FLOOR PLAN)



There is a clear-circulation envelope 101.5 cm (40 in.) between the work deck equipment and individual crew quarters at the top of the quarters dome. Individual crew quarters of 7-8.5 cu m (250-300 cu ft) each are defined by opaque, sound-absorbing fabric panels which attach to tension cables and form flat-sided volumes. The panels result in a reasonable amount of privacy as well as individual control of compartment sound, light, and temperature. Quarters are as remote from work and living area equipment as possible. These volumes can be expanded or the rigging and panels removed and stored, as needed. The centrifuge envelope is an enclosed ring at maximum diameter containing two cable-driven carts riding on the inner surface of the cylindrical rails.

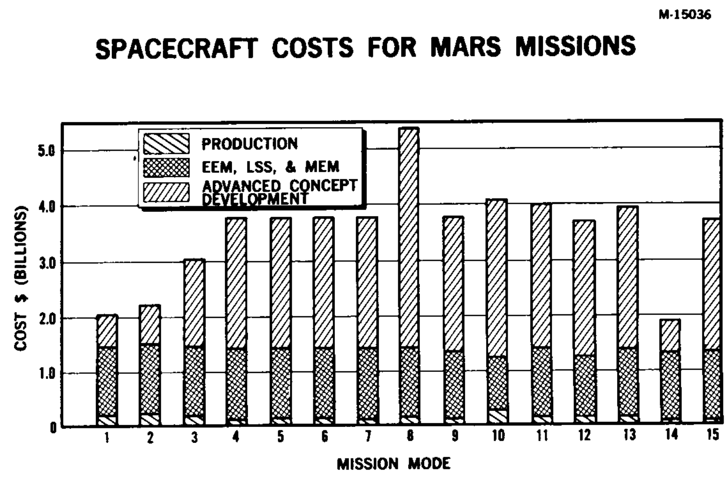
Section 7 PROGRAM DEVELOPMENT

The scope of program development in this study has included mission synthesis, production mass-cost analyses, cost estimates, schedules, manpower requirements, and facilities required to develop advanced system concepts. The intent is to establish the economic impact of particular missions and systems described in Section 4, and to indicate the feasibility of meeting the mission objectives within the 1975 to 1985 period.

7.1 BASIS OF ANALYSIS

Predictions of the production costs of the conceptually designed spacecrafts reported in this study were based on experience gained by Douglas in the production of other complex systems. Each system is analyzed on the basis of component weight, complexity of design and production, and learning cost factors. Analysis of the development programs was accomplished with a Douglas computer program simulation. The detailed description of the program is presented in Appendix I, Volume XII. This method of analysis allowed opinions and estimates concerning component technological problems to be categorized at the "expert" level of familiarity. The five technical areas defined to encompass all problems are: life support, propulsion, structures and materials, environmental problems and aerodynamic entry, and human training and preparation. It is important to recognize that the estimated development costs are for advanced concepts only and do not reflect the development costs of a complete vehicle. The complete vehicle development program could, in some instances, be concurrent with the advanced concept development, but would be several times as expensive.

Figure 68



7.2 ANALYSIS RESULTS

Results of mass-production cost analysis for each mode described in Section 4 are summarized in Figure 68. Estimates of life support system, EEM, and MEM costs were assumed to be constant. It can be seen that the magnitude of these costs ($1.25 billion) completely overshadows the variations in vehicle production costs ($200 x 10^6). The advanced concept development costs range from a low of about $600 million for an all-chemical vehicle to $4 billion for a vehicle using a gas-core engine on the first two stages and small nuclear metallic-core engines on the upper stages. It can be seen that the three top-ranked modes (5, 6, and 7) have nearly the same costs for both production and development programs. The relative cumulative advanced development program costs of the minimum, the maximum, and the adopted modes are shown in Figure 69.

Optimistic and pessimistic schedules were estimated for each problem in addition to the expected schedule. In Figure 68, the optimistic schedules for the adopted mode show approximately 10% less development time than the expected, but the pessimistic schedules show a possible increase of up to 60%. The associated cost variations are based upon the variations in salaries and facilities since equipment costs were assumed to be the same.

Expected program requirements for the adopted mode are shown in more detail in Figure 70. Total expected advanced system costs for this mode are $\$2.44 \times 10^9$ with a maximum yearly budget of approximately $\$0.485 \times 10^9$. Facility and manpower requirements reach maximums of 0.4×10^6 sq. meters, and 11,000 men, respectively, during the sixth and seventh years. Approximately 53% of the total expenses is paid in wages, 39% in supporting equipment and additional expenses, and 8% for facilities (considered rental).

Figure 70

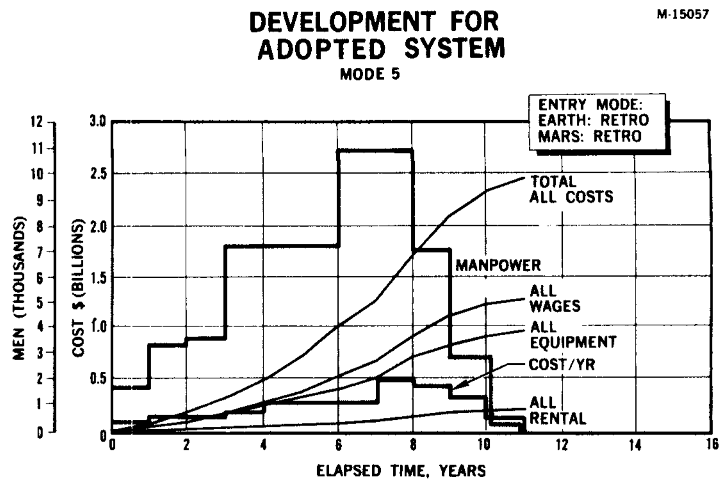
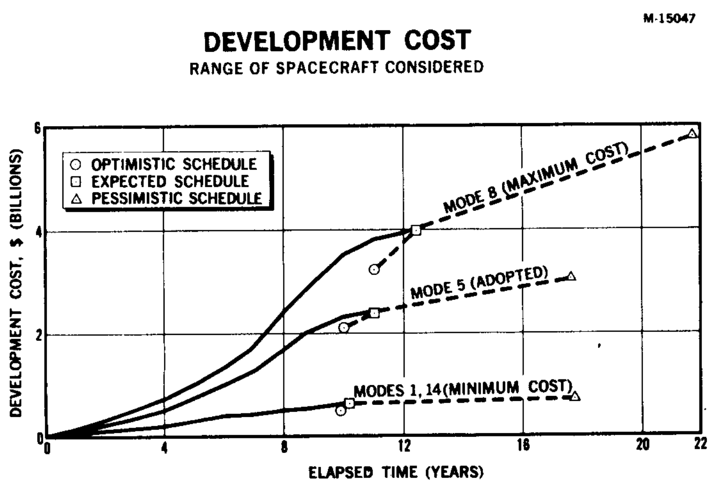


Figure 69



The items requiring development include a 250,000 lb. thrust nuclear graphite engine, a 30,000 lb. thrust nuclear metallic engine, and various other related propulsion items for a total cost of $2 billion. The human training and preparation is the next largest item, about $340 million.

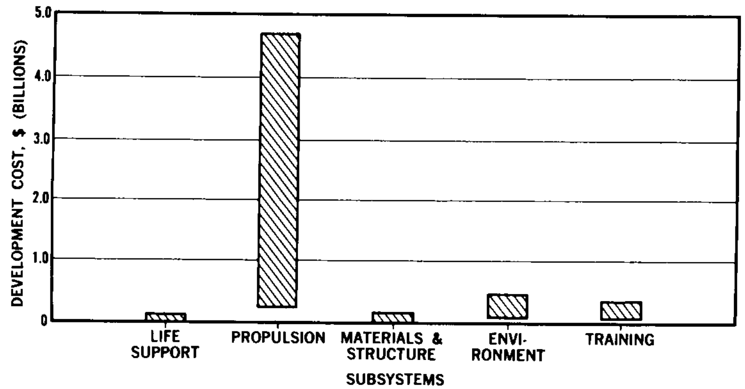
The estimated 11-year development time is established by the nuclear engines. Assuming that all advanced concept development must be completed three years prior to the actual launch date and allowing for flight testing, etc., the earliest launch date, based upon expected development, is 1979. The optimistic schedule could move this up to 1977; the pessimistic schedule could slip the date to 1986.

The sensitivity of the development cost to spacecraft subsystem selection is shown in Figure 71. This indicates that the largest single influence is choice of propulsion subsystem design. Although no mode was defined with the maximum propulsion cost, mode 8, which uses both gas- and metallic-core engines, was close at $4 billion, of which $3.5 billion is propulsion. The maximum costs

Figure 71

M-15040

RANGE OF COST FOR SUBSYSTEM DESIGNS



for environment (including aero entry) and training are nearly the same but lower than the propulsion systems by a factor of 10. The variation in environment costs depend upon how accurately the Mars atmosphere has to be defined for aerodynamic entry. The training costs variations result mainly from bookkeeping, i.e., assuming that confinement studies, radiation studies, reduced gravity studies, etc., are conducted for earlier programs and therefore the technology is independently available.

Section 8
CONCLUSIONS

- Study Objective 1 — Survey all attractive mission profiles for manned Mars missions during the 1975-to-1985 period, and select mission profiles of interest.

  Conclusions — (1) The most desirable short-duration missions (430 to 460 days) are highly dependent on synodic period; 1977 requires the largest and 1981-to-1985 the least gross weight for most missions. Desirable long-duration missions (800 to 900 days) are comparable in weight in all years. (2) Minimum gross weight is sensitive to stay time at Mars for both short and long missions. (3) Gross weight of short-duration missions is influenced by perihelion passage distance. (4) Adopted mission profile is launch from earth orbit, 325 km (175 n. mi); retro into circular Mars orbit, 555 km (300 n. mi); use of MEM for Martian operations; launch from same Mars orbit; retro to 12.2 km/sec (40,000 ft/sec) earth approach velocity followed by direct aerodynamic entry.

- Study Objective 2 — Define problem areas which appear capable of solution and which, if solved, can markedly improve mission capability during the 1975-85 period.

  Conclusions — (1) Restrict crew selection to 20-percentile men; (2) Reduce crew cabin volume to approach minimum anthropometric volumes; (3) a weightless or semi-weightless crew cabin design; (4) astronomical configuration reorientation; (5) use of elliptical orbits at Mars or earth; (6) aerodynamic attainment of Mars orbit; (7) duplicate staging; (8) combining meteoroid and insulation protection system with load-carrying structure; (9) propellant tank and meteoroid-insulation protection system jettisoning; (10) fiberglass tanks; (11) gas-core reactors; (12) propellant reliquefaction.

- Study Objective 3 — Develop a multimission vehicle where all subsystems are compatible with mission profiles and mission requirements in several synodic periods.

  Conclusions — The vehicle adopted possesses a launch capability in all synodic periods, where the available launch window varies with synodic periods. Available launch window was compromised in the most difficult years, 1977 and 1979 (i.e., stipulation of a 40-day launch window in 1977 and 1979 was waived), to effect a vehicle weight reduction of 23% in the other years. A short-duration mission (460 days with a 20-day Mars capture-orbit stay time) was selected over the longer mission. The multimission system weights are: 979 x 10^3kg (2,160 x 10^3lb) gross, and 278 x 10^3kg (613 x 10^3lb) dry. The crew size is six, the life support system weight (payload weight less the MEM and EEM) is 35,320 kg (77,910 lb). Propulsion is supplied by nuclear engines, large Phoebus-class and small metallic core.

- Study Objective 4 — Determine the nuclear thrust level which should be developed from interplanetary system requirements standpoint.

  Conclusions — Two nuclear engines are recommended: a 250K Phoebus-type for first- and second-stage application, and a 30K metallic core for upper-stage application; both to be used singly and in a cluster.

26988

PART II

SUBSYSTEMS REQUIREMENTS FOR A MARS MISSION MODULE

by

A. L. Jones
North American Aviation
Contract No. NAS9-1748

SUBSYSTEMS REQUIREMENTS FOR A MARS MISSION MODULE

by

A. L. Jones, North American Aviation
Contract No. NAS 9-1748

The objective of this study was to develop design criteria for the subsystems required for mission modules to be used in interplanetary missions. In order to be able to explore the constraints imposed upon subsystem design by the characteristics of the mission, and the interactions between subsystems and between subsystems and crew, a specific mission was selected.

The Mars mission concept established as a guideline for this study, Figure 1, is based on aerodynamic braking at Mars to establish an orbit, with separate modules for landing on Mars and Earth. A basic mission, within the time period of interest, was selected to develop subsystem requirements and for investigation of aerodynamic braking. This mission was launched from Earth on 5 August 1973, arrived at Mars on 3 December, and after a 40 day stay, returned to Earth 420 days after departure. Analysis of the characteristics of this mission established the crew and subsystem functions and requirements.

Other important guidelines were the requirement for artificial gravity of 0.4g to be obtained by deploying the Mission Module from the rest of the spacecraft through extension booms and imparting a spin on the overall spacecraft. The crew was to consist of four men, but the system should be designed to have a six man capability. Other modules that would be integrated into the spacecraft design were the Mars Excursion Module, which was to be utilized during the Mars stay period of the mission, and the Earth Reentry Module which would serve for Earth return at entry velocities of up to 65,000 feet per second. These two modules were being studied concurrently by other contractors.

The guiding philosophy for the investigation of subsystems was to develop "workable" concepts for all of the subsystems making up the Mission Module. Good examples of workable concepts are found in our everyday experience. For instance, automobiles that we drive everyday are highly workable systems, since they meet the same important design criteria that subsystems for interplanetary missions should attain. These are: First, long times between failures; second when they fail, they should not create a critical condition; third, the function performed can easily be made redundant; fourth, the mean time to repair is considerably less than the time allowable for the system to be nonperforming.

These criteria have been utilized as a first filter to guide the selection process and test for goodness of the subsystem concepts considered. For instance, in the power subsystems area, this criteria rejects the utilization of nuclear dynamic sources because they fail to pass the criteria of maintainability.

The requirement for maintainability becomes mandatory when it is realized that regardless of extremely high probabilities of nonfailure, the random characteristics of failure allow the failure to occur at any time during the mission. Therefore, it is impossible to expect subsystems to be failure-free and only through the process of designing subsystems that do not create critical conditions when they fail and that are maintainable, can the probability of success for an interplanetary mission be assured.

Interplanetary spacecraft design involves three major areas of investigation; mission profiles, transportation systems, and crew effectiveness. The subsystems study was directed to the question of subsystem design to achieve crew effectiveness with emphasis on synthesis of workable subsystems for crew survival, experimentation and spacecraft command and control. This study was also directed to the investigation of mission profiles in the vicinity of Mars and spacecraft configurations capable of utilizing aerodynamic braking upon arrival at Mars to establish an orbit.

Utilization of the Martian atmosphere for braking is a very promising mode to reduce the weight and size of the spacecraft to values commensurate with expected launch vehicle capability.

Since the Mars aerobraking phase of the mission produces specific requirements for subsystem and mission module design, I would like to first present some highlights of the results of that portion of the study before discussing the subsystem.

The entry corridors are shown in Figure 2 for a range of entry velocities which represent the upper range of those that may be encountered on missions launched in any year. The entry velocity associated with the basic August 5, 1973 mission is 27,600 feet per second. These corridor depths are defined as the difference in vacuum perigee altitude of the overshoot and undershoot limits. The entry interface was assumed to occur at one million feet altitude. The change of the corridor depth in the mean atmosphere is shown for L/D of 0.5, 1.0, and 2.0 at 30,000 feet per second. At this velocity the corridor increases by 55 percent between L/D of 0.5 and 1.0 and only 21 percent between L/D of 1.0 and 2.0. Increasing the undershoot load factor from 5 to 10 g's increases the

corridor depth by almost a factor of two. At this velocity, the upper and lower atmospheric models create about a 22 nautical mile difference which is only slightly less than the 28 nmi corridor for the mean atmosphere. The atmospheric uncertainty will be reduced by unmanned probe data. Operational margins can be obtained using spacecraft with L/D's greater than 0.5. The variation in corridor depth was found to be a very weak function of W/C_DA. Corridor depths for a spacecraft with an L/D of 0.5 is shown for a very low density atmospheric model (11 mb surface pressure). It can be seen that the available corridor becomes very small at the higher velocities.

During the deceleration phase there is maximum operating velocity at which the maneuver can be performed with a specific spacecraft. Figure 3 shows the limiting velocity for spacecraft designed for maximum load factors of 5 and 10 g's as a function of L/D. At this velocity, a spacecraft of a fixed L/D is operating at both the overshoot and undershoot limits. It can be seen that this velocity is a strong function of L/D. It is interesting to note that this limiting velocity is only a function of the limiting load factor and the L/D, provided only that an atmosphere exists with a density range of the models defined. At velocities in excess of these values the spacecraft is either skipping out and/or pulling a load factor higher than the design limit. The absolute altitude above the surface at which this limiting velocity is reached is determined by the atmospheric density and also the spacecraft W/C_DA. These altitudes are above 100,000 feet for a load factor of 5g, even in the low density model. At this load factor it appears that a significant increase in the operating velocity limit can be achieved using an L/D of about 1.0.

In these aerobraking analyses, deceleration was maintained at constant altitude until reaching the proper velocity to initiate the skipout maneuvers. Skipout trajectories were computed using a gravity turn control mode (by continuous roll) for W/C_DAs ranging from 500 to 2000 pounds per square foot. Since a gravity control mode was used, the aerodynamic accelerations experienced during pullup were independent of L/D.

The exit angles at the atmospheric interface of one million feet that result from a gravity turn pullup are shown at the left in Figure 4. Values are shown for trajectories whose apoapses altitudes are 250, 500, and 1000 nautical miles. The curves on the right show the required impulsive injection velocity as a function of exit angle. The shaded areas indicate the total variation in exit angle for the range of W/C_DA expected. The ΔV for a 500 nmi orbit varies from 600 to 665 feet per second. The required orbit circularization velocity increases rapidly with altitude.

An important aspect of the mission is the selection of the Mars parking orbit plane. Ideally, the parking orbit plane should be consistent with an immediate Earth return capability throughout the mission stay time. In addition, the orbit plane should satisfy the operational requirements of exploration. For specific missions these criteria may prove incompatible and require a degree of compromise.

The parking orbit must contain the incoming V vector. The vector can be oriented, on approach to Mars, in an infinite number of inclinations and result in clockwise or counterclockwise orbits. The minimum inclination, however, may be limited on any specific mission by the declination of the incoming V_∞ vector.

During the planned 40-day stay the precession (or regression) of the line of nodes is as much as 233 degrees. Thus the trans-Earth injection velocity will be a function of stay time as well as orbital inclination. Figure 5 shows the minimum departure velocities for stay times of 0, 10, 20, 30 and 40 days (arrival December 3, 1973). On each launch date, trans-Earth trajectories were analyzed, with transfer times from 225 to 285 days, to determine the injection ΔV from a series of 500 nautical mile circular parking orbits at various inclinations. A periapsis departure was assumed for all injections to satisfy the direction of the outgoing V_∞ vector which was fixed by the launch date and transfer time. A series of trajectories were calculated for each inclination and stay time and the transfer trajectory yielding the lowest ΔV was selected.

It may be noted that the declination of the incoming V_∞ vector for the reference mission (Earth launch August 5, 1973) was -22.41 degrees which limits the minimum inclination to that value. In Figure 5 it can be seen that the absolute minimum ΔV's do not occur at the same inclination for each stay time, near minimums occur at inclinations of approximately 60, 95 and 195 degrees for north injections and 95 degrees for south injection. The north and south injections refer to the hemispheric position of the B vector of the Mars approach conic. The spacecraft for performing this mission consists of a number of modules, each with special functions to perform, integrated into a shape having suitable aerodynamic characteristics. In addition, the spacecraft must be designed to permit staging of the Mars excursion module, Earth entry module, propulsion modules and to provide an 0.4g artificial gravity during interplanetary flight.

In order to select the most promising basic shape, five important factors were investigated: (1) entry corridors, (2) heat protection, (3) structural design, (4) packaging or staging complexity, and (5) Saturn V launch vehicle compatibility. For the purposes of initial analytical evaluation preliminary sizing drawings were made of spacecraft having a range of lift to drag ratios from 0.5 to 2.0. The basic shapes shown in Figure 6 were sized using preliminary estimates of the weight, volume and packaging efficiency of the various modules. A high density propellant, OF/MMH and fixed ΔV's were used to establish these sizes. These shapes have a wide range of volume to area ratios which was desirable for comparing the heating, packaging and structural problems. These factors were analyzed and the spacecraft properly resized to investigate their compatibility with the launch vehicle.

Because of the thermal radiation, the body shape plays an important role in the total heating to the vehicle. The radiant heating rate to the Apollo shape vehicle, for example, is much larger than the convective heating, whereas the radiant heating to the M-2 shape vehicle is negligible compared to the convective heating on the major surfaces.

In determining these radiative heating rates, both the possibility of temperature decay and absorption were considered. Since the free stream kinetic energy rate is high compared to the radiant intensity, little temperature decay occurs in the shock layer. However, the radiant heating rates are in the neighborhood of blackbody and were consequently limited by allowing gas self-absorption. Heat protection materials of current state-of-the-art so far as fabrication and environmental applicability are concerned were used. A comparison of heat shield weight for three of the basic shapes is shown in Figure 7.

For a purely convective thermal environment, it would be expected that heat protection rates would increase with increasing lift-to-drag ratio. This figure shows that the Apollo shape with an L/D = 0.5 has the greatest heat protection weight excepting at high backface temperatures. The large heat shield weight of the Apollo shape is due to the severe radiant heating environment on nearly the entire front face of the vehicle. The M-2 (L/D = 1.0) and Delta-Wing (L/D = 2.0) have only a small region which is affected by radiant heating. The Delta-Wing, however, has a larger surface requiring protection. For current state-of-the-art bond line temperatures of about one thousand degrees Rankin, this chart shows the heat protection for an entry velocity of 27,600 feet per second will be from about 2.8 to 5.0 percent of the vehicle weight. The design of the spacecraft is important in determining the effect of the heat shield weight on the gross weight of the vehicle. A major portion

of the heat shield for the Apollo shape can be designed so it can be jettisoned after aerobraking. The portions which are retained on this or other shapes increases the Mars departure propellant requirements, hence increases the gross weight rapidly.

Packaging and staging arrangements were developed to permit preliminary structural and weight analysis. The results of these preliminary analysis weights in thousands of pounds are shown in Figure 8 for two structural temperatures. Here the effects of staging can be seen, where the Apollo shape with the heaviest heat shield results in the lowest gross weight and the Delta-Wing with a lighter heat shield is considerably higher. The M-2 shape, basically half of a 13 $^1/_2$ degree half-angle cone, resulted in gross weights similar to the Apollo shape.

In addition to the shapes discussed, a conic shape with a nose radius of 0.2 base radius was considered. This configuration had characteristics similar to the M-2 (L/D = 1.0) and offered a design compatible with Saturn S-II stage load distribution requirements. The Mars excursion module design, developed by the Aeronutronics study, and the Earth entry module defined by Lockheed's study were incorporated into each of the aerobraking configurations.

Evaluation of the basic problems of entry corridors, heat protection, structural design, module packaging and launch vehicle compatibility, lead to the selection of a conical configuration for further evaluation. The subsystem requirements, as well as those accruing from aerodynamic braking were incorporated into an integrated design, shown in Figure 9. The mission module in the spacecraft extends during trans-Mars and trans-Earth flight to permit rotation which provides an artificial gravity. The spacecraft weight is about 550,000 pounds when fueled for the referenced mission. The spacecraft can be placed in Earth orbit by the Saturn V and after arrival of a crew, propellants can be loaded and the spacecraft mated with an Earth escape propulsion stage.

The initial analyses of the aerodynamic braking phase and investigation of spacecraft configurations and weights were based on the characteristics of a referenced 420-day mission launched from Earth on August 5, 1973. Spacecraft characteristics were determined for a range of values applicable to missions launched in other years. The effect on the spacecraft heat shield weight of higher entry velocities is shown in Figure 10.

A comparison of the convective and radiative heating indicates that the radiant heating is dominant in the stagnation region for this velocity range. The aerodynamic heating conditions imposed during entry along the overshoot and undershoot trajectories were compared at each vehicle point and the heat shield

was designed to the worst condition. The heat protection scheme used Avcoat X-5026-39 ablator contained in honeycomb to aid in retention of the char layer. A minimum gage titanium skin is bonded to the ablator exterior to afford protection from the space environment. This skin has little significance to heat protection since it would be consumed early in the entry over most portions of the vehicle. The heat shield weights may appear to be large, however, they represent only a small percentage of the gross weight of the spacecraft. The heat shield removes a large amount of energy. A propulsion system to do the same job for the same weight would require a specific impulse of about 8000 seconds.

Since only portions of the heat protection material are jettisoned in Mars orbit, extra propellant is required for trans-Earth injection as the heat shield weight increases. The effect of entry velocity on the gross weight of the spacecraft is shown in Figure 11.

The stay time at Mars for which the spacecraft system is designed will effect the gross weight. The weight of the Mars excursion module, being studied by Aeronutronics, varies as a function of crew size and stay time. The Mars departure velocity requirements vary as a function of orbital inclination and stay time (Fig. 5). The effect of stay time on the spacecraft gross weight is shown in Figure 12. An orbital inclination was chosen which resulted in the minimum change in ΔV over the 40 days. Aeronutronics data on a Mars excursion module with a three man crew was used.

The results of this portion of the study have indicated that utilization of aerodynamic braking at Mars for reducing propulsion requirements appears completely feasible based on the current knowledge relative to the limits of the Martian atmosphere.

Based on the study philosophy described earlier, the study was approached in a manner that optimized the investigation of the workability of the subsystems. This was accomplished (Fig. 13) by, first, defining the functional requirements, that is, what had to be accomplished in order to satisfy a functional requirement of the mission. This approach differs from the standard approach of trying to develop a specific item of hardware in that the objective is to meet the functional requirement during both normal and emergency modes. Therefore, the next step was to develop logic diagrams which indicated how the functional requirements would be met both during normal as well as emergency modes. In several cases there were various ways to approach the concept design. These concepts were then analyzed as to their main characteristics, such as weight, volume,

power and maintainability. The system concept was then selected and a failure mode analysis was performed at the main subfunction level to determine the required time to repair the failures. This served to further pinpoint areas where redundancy had to be designed into the subsystem.

At mid-study, a mission module configuration was developed to determine the interior arrangement of the mission module and the location of all the important subsystem components. This allowed the investigation of the integration effects as well as the crew interfaces. During the second part of the study, the selected concepts for each subsystem were developed to a finer level of detail and the failure mode analysis performed to determine the spares requirements for each subsystem.

While a great number of study guidelines were stipulated at the beginning of the study, it became obvious that other guidelines or design constraints would develop from the mission characteristics, the integration of the mission module within the spacecraft and the requirements posed by the presence of a crew with the mission module, and these were investigated at the study inception. The mission characteristics effecting the subsystem design were mainly the mission profile, the mission sequence, the environment, and abort considerations. In the systems integration area, important considerations developed from the physical, electronic, and thermal integration. In the electronic integration area it became apparent as the study progressed that a central electronic computer offered great advantages in data processing because of redundancy and spares. In the thermal integration area, it was found possible to utilize the rejected heat from the isotropic dynamic nuclear source to maintain the temperature of the Mars Excursion Module within acceptable limits. The presence of the crew, with their main functions of providing command and control, maintenance, and scientific observation, imposed requirements for grouping of controls in accordance with the probable skill distribution of the crew members, and provision for areas within the mission module to supply the various life support needs of the crew.

The Mars mission represents a considerably more ambitious undertaking than the lunar landing mission; since, instead of a 14-day mission time, the mission length becomes 420 days which represents an increase greater than an order of magnitude. If the standard approach of determining mean time before failure, based on pure statistical determinations is used, it is quickly realized that the Mars mission is considerably beyond the realm of possibility. As can be seen on Figure 14 in the Life Support subsystem for Apollo, the MTBF objective is 130,000 hours. The best predictions of MTBF for this subsystem for 1965 are only 43,000 hours. In the Apollo project, the objective will be met

through the utilization of redundant design in the system. To attain the much greater MTBF required for the Mars mission of 4,400,000 hours, in addition to redundant design, a high order of maintainability and a suitable supply of spares and repair tools is necessary. The requirement of MTBF for the auxiliary power is ten million hours for the Mars mission, which is not considered possible to attain for rotating machinery, thus, here again, the subsystem design must be based on considerable redundance and maintainability in order to assure a successful mission.

Figure 15 depicts a functional diagram for the Life Support subsystem in normal and emergency operating modes. Each one of these functions, such as atmospheric control, food management, and others, requires the performance of a number of subfunctions for successful operation. For instance, the atmospheric control function requires power for control and to produce a flow of fluids and CO_2 removal and reduction. The distribution function applies to electric power, gases and liquids; storage applies to electric power, oxygen, nitrogen, repressurization supply and backpack of suit supply; removal collection applies to CO_2, water, oxygen and hydrogen; the reduction subfunction applies to CO_2; the generation subfunction applies to O_2. As can be seen from the diagram, those subfunctions which are shown heavier than the others must operate at all times, that is, both during normal and emergency modes. The reduction and generation subfunctions which are shown in a lighter block are thus indicated as not being absolutely necessary during the emergency condition and therefore it is not required that they be redundant but rather that they are repairable within a reasonably short period of time.

Shown in dotted lines are some of the interfaces that occur between one of the system elements and the major functions. Water is utilized as a heat sink for temperature control and is utilized also for personal hygiene, for food management, and to supply the electrolysis unit in the atmospheric control function. Analysis of this logic diagram, shown here is a simplified form, provided data on the effects of failure. These data were used to establish the allowable down times or maximum times for repair. When the estimated repair time was longer than allowable the system was redesigned with the necessary redundancy. Such analysis provided the backup and emergency mode concepts.

Schematic diagrams were used to describe the design of each of the subsystems and to perform a detailed failure analysis to establish spares. From this analysis, realistic weight, power, volume, and spare requirements can be defined for workable systems.

A schematic diagram of the selected mechanization for the atmosphere control function of the life support subsystem is shown in Figure 16. Cabin air is circulated through a debris trap to remove particulate matter, is passed through a charcoal filter for removal of trace contaminants, is passed through a water separator after suitable cooling and then is recycled back to the cabin. A parallel loop picks up part of the air and circulates it through the CO_2 removal loop and feeds it back to the system between the debris trap and the charcoal filter. The removed CO_2 is then mixed with hydrogen generated by electrolysis and passed to a reduction unit in which the usable output is water and the non-usable output is methane with traces of CO_2. Methane and CO_2 are further reduced by utilizing a catalytic reaction with useful output of water and residue of carbon.

During the study, analyses were made of alternate methods of mechanizing each part of the system. Time does not permit presenting each of these, however, one has been selected as an example of the detailed analysis accomplished in the study. An example of some of the trade-off studies made is illustrated in Figure 17 which shows the relative weight, required radiator area and required electric power for two concepts of temperature control. The active cycle utilizing refrigeration with Freon II appears to be considerably less desirable than a semipassive cycle utilizing ethylene glycol and water. Because of this, the latter was chosen to be used in the selected concept.

The main physical characteristic of the selected telecommunications subsystems are depicted in Figure 18. The television equipment is similar to that utilized in Apollo having 320 lines per frame, 224 resolution elements per line and a 20 frame per second rate. This requires a 700 kc information bandwidth.

Signal conditioning and telemetry equipment is sized to accept a wide variety of engineering data and can handle an information bandwidth of 60 kc. A teletype unit has been incorporated in this subsystem since it is felt that the time delay involved in voice transmission will make conversation much less desirable than the printed information available with the teletype. All communications between the mission module and the Earth will be transmitted through the S-Band transponder which has an output power of 1 kilowatt. This power, in conjunction with a 10-foot diameter antenna in the spacecraft and a 210-foot diameter antenna of DSIF assures acceptable S/N ratios at the ground station. Communications between the mission module and the Mars Excursion Module will be carried out through the L-Band transponder. The weights shown

on this chart are for the basic system hardware; spare requirements will be summarized in a later chart.

The navigation requirements were investigated and several concepts were analyzed for mechanizing them. A guideline affecting guidance and navigation, particularly the approach to Earth, was the desirability to be completely self-contained, thus not requiring information from Earth in order to perform any guidance maneuver or in determining navigational position. Several analyses with different degrees of sophistication were made to determine if this requirement could be met. One of the critical requirements is that of navigating the spacecraft to meet the entry corridors upon return to Earth. The data on Figure 19 indicates the results of an analysis performed utilizing a computer program developed by the Autonetics Division of NAA. As an input to this calculation, it was given that the observational error made with a sextant would be 10 seconds for the 1 sigma cases, and that new measurement data would be obtained every two minutes starting at about 10 Earth radii.

As can be seen from these data, radial position error determination using sextant only for the 60,000 feet per second velocity is much too great to be acceptable and the 1 sigma error is 12 miles when the spacecraft is 1 Earth radii away. If very high confidence were required for the entry maneuver, or a 6 sigma case, the uncertainty of the corridor is in the order of 72 miles. Incorporating some stadiametric data improves the situation somewhat and for the same 60,000 feet per second case the same order of accuracy is achievable at distances of about 6 Earth radii. A considerable improvement can be achieved by utilizing DSIF data, which allows much more adequate error determination at distances in the order of 6 Earth radii.

The conclusion of this analysis is clear. If the guidance and navigation system is to have a complete onboard capability, it must contain its own ranging equipment.

During the first half of the study, three guidance and navigation concepts were developed. The first one utilized an inertial measuring unit, an optical device, a clock and a computer for the normal mode. For the emergency mode, a hand-held sextant, a chronometer, desk calculator, and tables were utilized to perform navigation with lesser quality. The second concept was quite similar except that it included as part of the computer element a failure determination and indication unit to allow location of malfunction and repair. The backup mode was also as described earlier. The third concept shown in Figure 20 utilized fully redundant elements and, additionally, a switching logic which automatically

detects the malfunction, locates it, and switches the standby unit instead. This concept has the advantage of offering high navigational quality, both during the normal and emergency modes and high reliability. Throughout the mission there are not many times when the guidance and navigation system is critical. It has a low duty cycle. However, at times such as midcourse correction, aerobraking, injection, and Earth approach, a failure of any element requires a backup of equal quality. Since the weight, volume, and power requirements for the guidance and navigation subsystem are only a small fraction of those for the overall system, the third concept was selected for continued development during the remainder of the study. Many of the major components of these navigation systems, such as the IMU and optics are not readily repairable. With poor maintainability, redundancy is required.

The three main flight control functions are shown on Figure 21. First, to maintain the vehicle attitude, second, to point the antenna, and third, to either point or determine the orientation of various scientific instruments. This function is accomplished through the subfunctions of sensor, computation and actuation. To take advantage of the inherent redundancy afforded by the presence of the crew, displays and controls to override the automatic system are available.

The sensing function for the vehicle orientation is accomplished by line of sight to the sun, stars, or planets. It may also be determined from an inertial table and, in the case of vehicle spin, from the measurement of attitude rates. Sensing subfunction for the antenna and scientific instruments may be determined principally by line of sight devices, or commanded orientations and orientation rates. In order to make the flight control system compatible with other electronic subsystems leading to a utilization of a central computer, all the sensing outputs are digital.

The actuation function as in the case of the vehicle, consists of translation, rotation, and stabilization. This is accomplished through the utilization of 100-pound thrust jet engines burning bi-propellants with 300 seconds of specific impulse. In the mission analyzed 2100 pounds of propellants were required to precess the spacecraft to maintain the solar power collector orientation to the sun. The spin up of the spacecraft to produce artificial gravity required 467 pounds of propellant. In the case of the antenna and scientific instruments, pointing of these devices is effected through the utilization of electric drives.

The flight control subsystem is used nearly 100 percent of the mission. However, there are only a few critical periods in which backup elements were needed. The pilot override is considered to be the best backup for many of the flight control functions.

The principal area of system integration occurs in connection with the electronic subsystems. A central computer was developed to meet the computation and memory requirements of all the subsystems in the spacecraft. Figure 22 shows only a few of the most important subsystems from a data processing point of view. However, similar functions of sensing, control, displays and storage are performed for the life support, scientific experimentation and the power subsystem. The utilization of a central computer results in the great advantage that both the displays, such as digital readouts, condition and status lines, and the tie-in to telemetry, can be readily rearranged to meet the requirements of the moment, thus providing great flexibility of operation and monitoring.

The computer was sized by developing a computer program for computer design. For the computer program each subsystem was coded and each signal from that subsystem to the computer was given its own code number. Each signal code also carried the function to the computer, its units of measure, type of signal, number of input channels to the computer, maximum and minimum values, where it is to be routed and, finally, its required accuracy. The computer program was then divided into the various phases of the Martian mission and a printout of each subsystem signal during each phase determined the Mission Module computing loads during that particular mission phase. Thus, the input loading, output loading, computer loading and storage capacity necessary were easily obtainable.

The necessary computer loadings indicated that a relatively advanced spaceborne computer would be necessary, which had a quick access time (not to exceed 4 micro-seconds). Its major elements are a channel router, digital-to-analog and analog-to-digital switches and converters, input-output shift registers, input scanner, priority trap section, primary storage, display and arithmetic and logic units. The computer is fully redundant, i. e., twice the number of elements required are available. The central computer loading requirements determined by the computer program indicate a maximum sampling rate of 15, 000 samples per second, and a minimum storage capacity of 24, 000 words. For design purposes a maximum total storage capacity of 32, 000 words was postulated. Each word consists of 32 bits of information and 4 bits for parity, giving a total word length of 36 bits. Therefore, the central computer would have a maximum sampling rate of 240, 000 bits per second and a storage capacity of 1. 152 million bits. This computer is within the present state-of-the-art and it has the weight, volume and power requirement characteristics shown in Figure 23. It would utilize microminiaturized components in modular form.

A computer program developed by Autonetics was utilized to perform a failure analysis upon the central data processor. An estimate of the components comprising each block of the computer was made and each block was typed, weighed, and rated according to 1963 MIL specs. Interconnecting lines are assumed to be in bus line form and are treated as one unit having no failures. The computation determines root square of the square of the sum for MTBF, taking into consideration the series and parallel connections of the blocks. The MTBF determined by this program indicate that the computer must be extremely easy to maintain in order to meet the mission requirements.

The power requirements for the mission module are shown in Figure 24. The three columns indicate the maximum power requirements for the subsystems at any time during the mission, (these loads do not occur simultaneously), the power requirements during the trans-Earth injection phase, which are the highest exhibited during any phase of the mission; and, finally, the emergency power which is that required to maintain life and to perform needed repairs of malfunctions. During the normal mode of operation, a maximum power requirement of about 11 kw is expected. The emergency power requirements are 1500 watts for life support and other miscellaneous requirements and 1000 watts for maintenance operations. The power system as proposed has a 19 kw capability which allows failure of any one generating unit without requiring load reduction. The intermittent loads are: for data transmission, those required to operate the transmitter for communications with Earth, and for the power subsystem, those required to deploy the solar collector.

To achieve reasonable values for crew safety and mission success, a power subsystem with a mean-time-before failure of 10^7 hours was required. Since equipment of this reliability is not forseen in this time period, repairability was an important consideration in evaluation and selection of systems to meet this requirement. One of the most important tradeoffs relating to the power subsystem is that of system weight versus electrical output. Figure 25 indicates the weight variation of three principal types of power sources that were considered for the mission module.

The two leading contenders for energy source were the nuclear systems which have the great advantage of not being dependent on distance to the Sun and the solar energy systems which, while dependent on spacecraft to Sun distance, are prominently advantageous from a point of view of maintainability. Of the nuclear systems, the reactor type, such as Snap 8, are considerably less desirable than the isotropic source since once it has been activated the residual radiation is much too great to allow maintenance to be carried out.

At the same time, the nuclear systems exhibit much greater weights for given electrical outputs and these two main considerations of maintainability and weight favored the choice of a solar dynamic system as the primary source of electric power. Since there is always the danger of loosing attitude orientation with respect to the Sun, to which solar dynamic systems are particularly sensitive, a secondary electric source was included consisting of an isotopic dynamic unit and several battery packs. The secondary system becomes the primary system during the aerobraking maneuver for which the solar collectors must be jettisoned.

The power subsystem, shown schematically in Figure 26, was defined after conducting analyses of failure modes and effects, and after conducting a thorough investigation of the spacecraft and mission integration and interface problems. The primary power supply consists of three solar dynamic units with a 5 kw rating. Two of these normally deployed. They utilize a rigidized Mylar collector which is jettisoned prior to attempting the aerobraking maneuver at Mars.

After injection into Mars orbit, spare collectors are deployed. The third unit is normally stowed within the spacecraft and will be utilized in case of major failure of either of the operating units. The secondary power source consists of an isotopic dynamic system utilizing Pm-247 and a set of four primary batteries, each one rated at 500 Whr. Heat rejection in the solar dynamic units is accomplished through a radiator that is oriented normal to the spacecraft-sun line. Heat rejection for the isotopic dynamic unit is also through a radiator; however, since the isotopic dynamic unit supplies the electric power during the aerobraking maneuver, the heat is rejected to a water boiler until the maneuver is accomplished and the space radiator can again be utilized. Redundancy of requirements for distribution of the electric power are met through the provision of two ac busses and two dc busses. Three ac to dc converters, each of which can handle the total conversion loads, are provided as indicated in this figure. The Mission Module configuration shown in this figure consists of one major cylinder with two floors and a smaller cylinder which serves as a storm shelter as well as a redundant pressurized container for crew survival. All the command and control functions are exercised from within the storm shelter. Stores of gases and liquids, as well as the major part of the dense equipment in the module, have been arranged in such a manner as to provide the greatest radiation shielding to the crew within the storm shelter. An evaluation of this average density indicates that 4.6 grams per square centimeter is available. The storm shelter has a volume

of 700 cubic feet and is designed for continuous occupancy for up to 48 hours for six men, at which time a resupply of food would be required. The normal seating provisions are for only four men since two crewmen would be sleeping, preparing food, or performing other duties. When the vehicle is experiencing high acceleration forces, these two crewmen would occupy folding entry seats. Areas have been provided within the Mission Module for the performance of scientific work, for sleeping, for performing minor surgical operations, and ample room and equipment for exercising. Recreation, study, maintenance, and personal hygiene areas also have been provided. Two transfer interlocks allow access to the Earth reentry module and to the Mars Excursion Module and their fully pressurized conditions. Both of these interlocks can also be utilized for access to the outside.

A detailed structural analysis was made of the mission module. Tradeoffs were prepared on all of the major design factors to permit development of the most efficient structural system. The analyses were based on requirements developed after a thorough investigation of the spacecraft loads and the mission environment. Preliminary comparisons of exterior shell tradeoffs indicated a considerable reduction in weight could be realized by selecting a structural concept in which the crew would maintain the shell pressure tight by repair. The weight of the structure designed to meet pressure and dynamic loads had to be increased when the meteoroid shielding requirements were added.

One of the detailed tradeoffs made is shown in Figure 28. Here, the unit weight of the exterior wall is shown for two constructions as a function of separation of outer meteoroid bumper from the inner shell. As can be seen from this figure, for the specific probability of no penetration of 0. 90 and for a 420-day mission, the optimum spacing seems to be between 1 and 1. 5 inches. The titanium curves indicate greater weight, which is due to the minimum available titanium gage. The truss core sandwich construction appears a little lighter than the honeycomb construction, however, the honeycomb construction is more readily within manufacturing state-of-the-art and meets the thermal requirements.

The design of the skin of the storm cellar was based on a probability of no penetration of 0. 999 and twice the normal pressure loads. For this analysis, the exterior cylinder wall was considered to be a small distance away from the shelter.

The selected material for the storm shelter was aluminum honeycomb sandwich for the outer wall, of 3/4-inch depth and total skin thickness of .082 inch. The exterior wall construction includes a 5-inch deep honeycomb sandwich plus a bumper sheet separated by 1.5 inch of filler and insulator.

The activities of the crew were studied for two principal mission phases during trans-Mars and trans-Earth, and during Mars orbit. The complete crew of four men is assumed to be available during the trans-Mars and trans-Earth phase and only two are considered to remain in the spacecraft during Mars orbital operations. (The other two crew members having departed in the Mars Excursion Module.) Figure 29 indicates the number of hours spent by each crewman in performing the various activities listed. Since the results of this study have shown the need for subsystems designs which are repairable, the maintenance and repair function becomes a very important activity and, as can be seen from this chart, a total of six manhours are available each day for maintenance during the trans-Mars and trans-Earth phases and only two manhours per day are available during the Mars orbit operations. In the case of a heavy maintenance load, the noncritical activities, such as scientific experimentation, training, and recreation would have to be replaced by the maintenance function as required.

This tabulation of crew activities has been utilized to perform analyses on the oxygen requirements based on experimental data. The results of this analysis indicating an average consumption rate of 2.4 pounds of oxygen in one day during the trans-Earth and trans-Mars phases; 2.7 pounds per man day during the Mars orbital flight.

A crew of four men could perform the mission if the Mars Excursion Module could operate safely and effectively with two. A minimum of three can perform the necessary spacecraft functions during interplanetary flight, but must have experience in a great number of skills. This crew size restricts the amount of time available for system maintenance and scientific experimentation, particularly in the Mars orbital phase. Here, where scientific interest is highest, the time available is a minimum. A restricted amount of time available for system maintenance places development requirements on the subsystem. As the maintenance time decreases, the systems have to be developed which have fewer failures.

In this study, instead of utilizing the concept of reliability to determine the required MTBF, the more useful concept of availability was utilized which may be defined as the fraction of total mission time when the system is completely operable, i.e., not being maintained. This assumes that the maintenance

action can be accomplished which will restore these systems to a "like new" operation. It may also be defined as the probability of functioning at any specific time or point in the mission profile. As in reliability, availability is a probability, as well as an expression of usable time. The important characteristic is that it expresses the probability of being completely operable at any given point in time. The availability of the system may be expressed mathematically as the ratio between MTBF and MTTR plus MTBF (MTTR = mean time to repair). Figure 30 indicates some of the relationships that exist between availability, mean time to repair, number of repair actions and availability requirements as a function of the number of crew. Since the mission lasts about 10,000 hours, a 2,000 hour MTBF would suggest that five repair actions would be needed, while a 1000 MTBF system would require 10 repair actions. Systems exhibiting a one hour mean-time-to-repair (MTTR) and having a 2000 hour MTBF would demand an availability of .9994. This appears noncompatible for a four-man crew, but quite acceptable for a six-man crew.

The final failure analysis performed on the mechanization of the selected concepts resulted in a subsystem concept which yielded realistic weights for equipment and for spares as shown in Figure 31. The overall subsystem weight includes in some cases expendables such as atmosphere constituents to make up for leakage loss and flight control propellants. The table indicates the weights of all the subsystems which results in an overall mission module weight of 37,000 pounds. This weight is quite compatible with the spacecraft design proposed under Part B of this study. The heaviest subsystem is life support which requires a total of 8,000 pounds which is a little over 20 percent of the overall mission module weight. The electronic subsystems are extremely light and their selection then is somewhat insensitive to weight. A contingency of 2,700 pounds was included in the total module weight.

The study has indicated that there exists a fine interaction between the crew size and skills, the subsystems, and the ability to perform maintenance. The crew is another subsystem that is prone to failure and that has to be maintained. Confidence in the practicality of the systems and crew sizes postulated can be answered by a program utilizing a prototype module as described in this report and a crew with appropriate skills. A program was defined beginning with ground prototype and ending with Earth orbital tests to yield valid answers to the problems of achieving crew safety and mission success. The resulting system would be applicable to any long duration and space venture.

The development program for the mission module, Figure 32, reflects an orderly progression of testing, culminating with an Earth orbital test for final qualification. The testing approach provides comprehensive ground tests with emphasis on integrated system-crew testing.

In summarizing it can be said that aerodynamic braking at Mars to establish an orbit is a feasible operation considering the expected knowledge of the atmosphere in the time period of the mission. A workable spacecraft concept was established for performing this maneuver which included the integration of specific special purpose modules for excursion to Mars' surface and atmospheric entry upon return to Earth.

Workable concepts for the mission module subsystems were developed which perform the necessary functions associated with the long duration flight to Mars. The system requirements were met using the concept of availability which infers that the hardware be maintainable. Integrated testing of subsystems and crew can establish confidence in achieving mission success and maximizing crew safety prior to undertaking the ultimate mission.

FIGURE 1

MANNED MARS MODULE SUBSYSTEMS STUDY

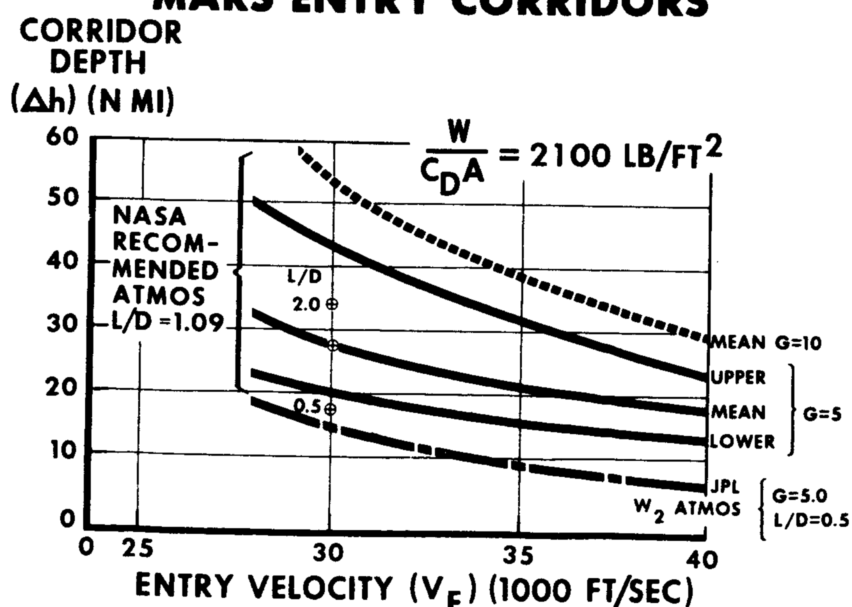
- **MISSION TIME PERIOD - 1970 - 1975**
- **MSC MISSION CONCEPT**
  - AERODYNAMIC BRAKING AT MARS
  - ARTIFICIAL GRAVITY 0.4g AT 75 FT R
  - CREW 4 TO 6 MEN
  - SEPARATE MODULES FOR LANDING ON MARS & EARTH
- **WORKABLE CONCEPTS FOR**
  - LIFE SUPPORT
  - GUIDANCE & NAVIGATION
  - FLIGHT CONTROL
  - POWER
  - DATA TRANSMISSION
  - SCIENTIFIC EXPERIMENTS
  - MISSION MODULE STRUCTURE

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

12 63 PS 46707

FIGURE 2

MARS ENTRY CORRIDORS



NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70293

FIGURE 3

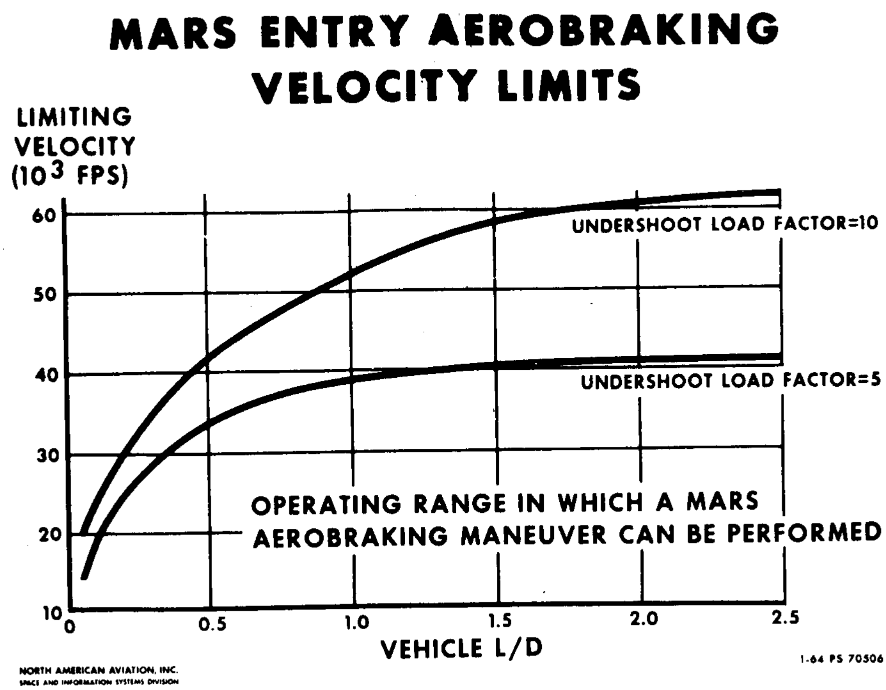


FIGURE 4

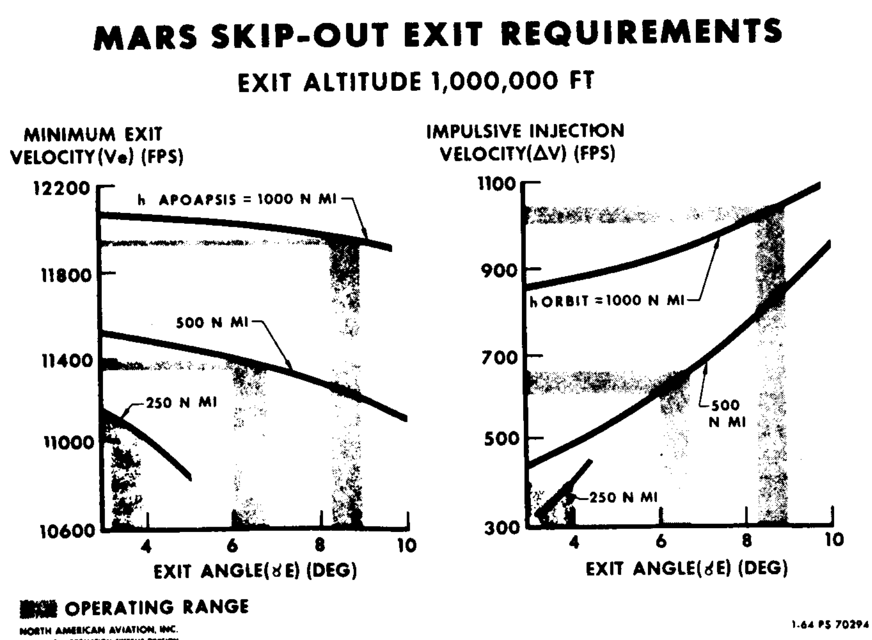


FIGURE 5

MARS PARKING ORBIT CONSTRAINTS
MINIMUM INJECTION ΔV
500 N MI CIRCULAR ORBIT PERIAPSIS INJECTION

COPLANAR ΔV'S

STAY TIME	ΔV MIN
0 DAYS	13,800 FPS
10	14,100
20	14,600
30	14,900
40	15,100

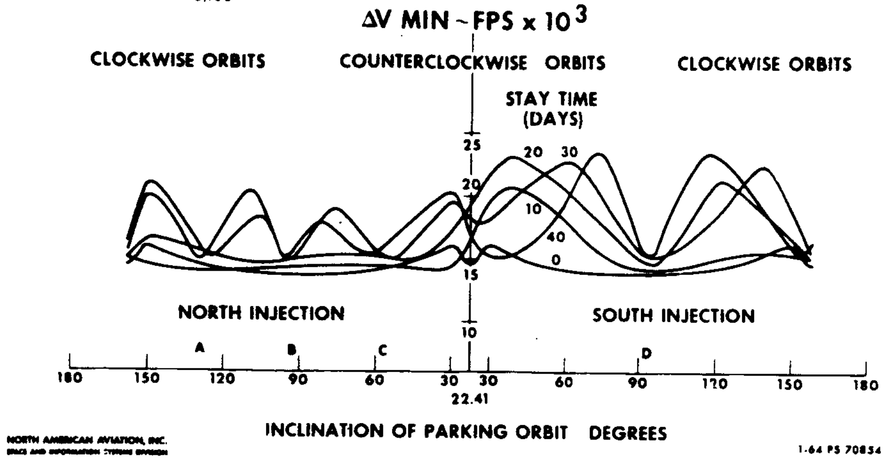


FIGURE 6

BASIC SHAPES

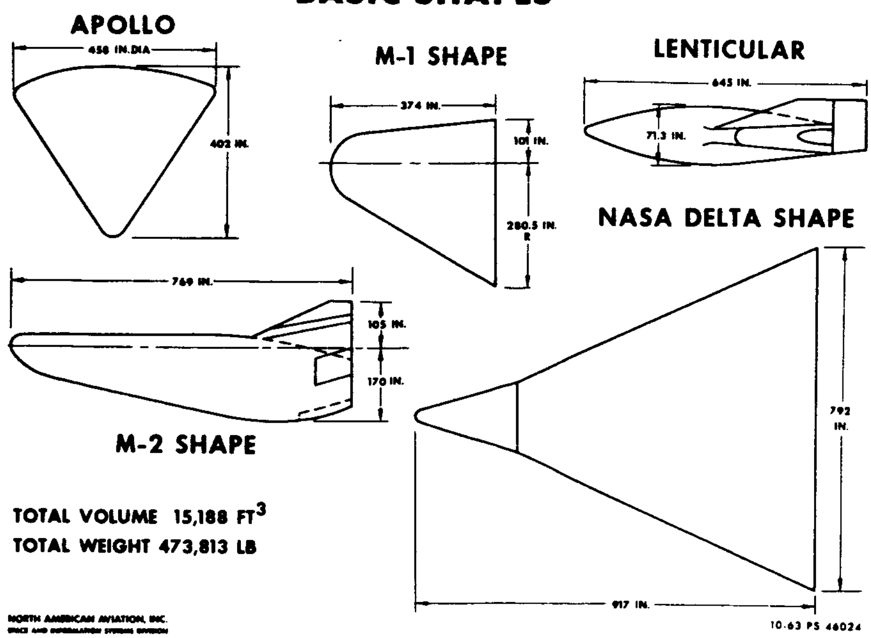


FIGURE 7

EFFECT OF CONFIGURATION ON HEATSHIELD WEIGHT

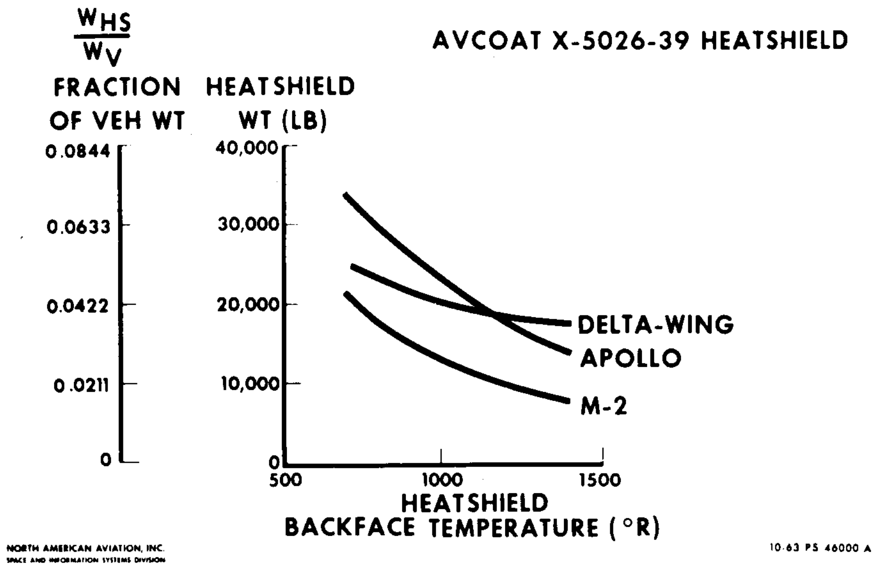


FIGURE 8

WEIGHT SUMMARY

	300°F			1000°F		
	APOLLO	M-2	DELTA	APOLLO	M-2	DELTA
• BASIC PAYLOAD	108	108	108	108	108	108
• PROPELLANT	274	307	508	264	277	409
• TANKAGE & SUPPORTS	4	4	9	4	3	7
• STRUCTURE & INSULATION	11	11	72	12	14	60
• HEAT SHIELD	30	22	41	20	8	17
• MISCELLANEOUS	5	6	7	5	5	7
GROSS WEIGHT	432	458	745	413	415	608

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

10-63 PS 46010

FIGURE 9

AEROBRAKING SPACECRAFT

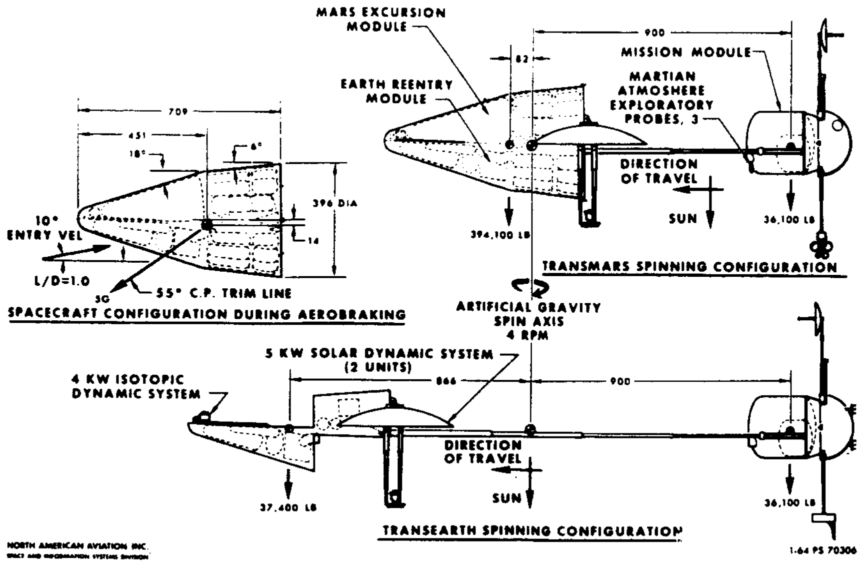


FIGURE 10

HEAT SHIELD WEIGHT VS VELOCITY

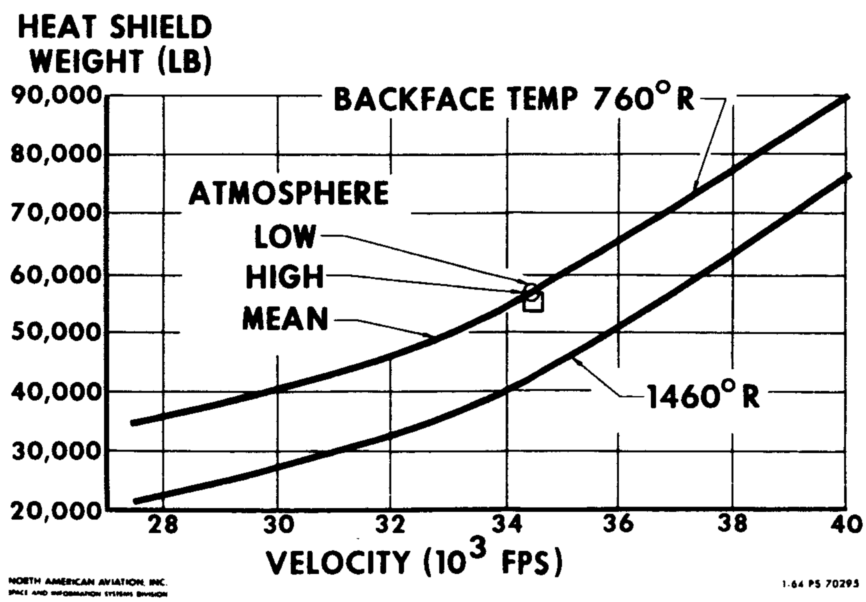


FIGURE 11

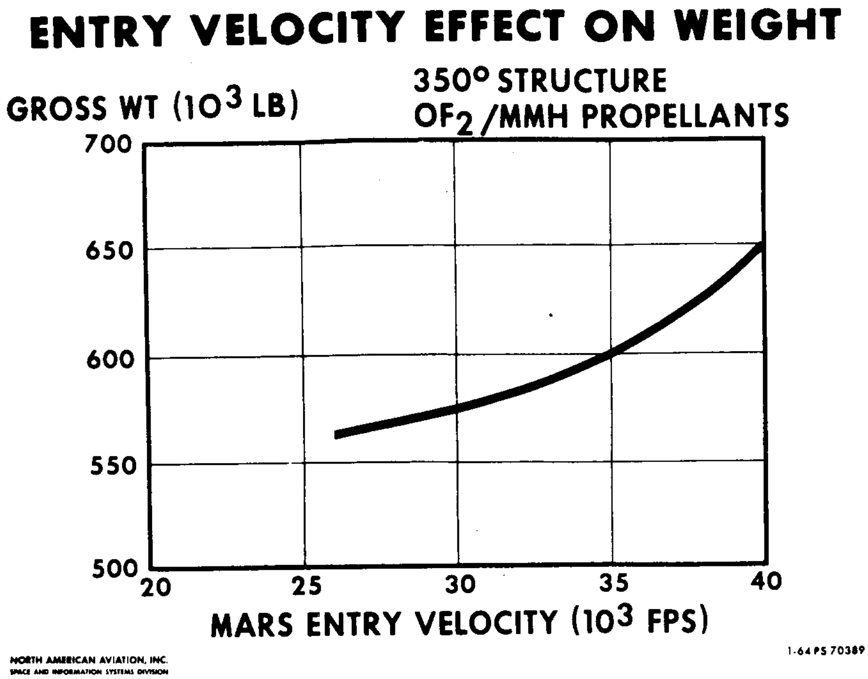


FIGURE 12

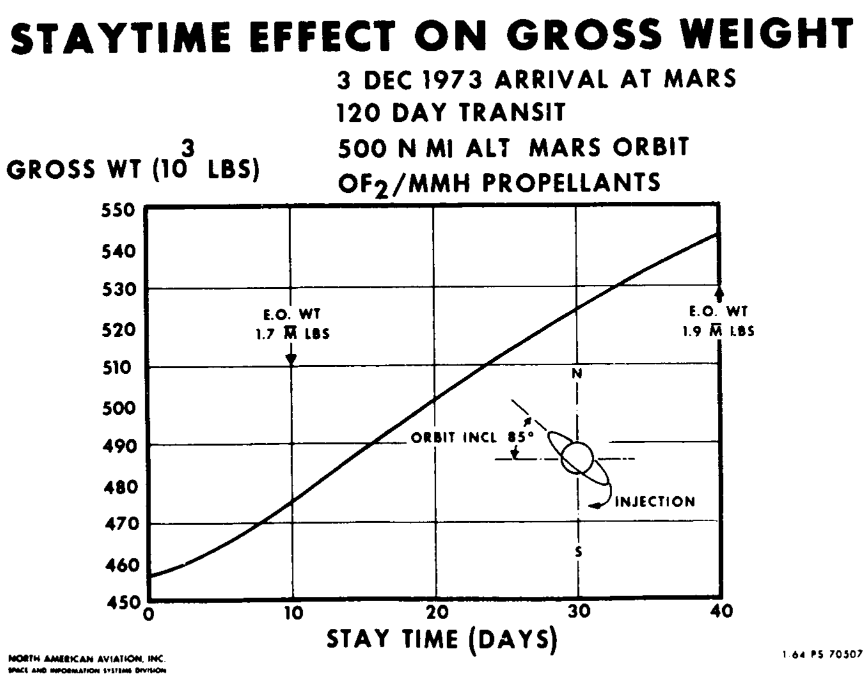


FIGURE 13

STUDY APPROACH

ANALYZE SUBSYSTEM ENVIRONMENT

- MISSION PROFILE
- MSC SPACECRAFT CONCEPT
- SPACE ENVIRONMENT

SELECT OPTIMUM SUBSYSTEM CONCEPTS

- PERFORM FAILURE MODE ANALYSES
- INVESTIGATE INTEGRATION EFFECTS
- DETERMINE CREW FUNCTIONS
- DEVELOP COMPARISONS OF RELIABILITY & MAINTAINABILITY

GENERATE ALTERNATE CONCEPTS

- ESTABLISH FUNCTIONAL REQUIREMENTS
- DIAGRAM NORMAL & ALTERNATE MODES
- DIAGRAM ALTERNATE SUBSYSTEM CONCEPTS
- ESTIMATE WEIGHTS, VOLUME, POWER

ANALYZE MECHANIZATION OF SELECTED CONCEPTS

- ESTABLISH ALTERNATE MECHANIZATION TECHNIQUES
- ANALYZE
- INTEGRATE
- COMPARE & SELECT

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

9-63 PS 60186

FIGURE 14

SYSTEM MAINTENANCE

MEAN TIME BEFORE FAILURE (10^3 HR)

SUBSYSTEM	MARS MISSION OBJECTIVE	APOLLO (1965) PREDICTIONS (NO MAINT)	APOLLO (1965) OBJECTIVE WITH MAINT
LIFE SUPPORT	4,400	43.0	130
GUIDANCE & NAVIG	190	7.0	180
CONTROL SYSTEMS	850	1.6	20
AUXILIARY POWER	10,000	2.2	250
COMMUNICATIONS	7,300	1.7	200
INSTRUMENTATION	330	2.0	10

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

12-63 PS 46708

FIGURE 15

LIFE SUPPORT FUNCTIONS

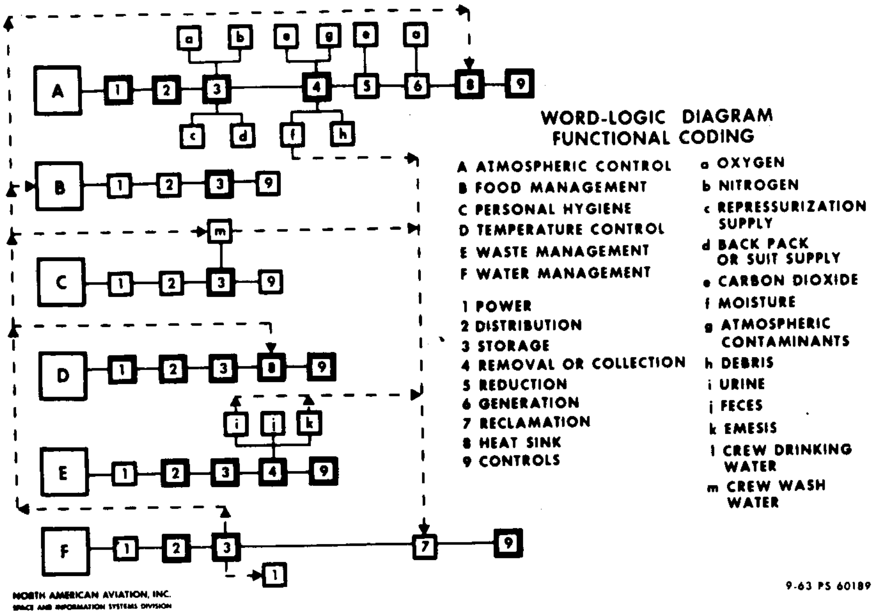


FIGURE 16

ATMOSPHERE CONTROL

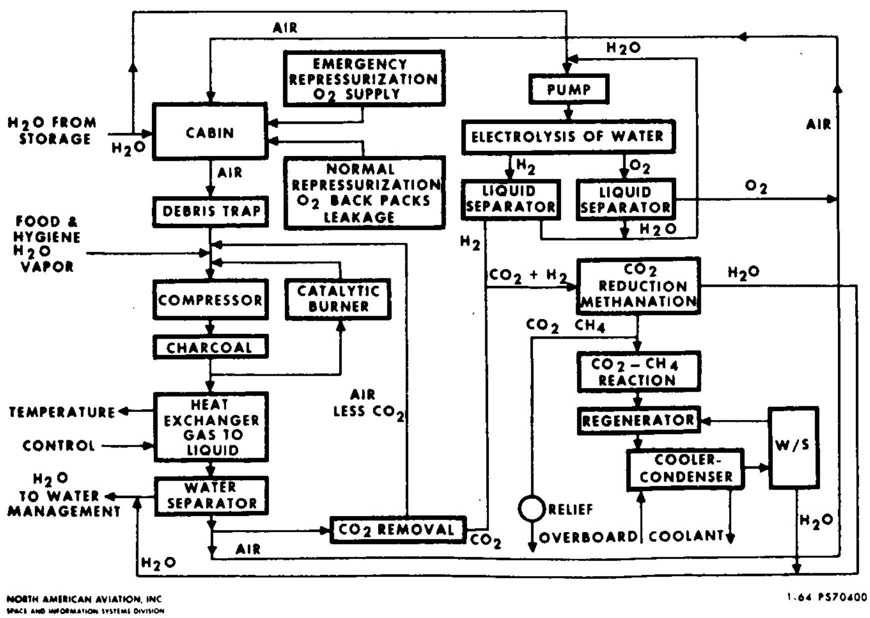
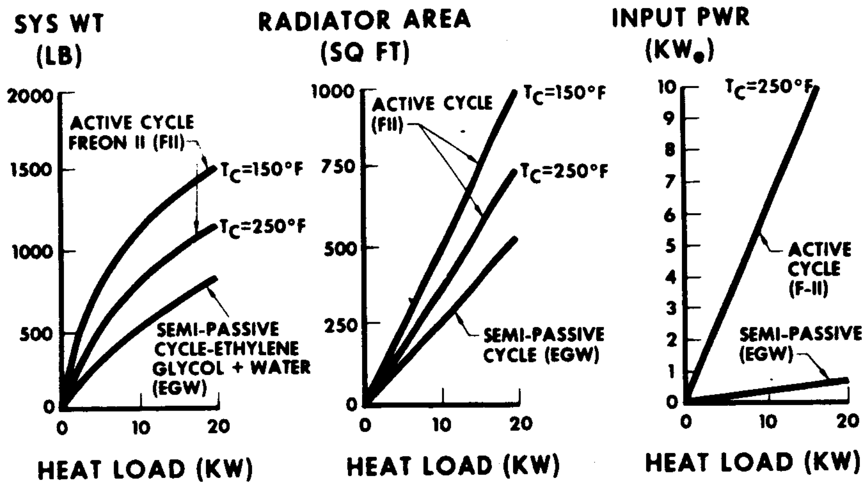


FIGURE 17

TEMPERATURE CONTROL COMPARISON



NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

9-63 PS 60202

FIGURE 18

TELECOMMUNICATIONS SUBSYSTEM

	WT (LB)	VOL (CU IN)	INPUT POWER
OPTICAL EQUIPMENT	6.7	172	---
TELEVISION EQUIPMENT	17.0	657.5	33
SIGNAL CONDITIONING	2.9	63.5	7.5
TELEMETRY EQUIPMENT	10.25	197.5	57.5
DATA STORAGE	39.0	2550.0	60.0
INTERCOMM EQUIPMENT	13.2	---	8.0
TELETYPE	35.0	1495.5	100.0
SIGNAL AMPLIFIER	18.8	25.6	7.5
S-BAND TRANSPONDER	139.0	2860.0	1915.0
L-BAND TRANSPONDER	46.0	1085.0	957.5
ACCESSORY EQUIPMENT	32.5	200.0	---
TOTAL	360.3	7811.1	3146

NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS70392

FIGURE 19

RADIAL POSITION ERROR

DUE TO RANDOM MEASUREMENT ERRORS

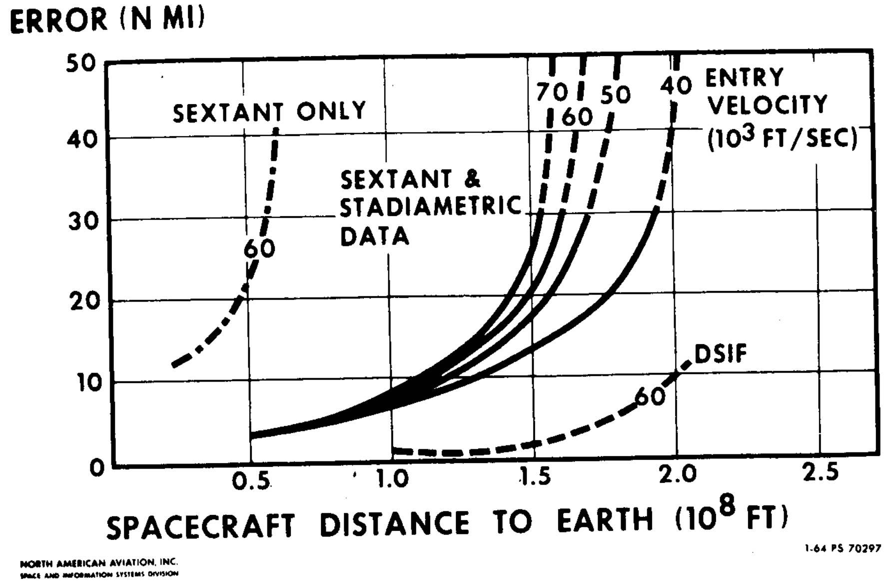


FIGURE 20

G&N SUBSYSTEM

CONCEPT NO. 3

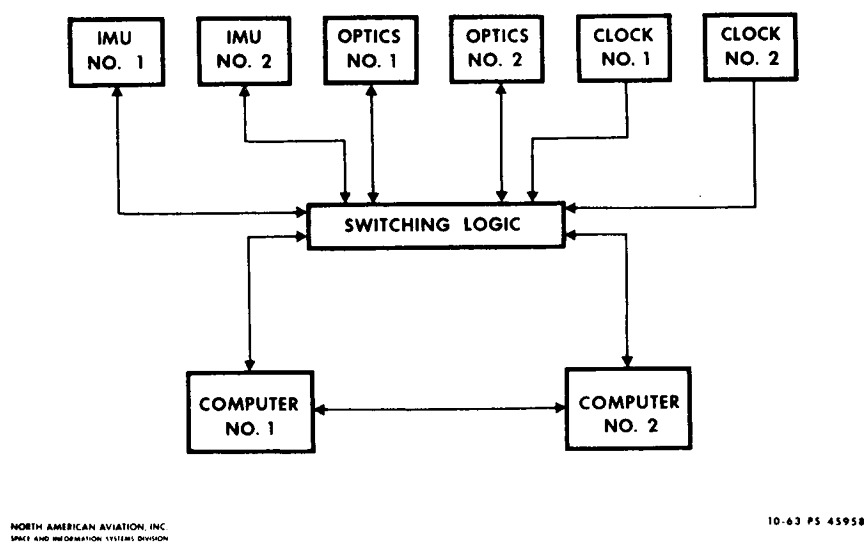


FIGURE 21

FLIGHT CONTROL FUNCTIONS

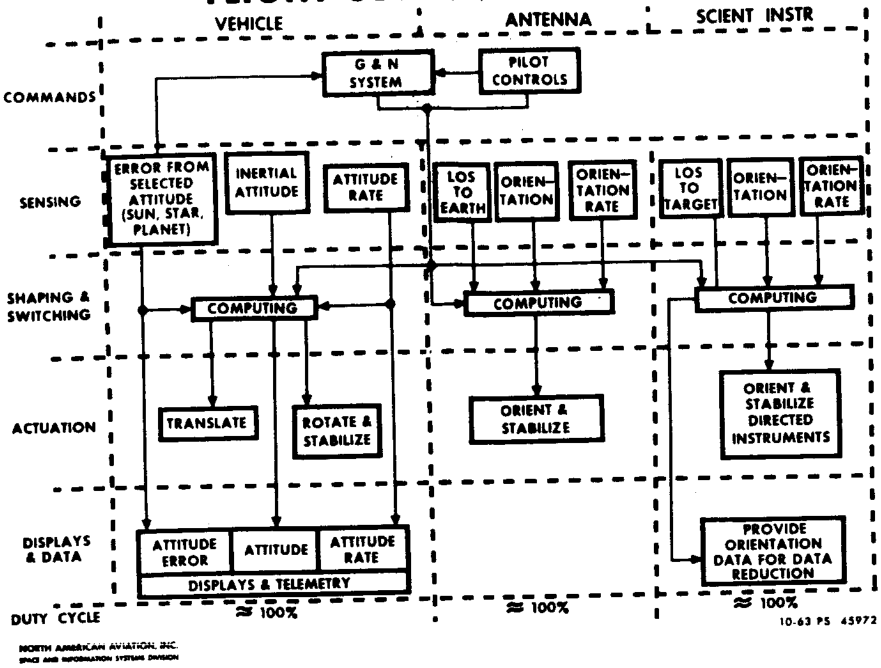


FIGURE 22

SIMPLIFIED BLOCK DIAGRAM OF ELECTRONIC SUBSYSTEMS INTEGRATION

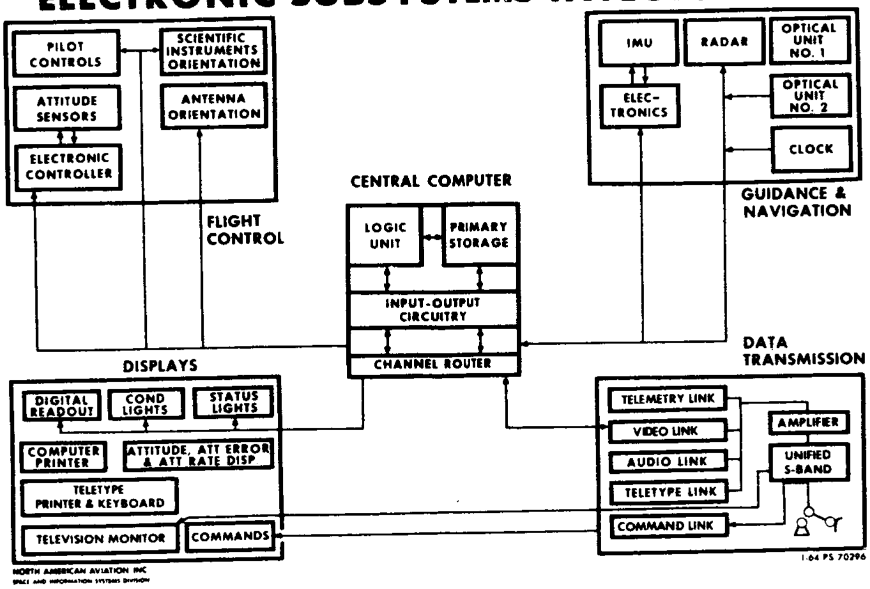


FIGURE 23

DATA PROCESSING DESIGN

ELEMENT	WEIGHT (POUNDS)	VOLUME (CUBIC FEET)	POWER (WATTS)
CENTRAL FRAME	82	2	300
PERIPHERAL EQUIPMENT	240	6	1200 (AVERAGE POWER) 3000 (PEAK POWER)
CONSOLES	45	2	750
	367	10	2250 (AVERAGE POWER) 4050 (PEAK POWER)

NOTE:

MAXIMUM COMPUTER STORAGE
32,000 WORDS X 36 BITS/WORD = 1.152×10^6 BITS

MAXIMUM COMPUTER SAMPLING RATE
15,000 WORDS/SEC X 36 BITS/WORDS = 240,000 BITS/SEC

NORTH AMERICAN AVIATION INC
SPACE AND INFORMATION SYSTEMS DIVISION

1-64PS70402

FIGURE 24

POWER LOAD SUMMARY

POWER REQUIREMENTS, (WATTS)

SUBSYSTEM	MAX. DURING MISSION		TRANSEARTH INJECTION		EMERGENCY	
	CONTINOUS	INTERMITTENT	CONT	INTERMITTENT	CONT	INTERMITTENT
LSS & ECS	4300		4300		1000	
FLIGHT CONTROL	1000			1000		300
DATA TRANSMISSION	1300	3550	1300	3450		600
GUIDANCE & NAVIGATION	550		550			400
SCIENTIFIC EXPTS	500		300			
SOLAR COLLECTOR		1450				
MISC	1020		950		500 (1)	1000 (2)
TOTAL			7400		1500	
MAX PROGRAMMED MAX UNPROGRAMMED			10700 11850		2500	

(1) INSTRUMENTATION DISPLAY, ILLUMINATION; (2) REPAIRS

NORTH AMERICAN AVIATION INC
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70399

FIGURE 25

POWER SOURCE WEIGHT

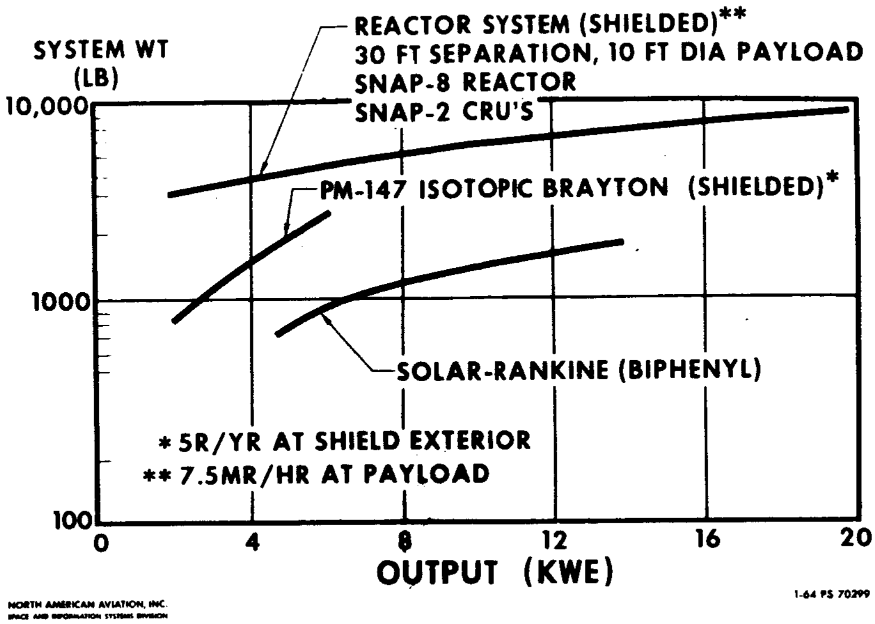


FIGURE 26

POWER SYSTEM SCHEMATIC

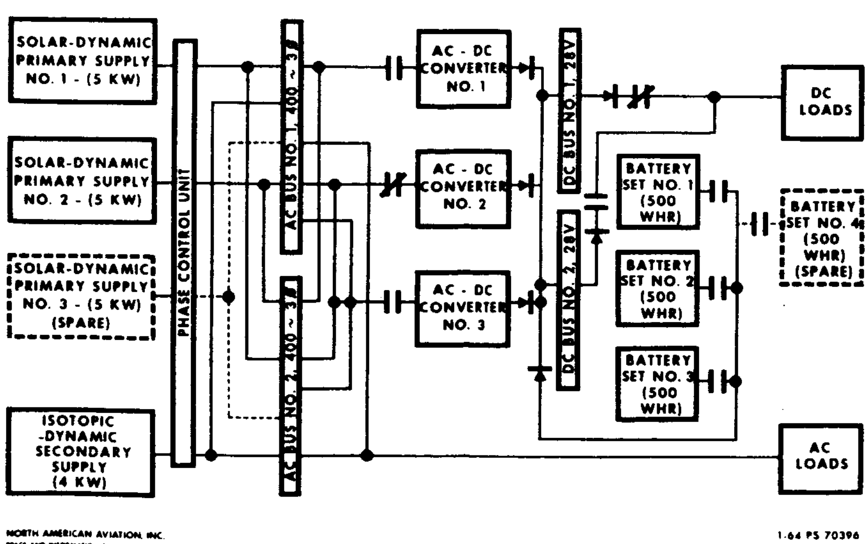
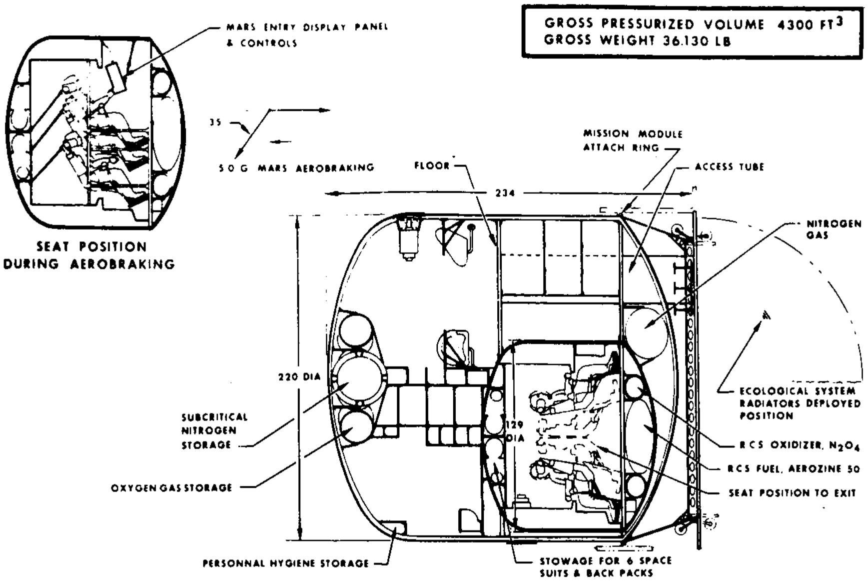


FIGURE 27

MISSION MODULE



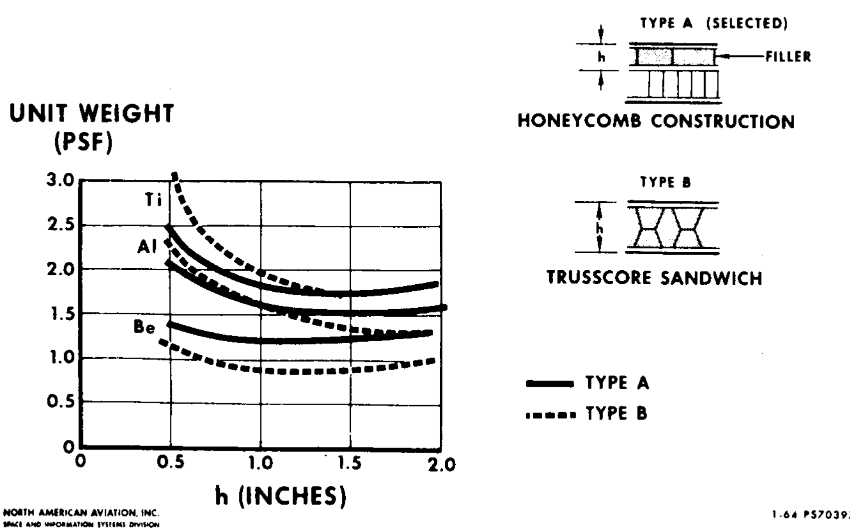
NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70305

FIGURE 28

EXTERIOR WALL UNIT WEIGHT METEOROID SHIELDING

P_o=.90, A=380 FT^2, T=420 DAYS



NORTH AMERICAN AVIATION, INC.
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS70397

FIGURE 29

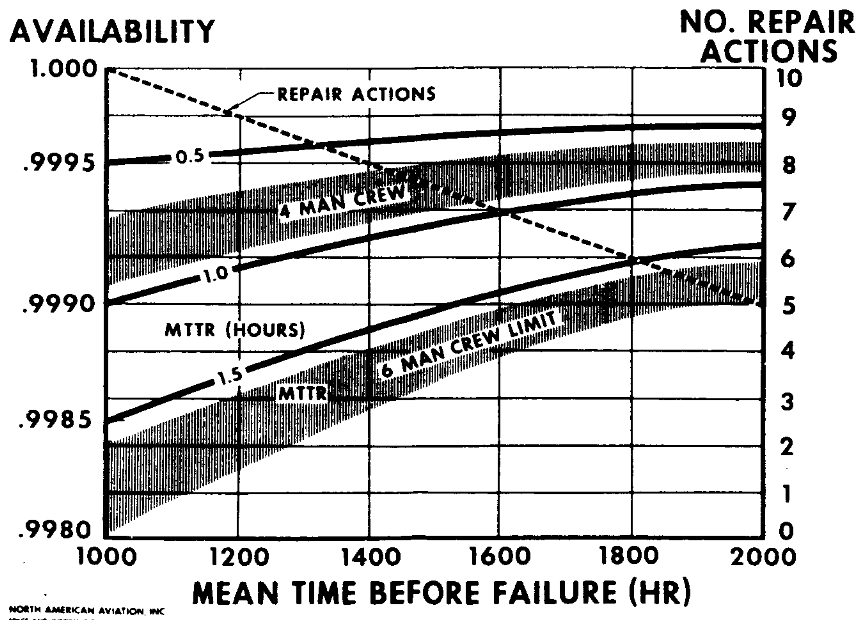
DAILY CREW ACTIVITIES
ESTIMATED

FUNCTIONS	HOURS/DAY BY CREWMAN: TRANS-MARS TRANS-EARTH CREWMEN 1	2	3	4	HOURS/DAY BY CREWMAN: MARS ORBIT CREWMEN 1	2	TOTAL HOURS: TRANS-MARS TRANS-EARTH	TOTAL HOURS: MARS ORBIT
S/C OPERATION								
MONITORING & CONTROL	8	8	8		12	12	24	24
HOUSEKEEPING	1	1	1	1	1	1	4	2
PERSONAL								
SLEEP	7	7	7	7	7	7	28	14
EXERCISE, HYGIENE, MEDICAL MONITORING	1	1	1	1	1	1	4	2
MAINTENANCE & REPAIR EXTRA VEHICULAR ACTIVITIES		2		4	1	1	6	2
SCIENTIFIC EXPERIMENTS	4	4	4	4	1	1	16	2
TRAINING								
PROFICIENCY								
EMERGENCY PROCEDURES	1	1	1	1			4	
OTHER								
RECREATION, UNSCHEDULED TIME	2		2	6	1	1	10	2
AVAIL MAN/HRS/DAY	24	24	24	24	24	24	96	48

NORTH AMERICAN AVIATION, INC
SPACE AND INFORMATION SYSTEMS DIVISION
9-63 PS 60206 A

FIGURE 30

CONTROL SYSTEM AVAILABILITY



NORTH AMERICAN AVIATION, INC
SPACE AND INFORMATION SYSTEMS DIVISION
1-64 PS 70301

FIGURE 31

MISSION MODULE WEIGHTS

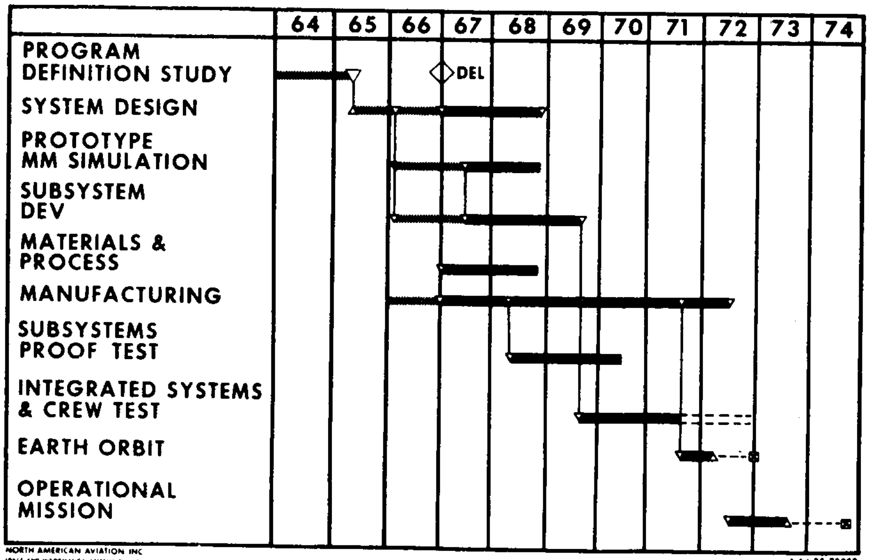
	EQUIPMENT (LB)	SPARES (LB)	EXPENDABLES (LB)
STRUCTURE	4550	15	
HEAT SHIELD & SUPPORT	1210	10	
LIFE SUPPORT	2890	460	4680
FLIGHT CONTROL	760	310	3320
NAVIGATION	290	170	
INFORMATION SYS & COMPUTER	1230	1135	
SCIENTIFIC EQUIP. & INSTR	3700		
AUXILIARY POWER	3700	1900	100
CREW FIXTURES	2170		
EXTENSION DEVICE	1700		
CONTINGENCY	2700		
TOTAL (37000 LB)	24900	4000	8100

NORTH AMERICAN AVIATION, INC
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70303

FIGURE 32

MISSION MODULE SUBSYSTEM DEVELOPMENT



NORTH AMERICAN AVIATION, INC
SPACE AND INFORMATION SYSTEMS DIVISION

1-64 PS 70302

26989

PART 12

STUDY OF A MANNED MARS EXCURSION MODULE

by

Dr. F. P. Dixon
Philco Corporation
Contract No. NAS9-1608

SUMMARY PRESENTATION

STUDY OF A MANNED MARS EXCURSION MODULE

By

Franklin P. Dixon
Manager, Advanced Space Systems
Aeronutronic Division, Philco Corporation

SUMMARY OF STUDY OBJECTIVES

The objectives indicated in Figure 1 were established by the Mars Mission Office of the Manned Spacecraft Center. The principal goal was the preliminary design of a feasible Mars Excursion Module for the manned Mars landing mission in the 1970 to 1975 time period.

The MEM performs the Mars entry, landing and return to the Mars Mission Module (MMM) or mother spacecraft in Mars orbit. In addition to the MEM and MMM a third spacecraft, the Earth Return Module (ERM), is required to return the crew and data to the Earth after return from Mars.

SUMMARY OF STUDY GUIDELINES

The MEM Study was performed under the guidelines shown in Figure 2. The desirable goal for a maximum weight on Mars orbit of 55,000 pounds proved to be optimistic. For the crew of three required by the MEM task analyses for a reasonable data return and experimental program on the Mars surface, the MEM weight is 58,000 to 67,000 pounds.

In addition to returning a minimum of 800 pounds of scientific payload from Mars to the Mission Module in Mars orbit, a minimum of 2,000 pounds of equipment for the surface experiments was to be delivered to the surface from orbit by the MEM.

MEM STUDY SCHEDULE

The MEM Study was initiated in May 1963 and was conducted in two phases (Fig. 3). The parametric study phase provided the board analyses for each major area. From these studies sufficient information was provided to arrive at the design point mission and vehicle choices for the preliminary design phase. The preliminary design was obtained in the second half of the study with the study conclusion in November 1963.

DESIGN POINT ORBIT AND TRAJECTORY CHARACTERISTICS

The values indicated for the Mars Excursion were the design point values of the important parameters based on the indicated considerations (Fig. 4). The important factors included the requirement for a lift-to-drag ratio (L/D) of 1.0 maximum to achieve lateral and longitudinal range control sufficient for site achievement, for error corrections during descent, and for reasonable descent initiation windows for the disorbit maneuver by the MEM from the MMM. γ_i is the entry angle; Λ is the Mars central angle or range angle from disorbit to entry initiation; $\Delta\Lambda_e$ is the variation in range angle at entry; Δt_e is the time from disorbit to entry initiation; ΔV is the disorbit velocity increment; and T/W is the Mars thrust-to-weight ratio for the MEM ascent stage. Partial staging indicates separation of empty tankage during the ascent phase.

DESIGN POINT VEHICLE SELECTION

The four aerodynamic shapes (Fig. 5) represent points on the parametric L/D curve and not specific vehicle choices. From the high-lift winged vehicle (L/D max = 3.6) to the near ballistic or Apollo type (L/D max = 0.5) the parametric study covered trajectory and heating characteristics. The second vehicle (L/D max = 1.15) was very close to the desired performance, however, the modified half-cone shown as the fourth vehicle more closely approximated the aerodynamic performance requirements (L/D max = 1.0) and was more easily treated from a packaging point-of-view. This last vehicle formed the basis for the preliminary design phase of the MEM Study.

MEM DESCENT SEQUENCE

For the most feasible Mars mission based on the present state of our knowledge, the MEM will be carried to Mars orbit attached to the MMM. Once the orbit parameters are established and the prelanding experimental probes and reconnaissance are completed the MEM descent phase will be initiated. The crew will enter the MEM in their protective suits and perform the checkout. The thermal and meteorite shield will be separated and the MEM separated to a distance of about one thousand feet by a relative velocity of five feet per second. The remainder of the sequence follows with deorbit, entry in the nose up attitude, parachute deployment below 100,000 feet altitude at Mach 1.5, reorientation into the tail down landing attitude for the tail lander version indicated, ignition of the retrorocket, hover and horizontal translation for up to 60 seconds, and vertical touchdown at the chosen landing site on Mars (Fig. 6).

MEM ASCENT SEQUENCE

After completion of the Mars surface operations phase or in event of abort before completion of the planned stay the crew will ready the MEM for ascent and execute the indicated sequence (Fig. 7). The command module and ascent engines will separate from the remaining living quarters and landing stage hardware. When the initial ascent tanks are empty they are separated to provide for minimum total weight. (This saves several thousand pounds in initial MEM weight.) A parking orbit is established until proper phasing between the MEM and MMM is achieved. Then the transfer to the MMM orbit is accomplished with rendezvous and docking maneuvers achieved with the lower thrust of the attitude control rockets. This allows the crew to drop the main propulsion rocket and tanks with only the command module performing the final maneuvers. After transfer of the crew with Mars samples and data, the MEM is discarded with final accomplishment of its mission.

MARS EXCURSION MODULE MARTIAN ENTRY

After going through the sequences of the entry and ascent the preliminary design conclusions are represented by Figures 8, 9, and 10. The artist's concept (Fig. 8) shows the modified 20-degree half-cone body at the entry point on the descent trajectory. The MEM enters with a pitched up attitude at a +24 degree angle-of-attack but at a -5 degree direction to the local horizon for the actual velocity vector. The attitude jets are used to maintain control until later in the entry when the elevons or aerodynamic control surfaces are useful. The retrorocket is seen in the aft closure and the bump beneath the plug nozzle is the airlock. The vehicle has a hot structure of Columbium or Titanium-Molybdenum with Nickel based alloy on aft surfaces.

MARS EXCURSION MODULE SURFACE OPERATIONS

After entry and landing the crew will establish the suitability of operations in the Mars surface environment then set up the experimental gear and gather samples in the immediate vicinity of the MEM. The tail lander version of the MEM is shown as the design choice in the artist's concept drawing (Fig. 9). Two of the three crew members are shown outside with the portable meteorological station and the 10-foot diameter antenna for communication with the Earth shown in the background. The crew windows are shown in the post entry exposed condition for landing and the airlock is the cylinder extended below the MEM. The four landing legs have crushable pads, as well as conventional shock absorbers. Two astronauts work outside while the third performs duties in the MEM.

ASCENT FROM MARS IN THE MARS EXCURSION MODULE

After completion of the stay on Mars the MEM crew will checkout the ascent stage, enter the command module, and depart the Mars landing site as shown in Figure 10. The ascent stage is just leaving the rest of the MEM with its automatic experiments remaining in operation as long as the Hydrogen-Oxygen fuel cell auxiliary power supply provides power. The data will be recorded on meteorological, thermal, and radiation experiments and sent to the Earth. The hemisphere to the right on the MEM landing stage is mothballed gear left for the possible use of later astronauts.

MARS ATMOSPHERE DENSITY-ALTITUDE SCHEDULE

At the start of the MEM Study a definitive model of the Mars atmosphere was not available from the broad range of possibilities. As is evident from Figure 11, even in March, 1963, two orders of magnitude variations in density at a given altitude were possible when comparing Mars atmosphere models of responsible investigators. In order to permit a reasonable design program it was necessary to establish a nominal atmosphere. The nominal Mars surface pressure was taken at 85 millibars with the following constituents: Nitrogen - 94% by volume, Carbon Dioxide - 2%, Argon - 4%, Oxygen-trace, Water-trace, Neon-trace, Xenon-trace, and Krypton-trace.

The atmospheric density limits were initially those due to Schilling. For the parametric phase the nominal atmosphere was the Aeronutronic model and for the preliminary design phase the NASA-MSC model indicated by the dots. Near the end of the MEM Study the recent data interpreted by JPL as a 20 percent CO_2 atmosphere with a surface pressure of 25 ± 14 millibars was postulated and this was used as a lower limit for study.

ORBIT ORIENTATION LOCI FOR SAMPLE 1971 MISSION

A study was made of the arrival and departure conditions versus stay time to determine if there was an orbit inclination which was compatible with both conditions (Fig. 12). The figure of the Mars gravitation field leads to perturbations similar to those of Earth satellites. That is advance of periapse and nodal regression for a posigrade orbit below about 63 degrees inclination. The perturbed locus is shown after 10 days in Mars orbit for the inclinations shown on the ordinate versus the longitude of the ascending node. The studies of this effect show that for orbits between 20 to 90 degrees an inclination exists for given arrival and departure conditions where no propulsive plane changes are required.

EFFECT OF CIRCULAR ORBIT ALTITUDE ON ORBIT RATE OF DECAY

The lower orbit altitude consistent with a rate of decay of 0.1 km/day was investigated to define a range of perigee altitudes acceptable for the Mars Mission Module stay in the Mars vicinity. The effect of the study for various atmospheres is shown in Figure 13 with the MEM design range of orbital altitudes between 300 and 600 kilometers. For study purposes, an altitude of 550 kilometers was chosen as the design point. A reasonable value of the ballistic coefficient for the Mars Mission Module as indicated on the abscissa would be around 5 to 10 $slugs/ft^2$.

LANDING FOOTPRINTS FOR REPRESENTATIVE VEHICLES

Three types of vehicles representing specific values of performance parameter are indicated (Fig. 14) with the landing footprints showing longitudinal and lateral range in central Mars angle. For the Apollo vehicle small maneuverability is indicated whereas the lifting body type shows reasonable performance for the Mars mission. The delta wing vehicle has a capability of varying longitudinal range to exceed a complete orbit of Mars. The lateral range of the lifting body offers sufficient performance for the MEM mission with reasonable flexibility.

G-LOADING DURING MODULATED ENTRY

From Figure 15, it is evident that the aerodynamic loads will be small compared to normal Earth entry from orbit. The maximum for all ranges of lifting body modulated entry (L/D = 1.0 max) being less than 1 Earth-g.

STAGNATION POINT CONVECTIVE HEATING RATE DURING MODULATED ENTRY

The heating rates for the design MEM entry to Mars from orbit are shown in Figure 16 with the heavy curve giving the design range of heating rates. The maximum will range from 25, 000 to 45, 000 BTU/ft$^{3/2}$ -sec where the $\sqrt{R}$ dependence has been included for convective heating referenced to 1-foot nose radius.

MEM ABORT ALTITUDE LIMITS

It is possible to abort the landing phase throughout a normal operation using the ascent propulsion (Fig. 17). The only critical point occurs at parachute deployment where a failure would not allow abort with only the ascent engine, since the surface would be impacted before the vertical velocity of 1500 feet per second could be removed for a thrust-to-Mars weight ratio of 1. 5 with the ascent engine. If parachute deployment is normal it would be possible to abort if a failure of the landing engine occurred.

PRIORITY OF EXPERIMENTAL DATA COLLECTION

From the study of surface operations it became apparent that a priority assignment of experimental functions was required. Figure 18 shows the relative priority of various experiments in order of importance to the first landing mission. Mission success is essential with gathering of data related to later missions as a secondary requirement. The biological evaluation of life forms is essential for the first purely scientific effort to allow for pre-contamination studies before man alters the Mars environment. The studies will be biological, geological-geophysical and meteorological in nature.

BIOLOGY STUDY PROGRAM

For the biological study program (Fig. 19) there are several branch or decision points that will affect the course of the investigations. The important result of pathogenic investigations is essential before safety of exterior ventures is guaranteed. The fundamental question of whether life does or has existed on our neighboring planets might well wait for the MEM landing to allow for adequate investigative resources. However, for an uncontaminated investigation unmanned surface probes may offer the only means of avoiding man's contaminating influence or deleterious back contamination from Mars to man. A 300-pound armored cassette will be used on the ERM to return samples from Mars to Earth. This will allow adequate protection from back contamination of the Earth's biosphere even if a catastrophic failure of the Earth Return Module were to occur during reentry.

GEOLOGY STUDY PROGRAM

The geological (really aerological) study program would have similar branching points as shown in Figure 20. The decision would be based on results obtained up to that point and observations of the Mars terrain. A reasonable local traverse could be run in 10 to 20 days and greater detail or wider area investigations would be allowed with larger stay times. Meteorological data is taken by automatic and manual operations throughout the Mars stay.

MARS MAP SHOWING A 70 DEGREE ORBIT GROUND TRACK AND THE EXTENSION OF THE SOUTH POLAR CAP

The small figures on the polar cap extension lines are days from the start of spring for the southern hemisphere (Fig. 21). Since life forms might follow the retreat of the cap and possible presence of water it might be desirable to land near a retreating pole. The north pole rate of retreat is slower and thus the dark mare area at 65 degrees north latitude in Cecropia might offer a good landing site. The choice of hemisphere, hence the site, will depend on the year of landing.

CAPTAIN-ASTRONAUT-SCIENTIFIC AIDE

The crew task analysis based on a reasonable scientific and engineering data return from the surface of Mars indicated a need for three crew members for the MEM rather than the minimum of two (Fig. 22). The first will be selected primarily for his flight and command capability. In addition, he will assist in the experimental programs on the surface (Fig. 22).

FIRST OFFICER-SCIENTIST-ASTRONAUT

The second crew member will be selected for geological and meteorological capabilities first and astronaut performance second (Fig. 23). He will be responsible for the experimentation related to these areas and will assist in flight duties and biological experimentation.

SECOND OFFICER-SCIENTIST-ASTRONAUT

The third crew member will also be first an experimenter and second an astronaut (Fig. 24). He will perform the biological programs and also be responsible for the crew health and physiological performance.

WORK-REST CYCLE

The three crew members will have a 12-on, 12-off cycle during the approximately 24 hour Martian day (Fig. 25). There will be a staggered cycle for each man with a two hour variation in each man's day. This provides for an 8-hour period with all three men on duty and also for an 8-hour period when all three crew members are asleep. The important systems, such as environmental control, will be set with malfunction alarms to awake the crew when failures occur. Two crew members will be outside with the third inside during surface operations.

GENERAL ARRANGEMENT - TAIL SITTER VEHICLE

Figure 26 shows the outside envelope of the MEM Tail Lander configuration. The modified half-cone body has the dimensions given in inches with the nose and base cone radius shown by R. The rounded portion of the cone is down during atmospheric flight and the top is capped with a slightly convex surface for packaging and fabrication efficiency as well as aerodynamic performance.

INBOARD PROFILE MEM CONFIGURATION D9-I

Figure 27 shows the basic design choice MEM. This Tail Lander vehicle has four legs with crushable pads for landing and living quarters in the lower part of the cone base for surface operations. The command module is at the top and is the control cabin for descent and ascent.

MEM WEIGHT SUMMARY - TAIL LANDER

The weight table (Fig. 28) shows the range of weights for the three crew 40-day MEM to be from 58,070 to 66,964 Earth pounds before deorbit at Mars. The minimum is the expected weight with minimum velocity and structural factor contingencies for a successful mission guarantee. The nominal weight MEM is for a highly contingent or maximum success design.

GENERAL ARRANGEMENT - CANTED LANDING VEHICLE

The envelope of the canted lander is shown in Figure 29. The dimensions in inches show that a slightly larger MEM results from the requirement to land in a favorable nose high attitude. The width is about four feet greater than the tail lander MEM and about three feet longer. The main disadvantage other than size and weight is the tip-off problem for ascent vehicle launch from Mars at a 45 degree angle to the horizon versus vertical launch for the Tail Lander.

INBOARD PROFILE MEM CONFIGURATION D8-I

The Canted Lander is shown in Figure 30 with its relative landing attitude in the side view. Note that the landing engine points downward through the conical surface. The plug nozzle engine is identical to the Tail Lander. This version lands on three legs with crushable pads.

SUMMARY WEIGHT ENVELOPE

The variation in weight for various design mission parameters is shown by Figure 31. The propulsion system weight factor is the dry weight over the dry weight plus propellant weight for the propulsion system. The nominal is the Canted Lander which weighs about two thousand pounds more than the Tail Lander. In order to attain the desired 55,000 pound goal for a crew of five it would be necessary to reduce the propulsion safety factor load to an unreasonable value and to reduce structural factors to a marginal figure.

GROSS WEIGHT VERSUS MARS STAY TIME AND CREW NUMBER

By reducing the Mars stay time to 10 days and taking a crew of two, the MEM Canted Lander can attain the desired weight (Fig. 32). Neither of these adjustments is reasonable since it would reduce the experimental program to a level that might not justify the expense of the trip to Mars and it would render Mars surface operations and flight emergency operations marginally critical for a crew of two. The Tail Lander would of course be about two thousand pounds lighter than the Canter Lander for any point on this figure.

GROSS WEIGHT VERSUS SPECIFIC IMPULSE

Figure 33 shows the weight variation for the Nominal Canted Lander MEM with a change in propellant specific impulse. The design point value of 360 seconds is for the OF_2 + MMH propellant. Improvement of data to provide for higher specific impulse would not appear to offer enough weight reduction to reach the 55,000 pound goal for the Canted Lander or even the Tail Lander MEM. The best performance will probably not exceed 375 seconds vacuum impulse for this fuel-oxidizer combination.

GROSS WEIGHT VERSUS NUMBER OF ASCENT STAGES

The nominal MEM ascent vehicle of two stages with staging of primary tanks at about 7000 feet per second velocity attainment (Fig. 34). The single engine will stay with the vehicle for later thrust increments. It is evident that little gain is to be realized by going to three stages. The actual savings with tank staging is about 8000 pounds in initial MEM weight rather than the 12,000 pounds shown for complete staging including two separate engines.

MEM BASIC AERODYNAMIC DATA TABLE

Comparison of the trimmed MEM theory and test data converted to the MEM configuration shows an interesting variation (Fig. 35). Based on the analyses performed the importance of actual test data for the design vehicle is evident.

MEM DECELERATION AND RETRO SYSTEM OPTIMIZATION

The parachute-retrorocket parametric study showed that a terminal velocity of approximately 175 feet per second with the parachute and MEM drag effects provides a minimum landing system weight (Fig. 36). This was for the nominal MEM Martian atmosphere and would be altered for the minimum density at the surface of 25 $\pm$ 14 millibars predicted by interpretations of recent spectroscopic data.

RADIATIVE HEATING TESTS IN CO_2 - N_2 ATMOSPHERES AT AMES

Recent estimates offer higher CO_2 content of 20 percent for the Martian atmosphere (Fig. 37). If this is true, the smaller radiative heating rates of 2 percent of the convective peaks for entry from Mars orbit and the 2 percent CO_2 atmosphere would be increased accordingly. The entry velocity of about 11,500 fps (3.5 km/sec) would offer a lower heating rate than indicated here. For direct entry at high velocities the radiative component is much more important and produces severe peak heating at velocities above 20,000 fps.

HEAT TRANSFER DISTRIBUTION FROM STAGNATION POINT

Figure 38 shows the heating rate on a sphere-cylinder body as a function of relative distance from the stagnation point. Also, the convective heating is compared with turbulent heating to show the relative importance. Fortunately, when turbulent effects occur, the peak heating has passed and the total heating on the after body does not reach excessive rates during this activity. The variations shown in these curves were used in estimating the heating rates for the MEM configuration back from the stagnation point.

PEAK TEMPERATURE DISTRIBUTION ON VEHICLE AT $\alpha = 30°$

Figure 39 shows the temperatures on the MEM vehicle at an angle-of-attack of 30 degrees at the maximum heat transfer point of the Mars entry trajectory. The fall off in temperature is more rapid on the top than on the underside in this nose up attitude. The gradient from nose to tail allows for use of lower temperature alloys on the after part of the MEM.

PEAK TEMPERATURE DISTRIBUTION ON VEHICLE AT $\alpha = -10°$

Figure 40 shows the higher temperature peak for maximum heating rate at a negative angle-of-attack (-10°). Here the after heating on the top of the vehicle exceeds that on the underside. The gradients are steeper but the higher stagnation temperature gives fairly high temperatures on the aftermost part of the MEM.

ESTIMATED HEATING RATE (MAXIMUM RANGE)

The effect on materials of the maximum heating rates are shown in Figure 41 for the design structures of MEM. Columbium is not affected by these temperatures but the 50 percent Mo-Ti and Inconel X are affected if the temperatures exceed about 2500° and 1400°, respectively. The total heating on the right and the heating rate on the left are shown as a function of time for the maximum range or maximum heating rate entry. The parameter is nose radius of curvature in feet. The design point MEM has R = 4 feet but the heating rate is less for the modulated entry.

TEMPERATURE TIME HISTORIES FOR MODULATED ENTRY

For the design point entry the effects start at about 500,000 feet where the maximum lift nose-up attitude is employed to give maximum deceleration (Fig. 42). This continues until skip-out would begin. At this point, around 250,000 feet constant altitude is held by nosing down and then raising the nose until maximum lift is reached again. Descent then starts in this attitude (24° angle-of-attack) until a velocity of 1500 fps is reached near an altitude of 75,000 feet. Here the parachute is deployed. The temperature is shown for the stagnation regions and the surfaces with the effect of turbulent flow indicated around 1400 seconds from start of the atmospheric entry.

MEM PROPULSION SEQUENCES

Figure 43 shows the various propulsion phases of the MEM portion of the Mars mission. There are increments required for deorbit, attitude control, retro-hover-landing, abort, ascent, orbit insertion, orbit transfer, and rendezvous with docking. The intermediate orbit is essential to proper phasing from abort and to widening of the ascent launch windows to reasonable values.

MEM MAIN PROPULSION INCREMENTS

The main propulsion systems for the MEM require the velocity increments shown in Figure 44. Gravity effects are included, thus these values give the actual performance requirements for each subsystem velocity increment. The rendezvous velocity increment is part of the attitude control system capability and the total characteristic velocity of 19, 370 fps is independent of this final requirement. The contingencies shown are deemed to be conservatively adequate for the design point MEM mission.

DELIVERED SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE

The specific impulse as a function of chamber pressure is shown (Fig. 45) for a plug nozzle engine with an expansion ratio of 40 to 1 and a bell nozzle engine for various expansion ratios in order to indicate the obvious performance advantage to be gained by the altitude compensating engine. A value of 360 seconds was employed in the main propulsion system performance estimates and for estimating the fuel, oxidizer, and tankage requirements. The main engine for deorbit, retro, hover and landing is a 5 to 1 throttleable 30, 000 pound maximum thrust plug nozzle engine with a 6° gimbal.

The ascent engine is a fixed-thrust 18, 000-pound plug nozzle engine. Both are capable of multiple restart and both are pump fed at a chamber pressure of 1225 psi to reduce the overall diameter of the engines. The optimum range of thrust for the ascent engine goes from 18, 000 to 30, 000 pounds and the smallest diameter engine was chosen for this MEM design.

PLACEMENT OF MEM DESCENT VEHICLE RCS ENGINES

The reaction control nozzles for the MEM are designed in two systems for the descent and ascent vehicles due to differing requirements (Fig. 46). The oxidizer is N_2O_4 with MMH fuel to avoid the cryogenic plumbing requirements for the OF_2 oxidizer. The N_2O_4 is stored near each nozzle group. The descent RCS has 12 nozzles of conventional radiative cooled design. These are used in conjunction with the elevons to provide attitude control during the deorbit, descent, entry, and landing phases. Each descent RCS nozzle provides 200 pounds of thrust. Total impulse for the descent RCS is 100, 000 lb-sec.

PLACEMENT OF THE ASCENT RCS ENGINES

The ascent attitude control is provided by the 6° gimbal of the main ascent engine when thrusting and the 14 RCS nozzles when coasting (Fig. 47). The RCS nozzles for ascent consist of 14 radiation cooled and 4 ablative nozzles, each with a thrust of 100 pounds. The placement on the MEM ascent vehicle consisting of the command module and the ascent engine with tankage is shown schematically in this drawing. Total impulse for the ascent RCS is 150,000 lb-sec to allow for ascent to intermediate orbit, orbit injection, separation of the command module from the engine and tanks, and rendezvous with docking (250 fps ΔV).

ELECTRICAL POWER DUTY CYCLE

The power requirements are shown in Figure 48 for the entry, abort, and launch phases of the MEM mission. The design primary electrical system is a H_2 - O_2 fuel cell with a 2.5 kilowatt electrical power output. The back-up system indicated for primary EPS failure is a 1.0 kw Ni-Cd battery system capable of 10 hours of operation. The systems indicated in Figure 48 requiring power are the Guidance and Control System, Primary Atmospheric Control System, Communications, Thermal Control System, Secondary Atmospheric Control System, and Auxiliary Systems. The 2.5 kw PEPS is capable of operation in all required conditions of Mars environmental extremes.

REQUIRED MEM SUBSYSTEMS

The subsystems required for life support and thermal control are summarized in Figure 49. A dual atmosphere with 3.5 psi O_2 and 3.5 psi N_2 is employed in the MEM living quarters and command module. CO_2 control is achieved with an amine absorption and rejection drum. The backpack ECS is an O_2 atmospheric system at 3.5 psi with LiOH CO_2 absorption. Thermal control protects against the extremes of Mars thermal environment keeping a shirt-sleeve environment from -90° F to +85° F outside temperature. While in transit aboard the MMM from Earth to Mars the MMM meteorite-thermal shield protects the MEM from extremes of heat loss or gain. The expendables are stored as liquids with sub-cooling for all but the hydrogen providing adequate reserves for the transit plus 40-day stay on Mars. The hydrogen requires a dual vacuum insulation tank with a cooled hydrogen gas expansion "curtain" separating the concentric cryogenic tank layers. The hydrogen is maintained as a liquid in the inner chamber.

Water is obtained from the H_2 - O_2 fuel cell and from reclaimed waste liquids. This is partly boiled away for thermal control at elevated outside temperatures or is reused by the crew. Solid wastes are not reclaimed.

PRIMARY ATMOSPHERIC CONTROL SYSTEM

Figure 50 shows the primary ACS, O_2, CO_2, and relative humidity are controlled and trace contaminants are removed by this system. N_2 and O_2 are supplied as needed from cryogenic storage. The crew suits can tap off this system. This is used for surface operations. The secondary ACS is used for flight or emergency conditions. It is similar in operation except that LiOH is used for CO_2 absorption. The PACS is located in the living and service area and the SACS is in the command module.

LIFE SUPPORT AND THERMAL CONTROL SYSTEMS

Figure 51 indicates the weight variation in expendables as a function of stay time on the Mars surface from 10 to 40 days. For a successful exploration, the first mission should have a minimum stay of 10 days but 40 days would provide a much greater return of useful experimental data.

TOTAL POSITION ERRORS

The guidance and control system based on Apollo state-of-the-art appears adequate for the MEM mission if the MMM ephemeris is updated from the MEM landing site during the Mars stay. Figure 52 lists the one-sigma errors for the design point mission under these conditions. The main error at landing would have been corrected visually by the pilot to actually land at the desired spot during the entry and hover maneuvers.

GUIDANCE AND CONTROL FUNCTIONAL SCHEMATIC

Figure 53 shows the MEM Guidance and Control system. Two modes of operation are shown with the programmed autopilot on an open-loop phase and the alternate guidance phase. The open-loop phase represents the initial ascent portion or launch phase, which might last up to 200 seconds, followed by the guidance phase until burnout of the ascent engine. The RCS provides corrections in attitude based on the G&CS error signals.

SCHEDMATIC, MEM RADAR SYSTEM

The MEM double mode radar is used for landing and rendezvous with the landing mode indicated by L and the rendezvous mode by R in Figure 54. For landing, the system provides range and velocity data in three dimensions to provide for the pilots display enough information to allow touchdown with negligible horizontal drift and a vertical velocity less than 10 fps. In the rendezvous mode the radar provides range and range rate data with relative orientation and the location of the MMM and MEM. The range capability for this maneuver is 12 nm with skin track on MMM and 250 nm with a transponder.

AVERAGE DAILY DATA LOAD

The communications system must be capable of transmitting a total of 5×10^7 bits of information per day from the Mars surface. The form of data is indicated in Figure 55. The weather station daily output of 15,000 bits and the radiosonde output of 22,000 bits are small compared to the other data outputs. The page reader transmits charts, drawings, or tables at the rate of 50 per day and the image system transmits slides, sample images, or photographs at the same daily rate. The total information capability can be used for transmission to the MMM or to the Earth directly and was used in design of the communications system.

EARTH-MARS-SUN GEOMETRY

For the 1970-75 time period investigated in the MEM Study the geometry of the three bodies are shown (Fig. 56) to provide the following constraints on the communications system. The maximum range from Earth to Mars is 102 million miles. The sun is outside the second lobe of a parabolic 10-foot antenna on Mars.

VISIBILITY OF EARTH FROM MARS, 1971-1976

The visibility ratio is shown (Fig. 57) as a function of latitude on Mars for the three opposition landing periods in the 1970-75 launch time for a Mars mission. From these curves it is apparent that the high latitude landing sites are desirable for higher direct communications duty cycles, but the north or south latitude choice depends on the year of landing. In order to have greater than 50 percent line-of-sight from the MEM landing area to Earth it would be

best to land above 70° south latitude in 1971 and 1973 but above 70° north latitude in early 1976 for the 1975 launch date. It is evident that this consideration could greatly influence the landing site choice.

VISIBILITY OF ORBITING MMM FROM MEM LANDING SITE

This plot of visibility time (5 degrees and higher above the Mars horizon) of the MMM from the MEM shows that only 7.7 percent of the time is useful for communicating to MMM. Also, a fast slowing antenna or hemispherical coverage antenna is desirable. With the data coverage to simplify the MEM surface mission it is possible to transmit only as much data as could be sent directly to Earth with a visibility ratio of 0.5. Since this can be exceeded by the proper choice of landing site it appears best to send the data directly to Earth with only limited communication with the MMM on narrow bandwidth and low duty cycle.

The MEM design system uses 100 watts of power into a clock driven parabolic antenna (10 feet in diameter) to transmit information with a 3 kcs bandwidth to the 210-foot DSIF antennas on Earth. This is in the S-band frequency range. A power of 83 watts into a discone hemispherical antenna provides S-band commucation from the MEM to the MMM.

MARS MISSION DEVELOPMENT PLAN

The national space program necessary to accomplish the Manned Mars Landing Mission is shown in Figure 59. The arrangement tiers are indicated as an effort to demonstrate the management complexity. The MEM is only a part of the overall program which has been separated into six basic functional areas. The Apollo lunar mission, the unmanned planetary probes to Mars, development of Earth rendezvous operations, and verification of man's capability to operate adequately for extended periods (up to 400 days) in space are considered to be prerequisite or concurrent developments with the Mars landing development program.

MEM DEVELOPMENT SCHEDULE

For only the MEM portion of the Manned Mars Landing Mission, a schedule period of nine years is required for a normal development program that is neither too long (or more expensive) nor too accelerated (and more expensive) (Fig. 60). Apparently, the necessary developments in technology

and hardware could be accomplished during this period if immediate approval were given at the national level with adequate funding. Of course, this is only a part of the requirement and the auxiliary elements must also be pursued at a compatible rate for this to be worthwhile. It should be noted that the 10-year program including the actual mission requires selection and training of the astronauts in the third year. This is due to the complexity of the experimental programs, flight training, and the MEM subsystems with requirement for intricate knowledge on the part of all three crew members as to the functions of each other. The design of the MEM required for the Mars mission is such that extensive (although apparently quite feasible technically) development of various subsystems must be accomplished to make the technological transition from the lunar or near Earth orbital missions to Mars.

MARS EXCURSION MODULE COST ANALYSIS

Due to the required developments that must be charged directly to the MEM, it is not unreasonable to develop an upper limit of total MEM Program Costs (exclusive of the Earth based costs during the Mars mission) according to Figure 61. The development of Columbium hot structure, OF_2 - MMH altitude compensating engines, highly reliable and easily maintained subsystems capable of 40-day operation after a 200-day transit to Mars, and verifying the systems operation under conditions that cannot be readily duplicated for the Mars entry, landing, and ascent add to the estimated costs compared to previous space programs. The total of 6.2 billion dollars for a 10-year program represents design, development, test, and provision of a mission and back up MEM with a total of 11 MEM systems for test and application. It is not inconceivable that this would represent about 15 to 20 percent of the actual mission and development costs. Estimation of total program cost is extremely difficult this far in advance, but all recognizable costs have been included conservatively to attempt an assessment of the upper limit for MEM.

MARS EXCURSION MODULE FUNDING PROFILE

The relative phasing of the funds is shown in Figure 62 for the 6.2 billion dollars in support of the MEM Program. The peak funding per year is shown between the fifth and seventh year reaching a maximum of approximately one billion dollars per year at this time. A true picture of the funding profile for the total Mars mission would probably peak slightly earlier on a relative scale when the booster development testing and the near Earth operational tests for the other program elements are included.

CONCLUSIONS

A manned landing on Mars from Mars orbit in the MEM is feasible in the 1970-75 time period provided the necessary interplanetary spacecraft (MMM). Earth Return Spacecraft (ERM), launch boosters, and Earth-orbit-to-trans Mars injection propulsion systems are developed concurrently. A MEM with L/D of the 1.0 offers sufficient flexibility to perform the assigned mission with high probability of successful return of the crew, samples, and experimental data to justify the expedition. A crew of three is required. Altitude compensating engine development is required. Planetary probes by unmanned spacecraft, orbital rendezvous capabilities development, system developments for the Apollo mission, efficient interplanetary rocket engines, and verification of man's space operation capability are essential.

AERONUTRONIC DIVISION

FIGURE 1

SUMMARY OF STUDY OBJECTIVES

1. PRELIMINARY DESIGN OF THE MARS EXCURSION MODULE
2. DEFINITION OF ORBITAL OPERATIONS AND TRAJECTORY CHARACTERISTICS
3. DEFINITION OF ENTRY VEHICLE CHARACTERISTICS
4. DEFINITION OF TOUCH-DOWN VEHICLE CHARACTERISTICS
5. DEFINITION OF ABORT AND ASCENT VEHICLE CHARACTERISTICS
6. DEFINITION OF PRINCIPAL VEHICLE SUBSYSTEMS
7. DEFINITION OF SURFACE OPERATIONS
8. DEFINITION OF CREW SIZE AND TASK ANALYSIS
9. ESTABLISH VEHICLE WEIGHT REDUCTIONS FOR STAY TIMES LESS THAN 40 DAYS
10. INTEGRATION OF STUDY RESULTS WITH OTHER MARS MISSION CONTRACTORS

FIGURE 2

SUMMARY OF STUDY GUIDELINES

GENERAL	
MISSION TIME	1970 – 1975
MAXIMUM ACCELERATION LIMITS	10 EARTH g'S
GUIDANCE TECHNIQUE	APOLLO STATE-OF-THE-ART
MEM	
SURFACE STAY TIME	10-40 DAYS
PROPELLANT	OF_2-MMH
GROSS WEIGHT ON MARS ORBIT (MAXIMUM)	55,000 LB
WEIGHT OF SCIENTIFIC PAYLOAD RETURN TO ORBIT (MINIMUM)	800 LB
MINIMUM CREW SIZE	2 MEN
MAINTENANCE PHILOSOPHY	COMPLETE ON BOARD CAPABILITY
SUPPORT PHILOSOPHY	ALL EXPENDABLES AND SUPPLIES ON BOARD
PRINCIPAL INTERFACES	
TOTAL MISSION TIME	390 – 440 DAYS
MMM CREW	6 MAXIMUM, 4 NOMINAL
MMM AERODYNAMICS	MEM INTEGRATION COMPATIBLE WITH AERODYNAMIC BRAKING ENTRY AT MARS
MMM PHYSICAL	SEPARATION AND DOCKING INGRESS AND EGRESS
MMM TRAJECTORY	DEPART AND RENDEZVOUS

Ford Aeronutronic Division

FIGURE 3

MEM STUDY SCHEDULE

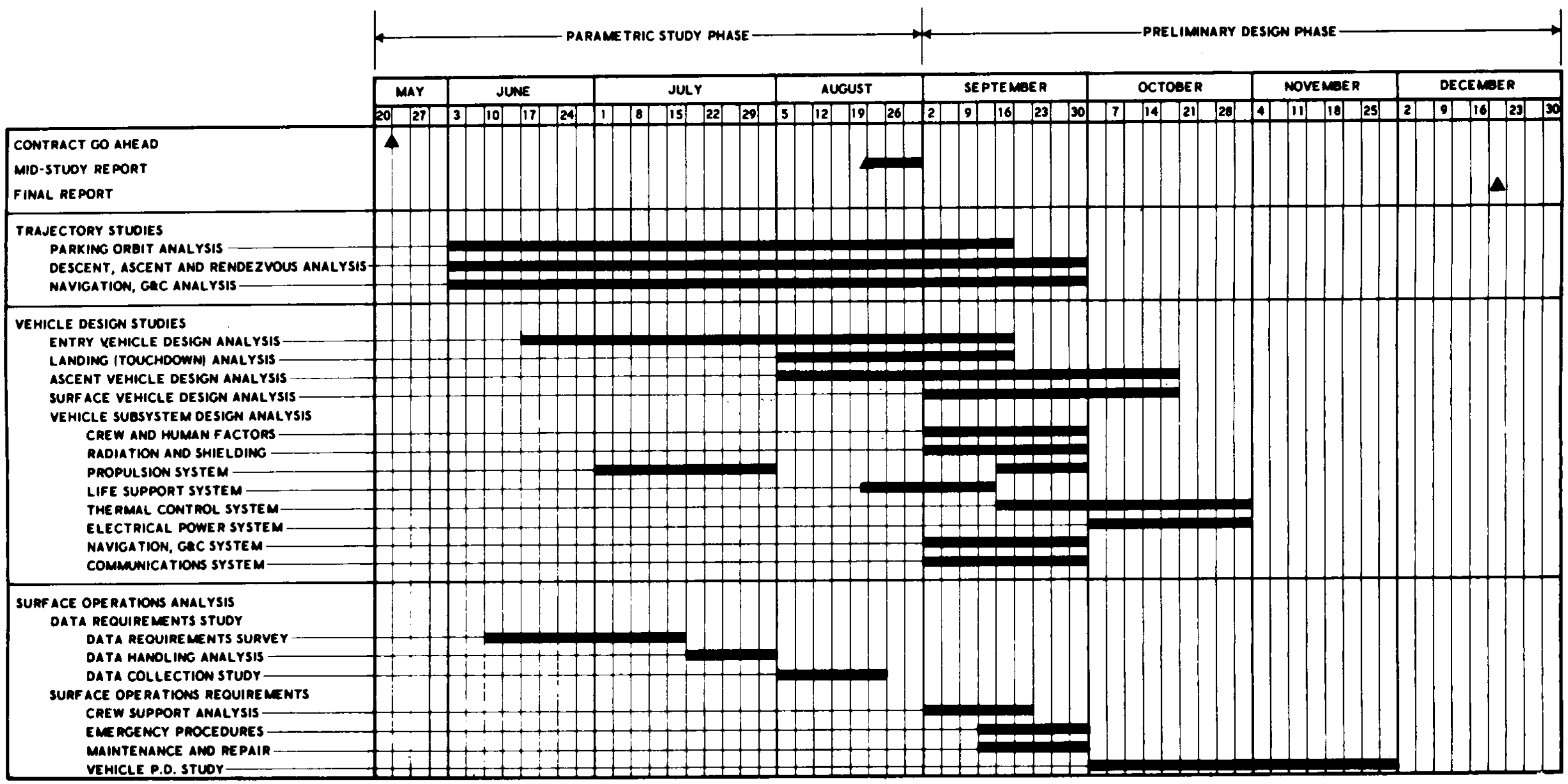


FIGURE 4

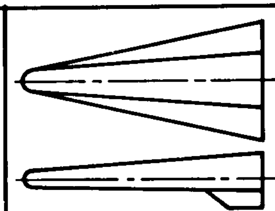
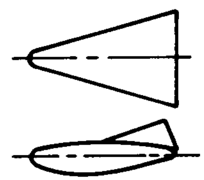
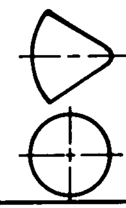
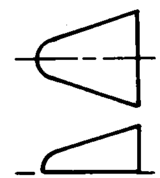
DESIGN POINT ORBIT AND TRAJECTORY CHARACTERISTICS

PARAMETER	CONSIDERATIONS	DESIGN VALUE	
PARKING ORBIT PERIAPSE ALTITUDE	ENERGY REQUIREMENTS DECAY RATE RECONNAISSANCE	550 KM	
PARKING ORBIT INCLINATION	ENERGY REQUIREMENTS SITE ATTAINMENT RECONNAISSANCE COVERAGE	70°	
PARKING ORBIT ECCENTRICITY	GUIDANCE CONSIDERATIONS RENDEZVOUS CONSIDERATIONS	.025	
DE ORBIT MANEUVER	ENERGY REQUIREMENTS ATMOSPHERIC ENTRY REQUIREMENTS GUIDANCE CONSIDERATIONS MEM-MMM RELATIONSHIP	γ_i Λ $\Delta\Lambda_e$ ΔV Δt_e	-5° 90° 4° 550 FPS 28 MIN
ENTRY CHARACTERISTICS	LATERAL AND LONGITUDINAL RANGE REQ. (ERROR CORRECTION AND SITE ATTAINMENT) ACCELERATION LEVEL ALLOWABLE HEATING	L/D	1.0
ASCENT AND RENDEZVOUS	CROSS ORBIT REQUIREMENTS T/W RATIO STAGING ACCELERATION LEVEL ABORT CAPACITY	RANGE ANGLE T/W	10° NOMINAL 1.5 - 2.5 PARTIAL STAGING

AERONUTRONIC DIVISION

FIGURE 5

DESIGN POINT VEHICLE SELECTION

				
LENGTH (FT)	67.3	41.6	20.0	25.8
SPAN (FT)	33.8	25.0	23.0 (DIA)	26.0
USABLE VOL (FT^3)	2950	2950	2950	2950
WEIGHT (LB_e)	66 K	58 K	52 K	56 K
L/D (MAX)	3.6	1.15	0.5	1.0
g (MAX)	1.3	2.3	4.2	2.6
LATERAL RANGE (Λ) (TYPICAL) (km)	66° 3820	15° 870	4° 232	10° 580

AERONUTRONIC DIVISION

FIGURE 6

MARS EXCURSION MODULE DESCENT SEQUENCE

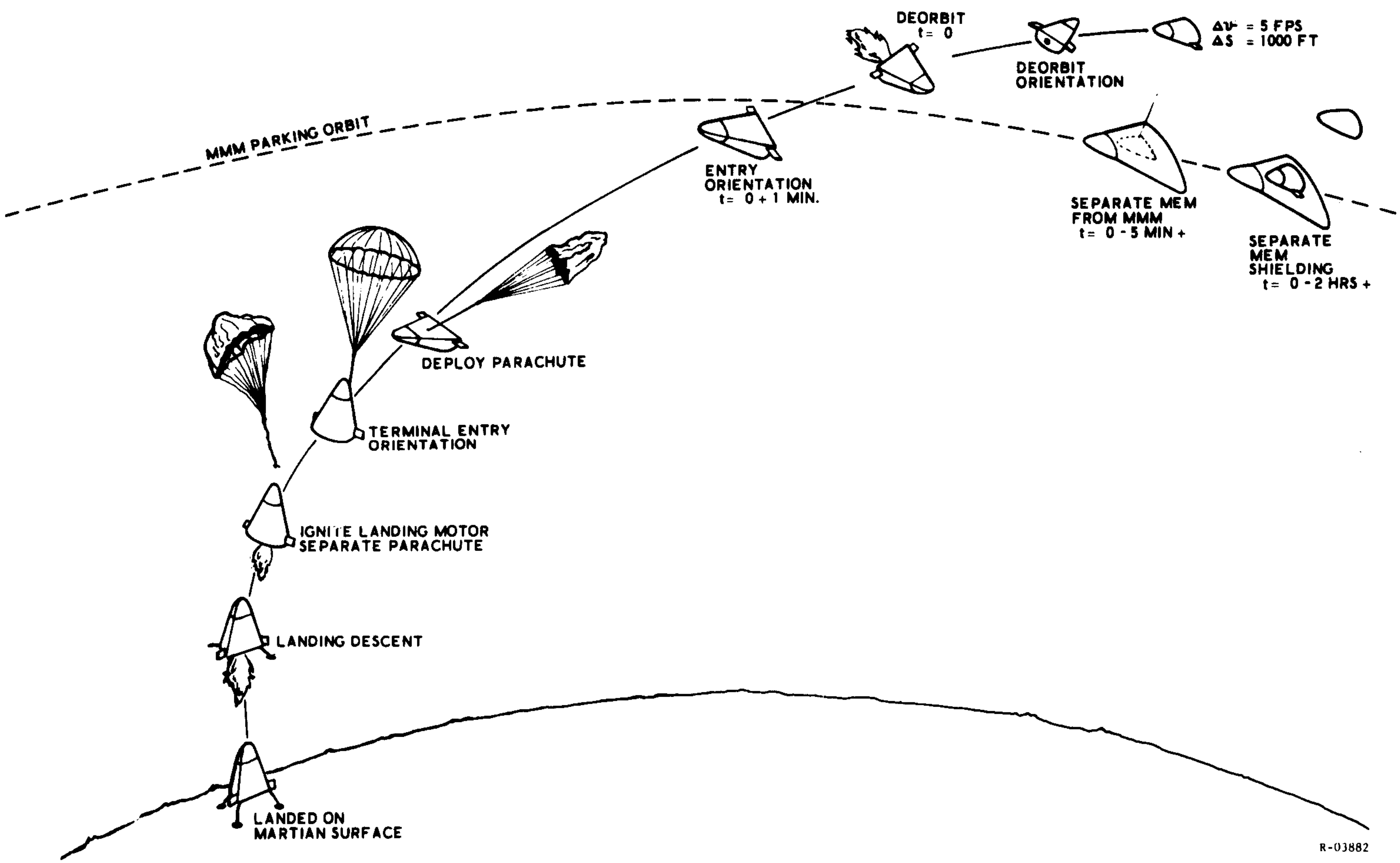


FIGURE 7

MARS EXCURSION MODULE ASCENT SEQUENCE

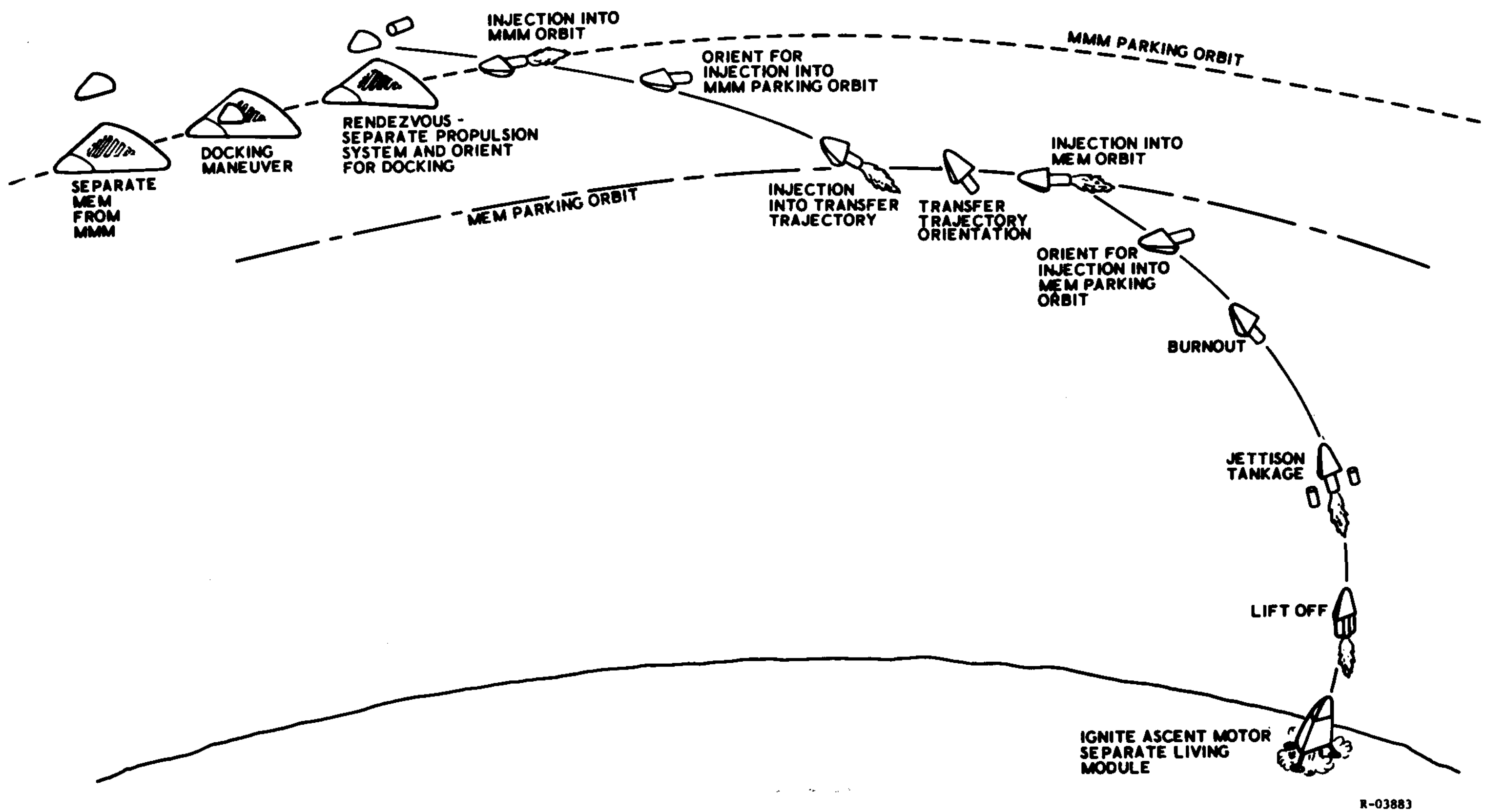


FIGURE 8

MARS EXCURSION MODULE MARTIAN ENTRY

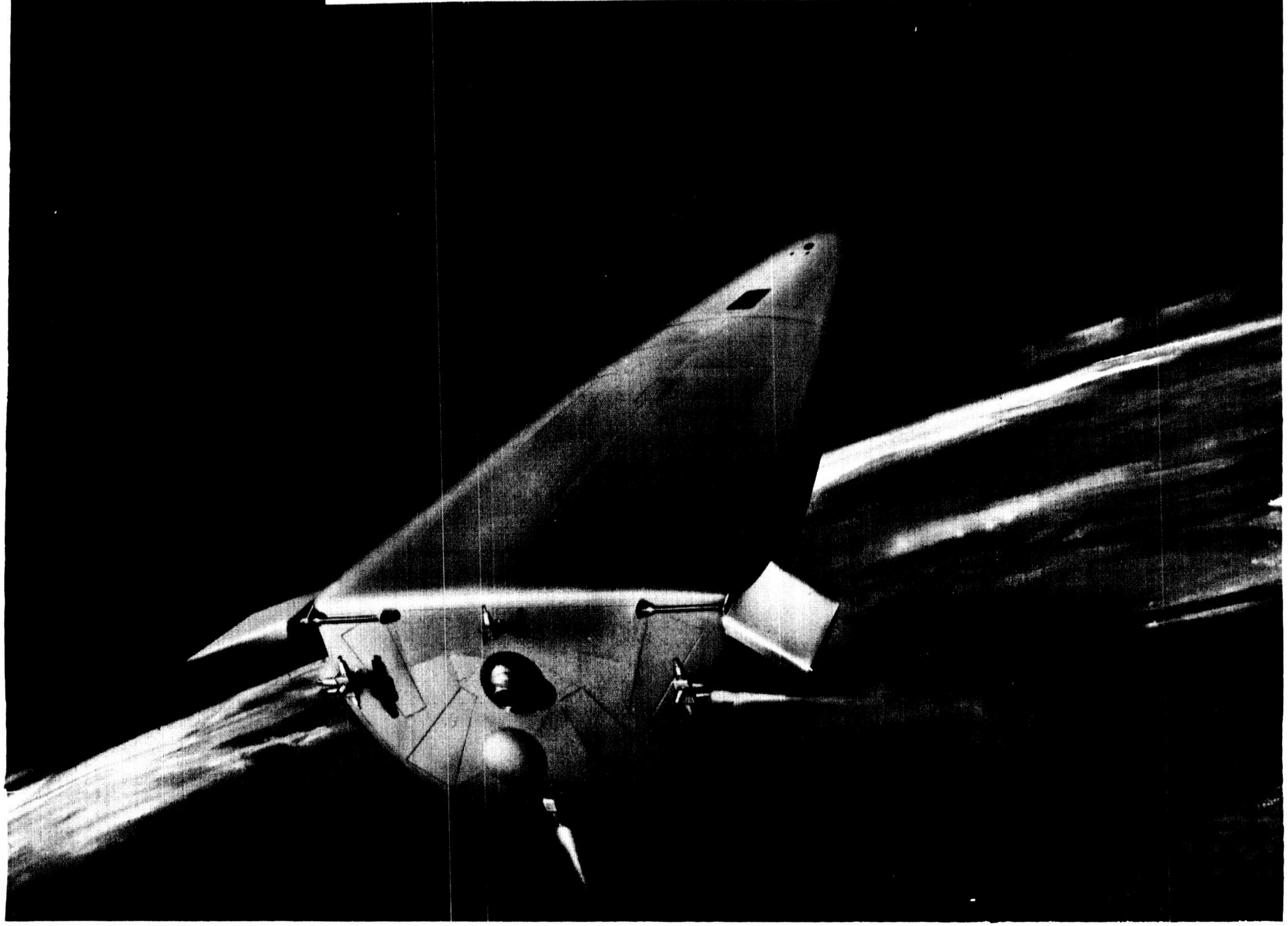


FIGURE 9

MARS EXCURSION MODULE SURFACE OPERATIONS

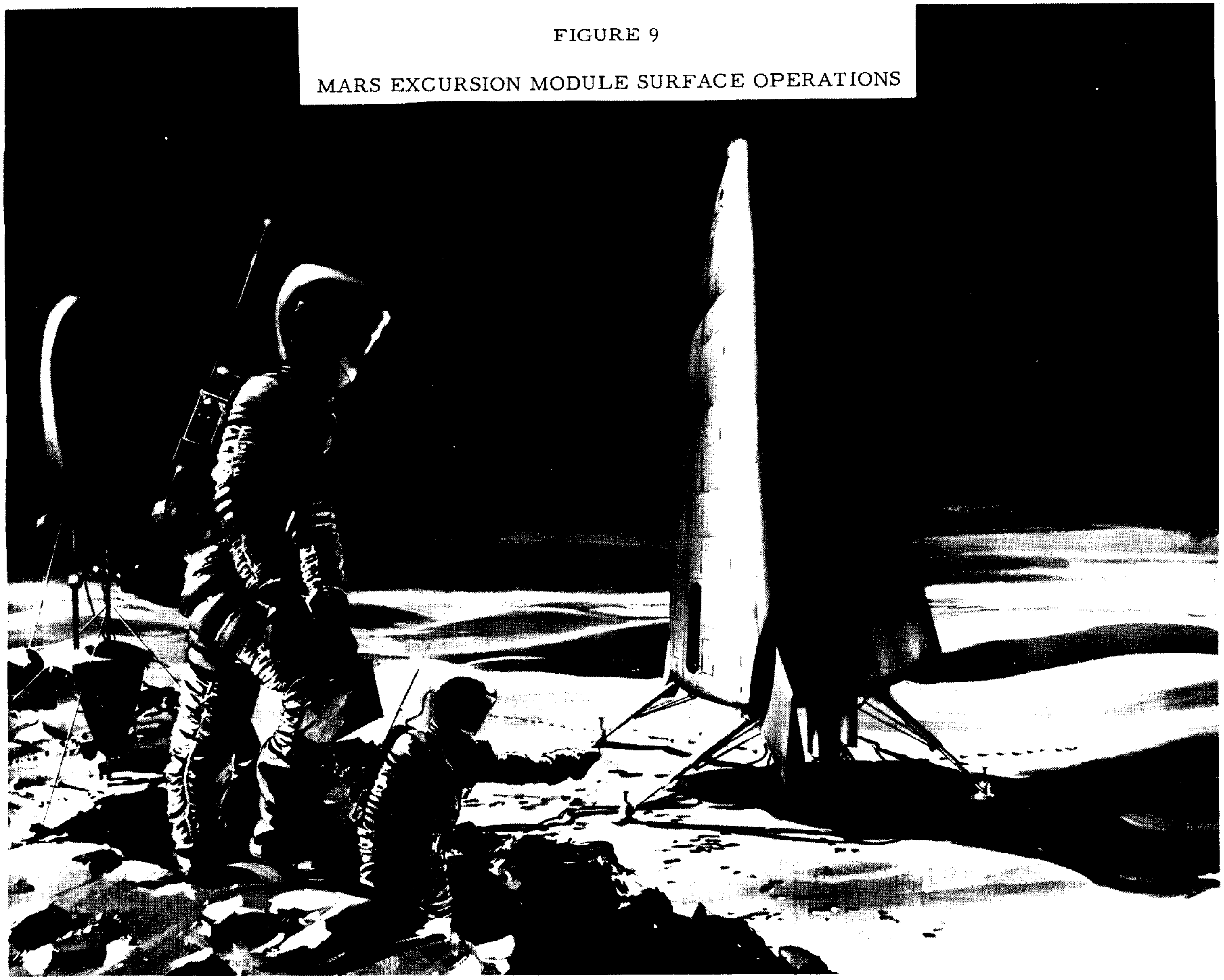


FIGURE 10

ASCENT FROM MARS IN THE MARS EXCURSION MODULE

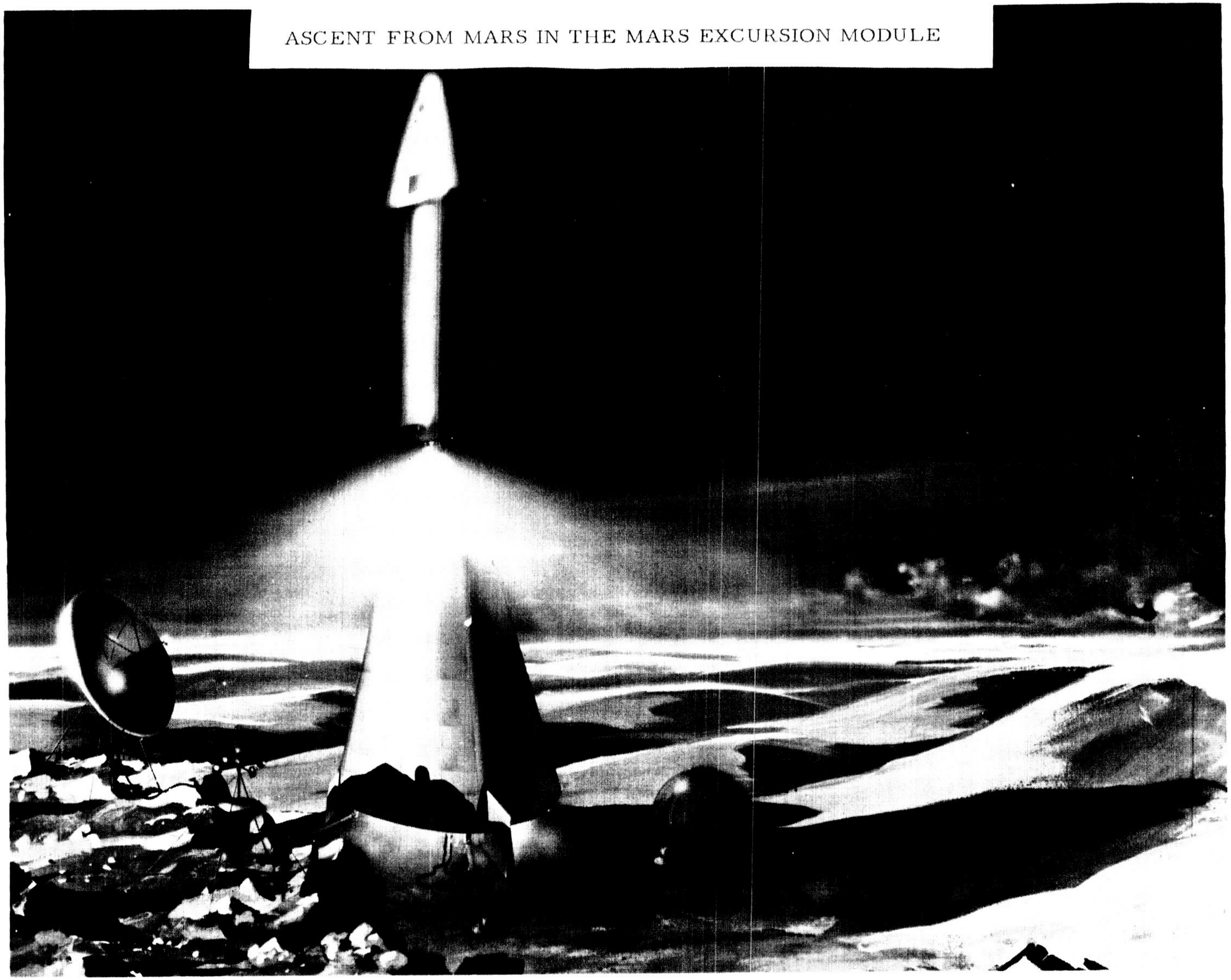
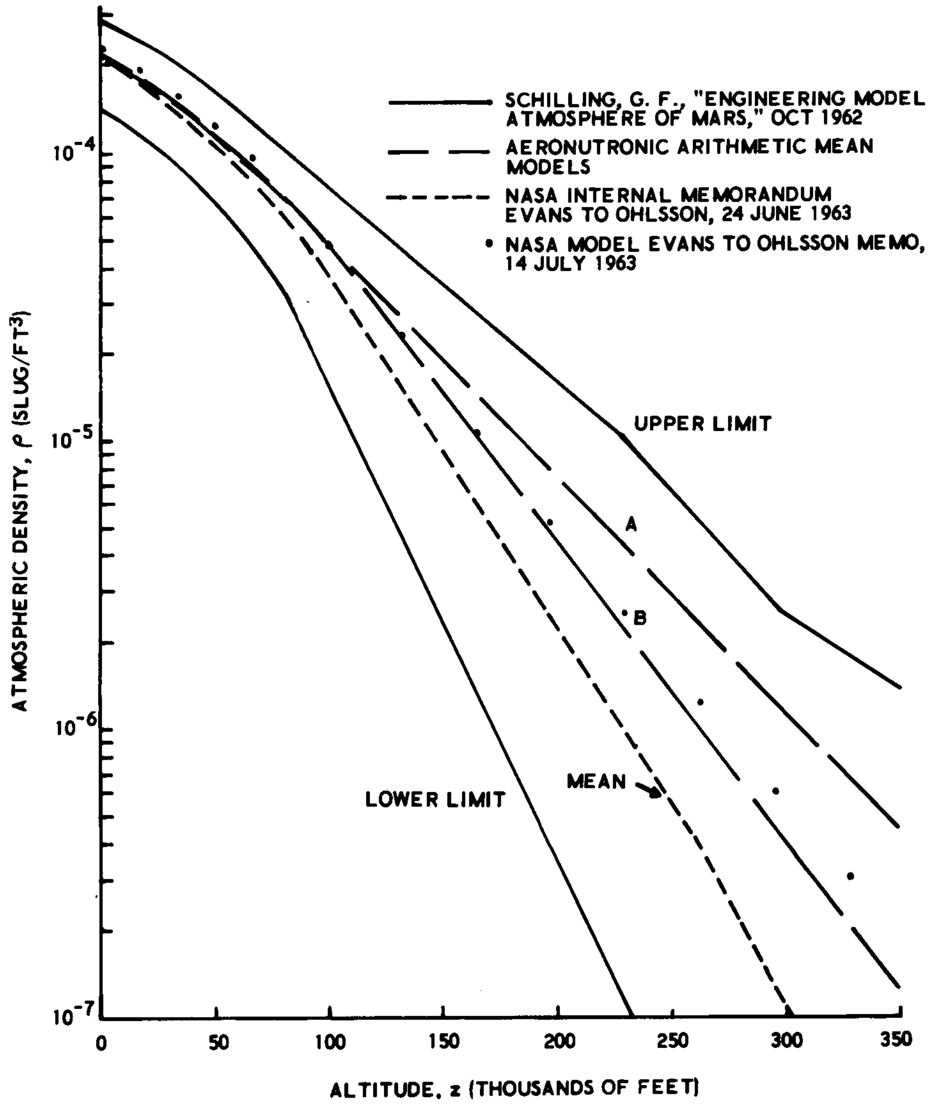


FIGURE 11

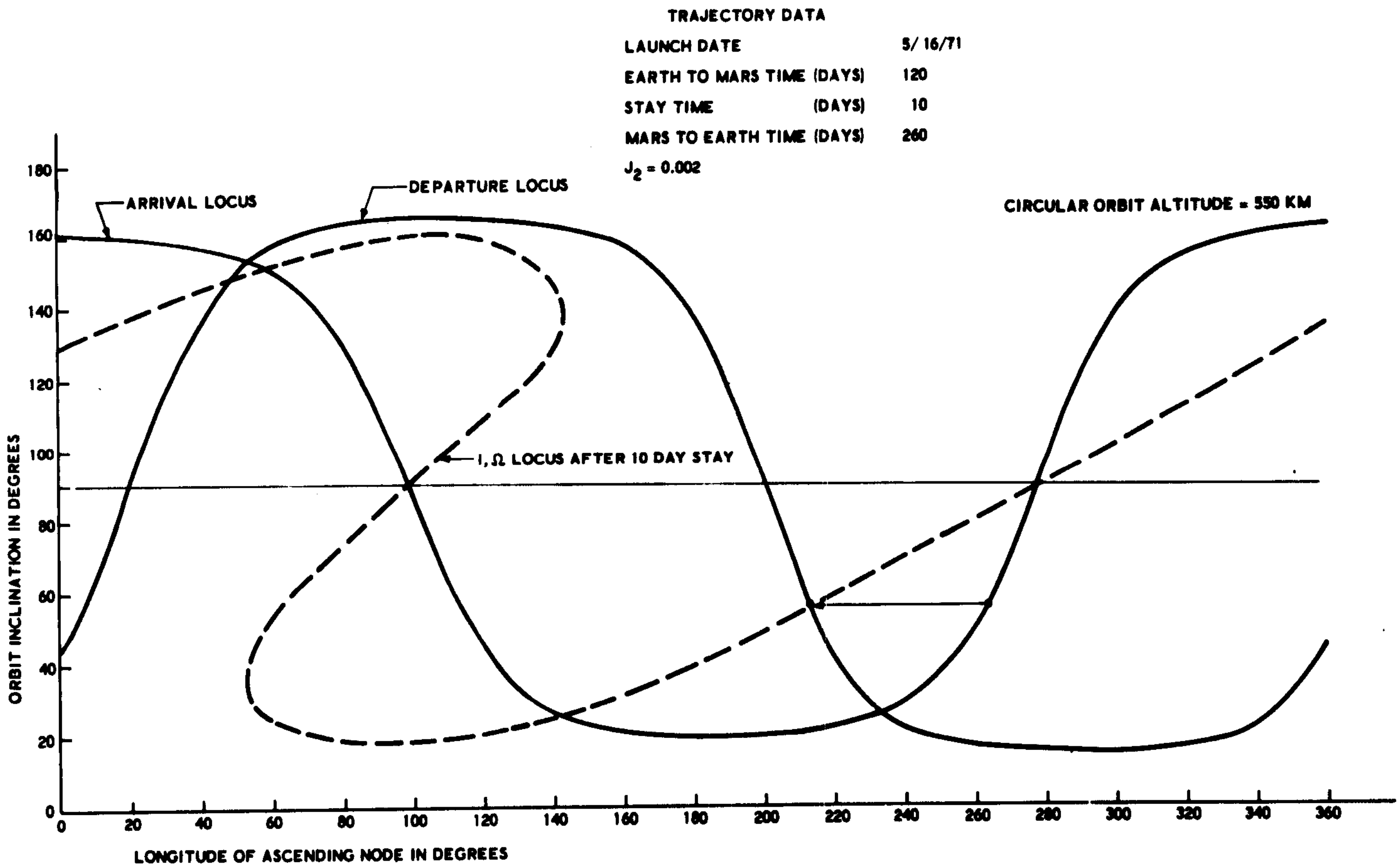
MARS ATMOSPHERIC DENSITY – ALTITUDE SCHEDULE



Ford
AERONUTRONIC DIVISION

FIGURE 12

ORBIT ORIENTATION LOCI FOR SAMPLE 1971 MISSION



Ford
AERONUTRONIC DIVISION

FIGURE 13

EFFECT OF CIRCULAR ORBIT ALTITUDE ON ORBIT RATE OF DECAY

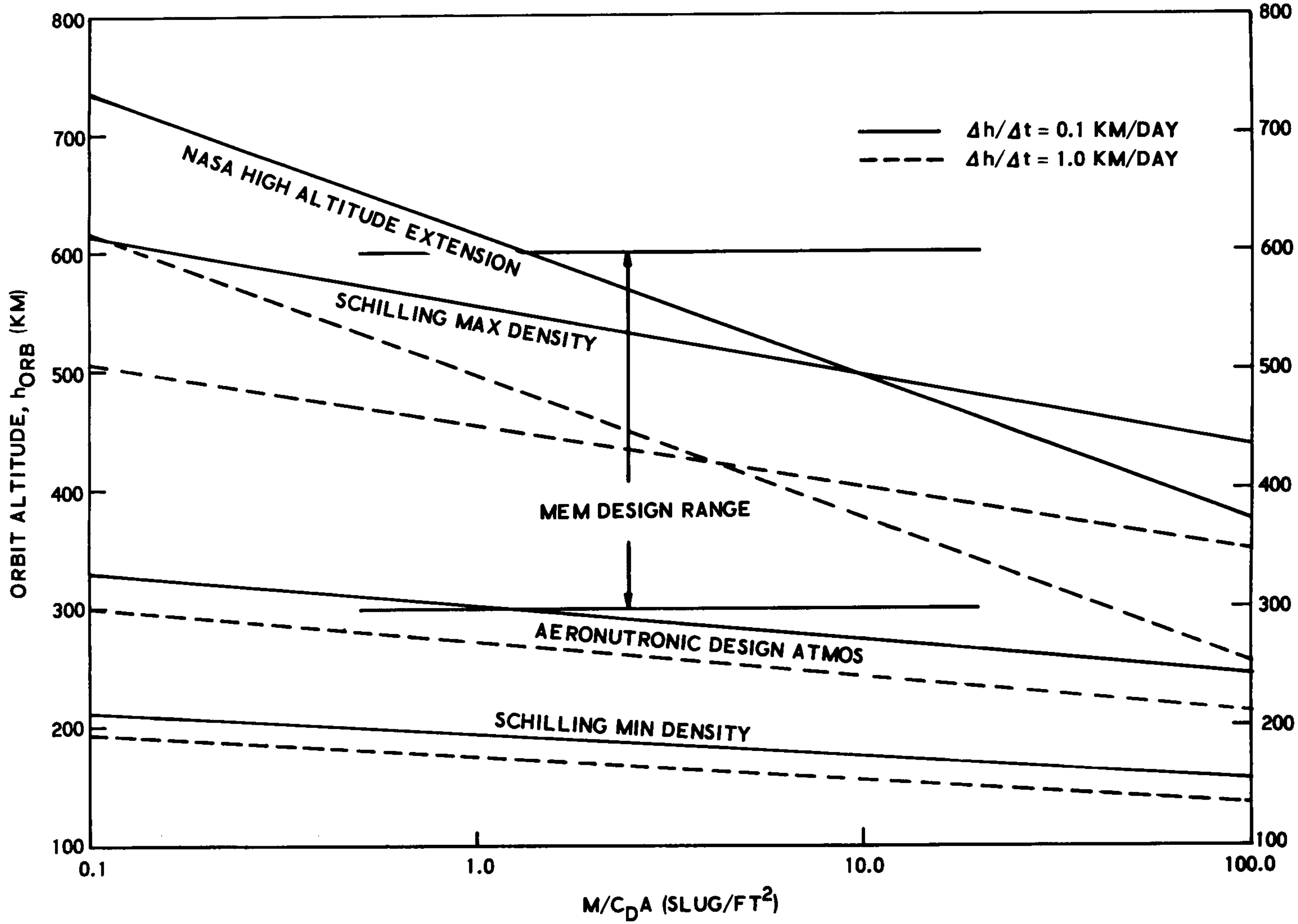


FIGURE 14

LANDING FOOTPRINTS FOR REPRESENTATIVE VEHICLES

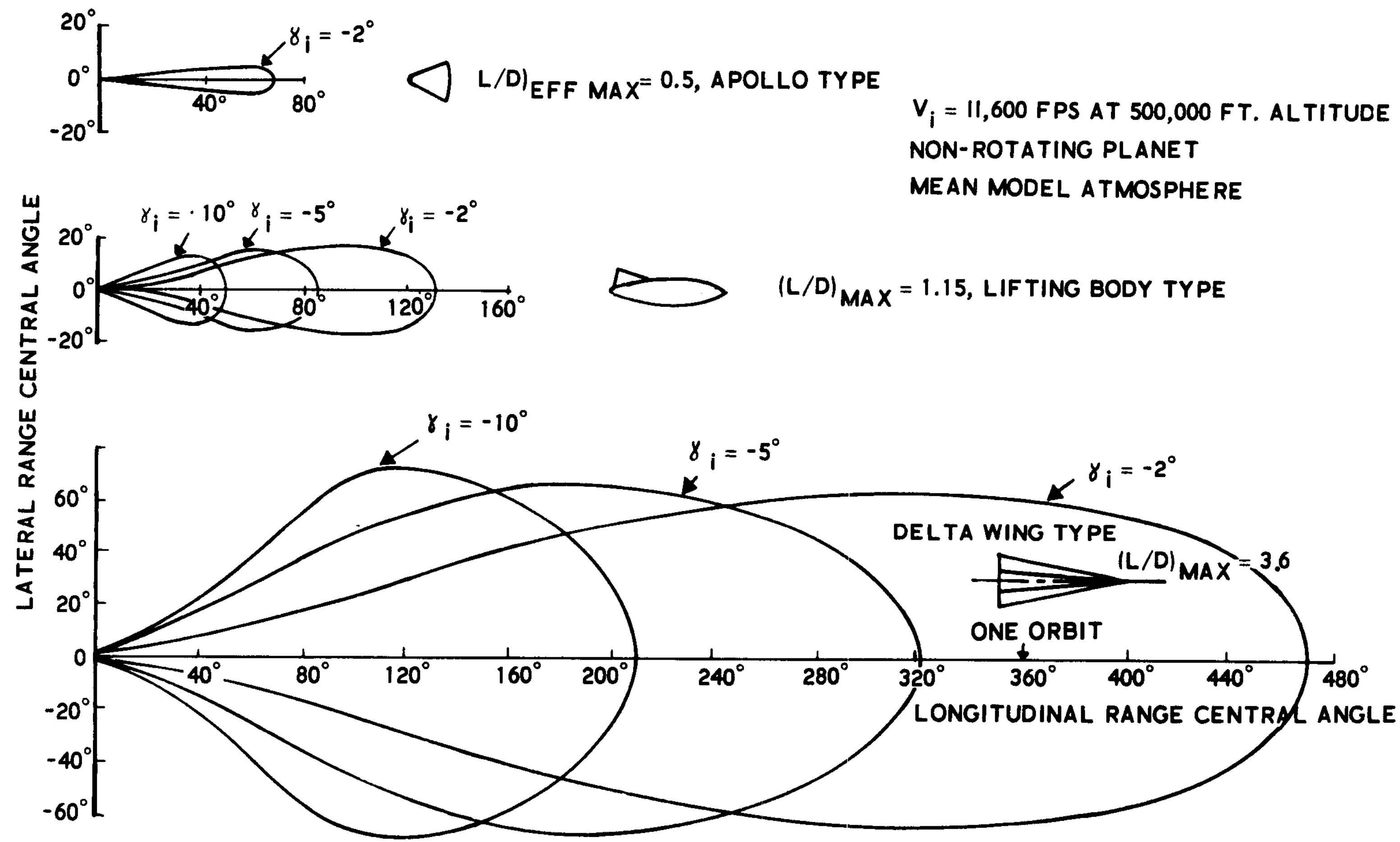


FIGURE 15

G-LOADING DURING MODULATED ENTRY

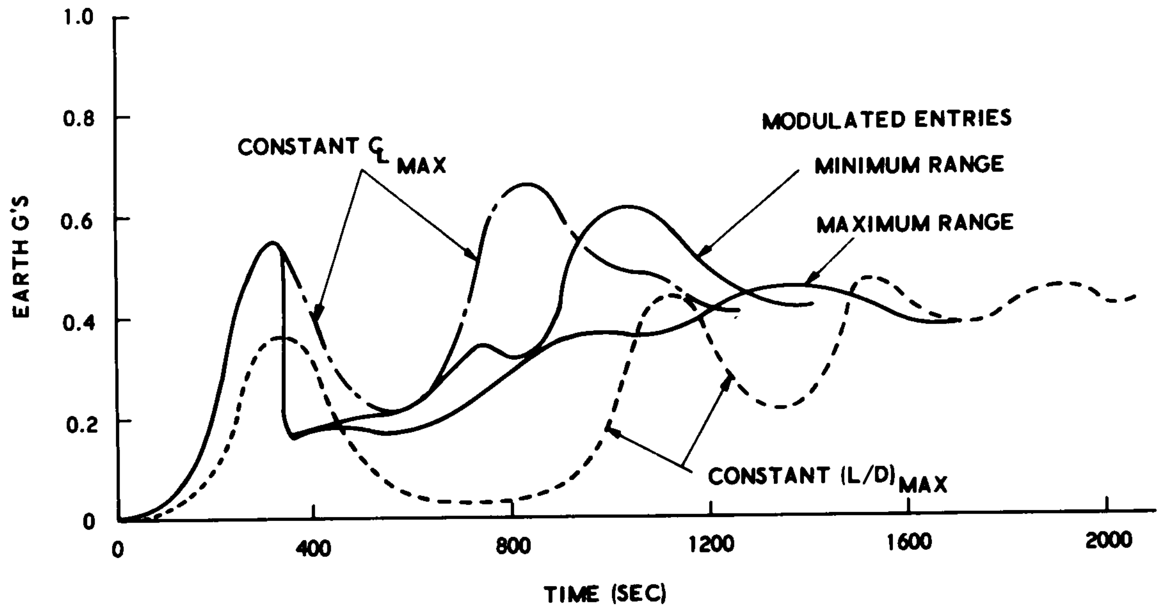


FIGURE 16

STAGNATION POINT CONVECTIVE HEATING RATE
DURING MODULATED ENTRY

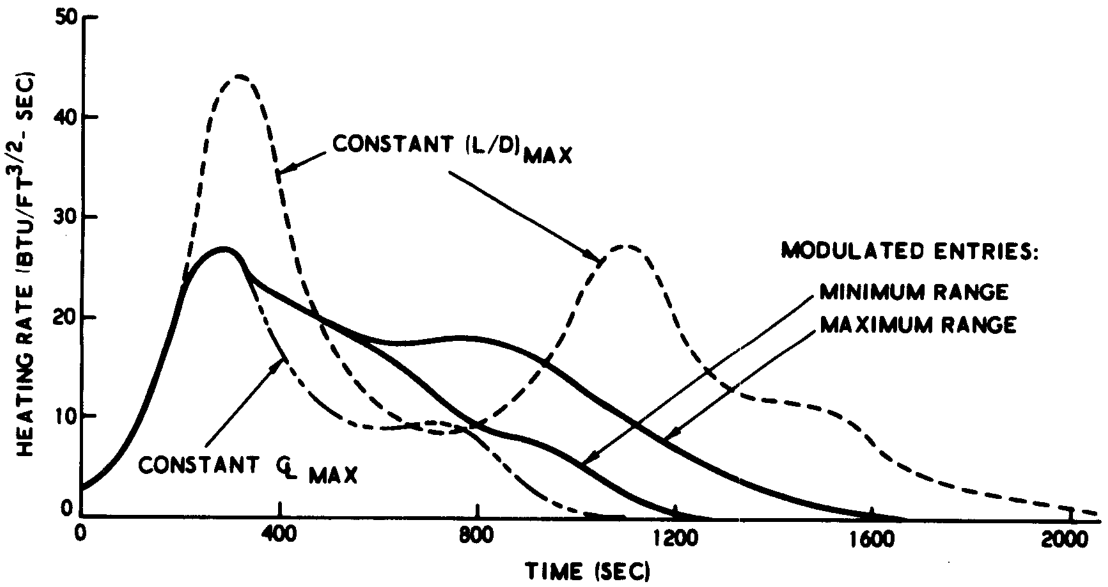


FIGURE 17

MEM ABORT ALTITUDE LIMITS

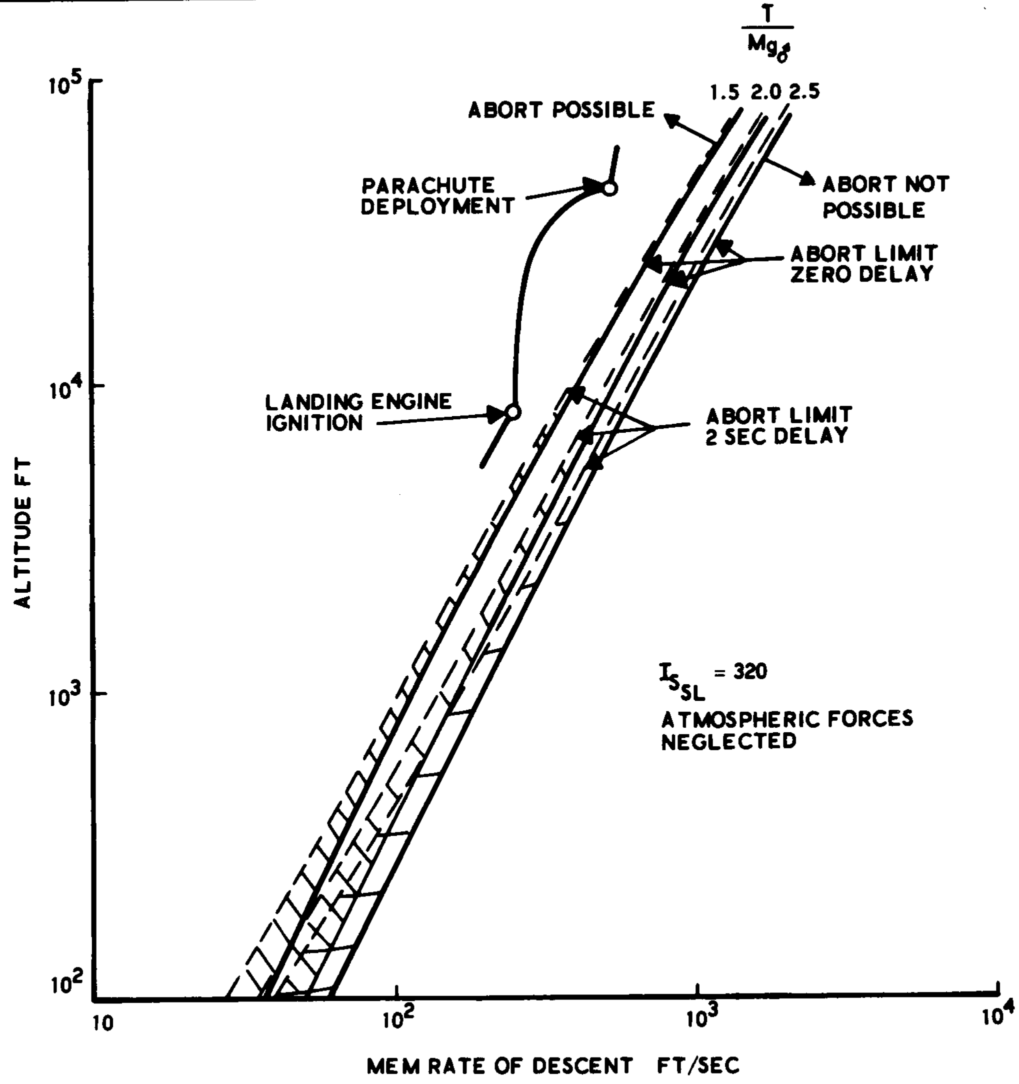


FIGURE 18

PRIORITY OF EXPERIMENTAL DATA COLLECTION

PRIORITY	DATA
1	EVALUATION OF LOCAL HAZARDS PRIOR TO EGRESS
	SEARCH FOR UNFRIENDLY LIFE FORMS
	SOLAR AND PLANETARY RADIATION
	SEVERE TOPOLOGICAL FEATURES
	ATMOSPHERIC CONSTITUENTS
	DIRECTION AND VELOCITY OF SURFACE WINDS
	SURFACE TEMPERATURE
2	DIURNAL VARIATION IN SURFACE PRESSURE, DENSITY, TEMPERATURE, HUMIDITY, AND WIND SPEED AND DIRECTION
3	BIOLOGICAL EVALUATION OF LIFE FORMS
4	PHOTOGRAPHIC MAPPING AND RECONNAISANCE
5	SURFACE COMPOSITION
6	SUBSURFACE STRUCTURE
7	TRAFFICABILITY
8	SURFACE BEARING AND SHEAR STRENGTH
9	DIRECTION AND STRENGTH OF MAGNETIC FIELDS
10	MAGNITUDE AND VARIATION IN LOCAL GRAVITY
11	SEISMIC ACTIVITY
12	THERMAL GRADIENT AND THERMAL CONDUCTIVITY
13	VARIATION WITH ALTITUDE OF ATMOSPHERIC PROPERTIES
14	ABUNDANCE AND WEIGHT DISTRIBUTION OF PRINCIPAL ATMOSPHERIC CONSTITUENTS
15	PARTICLE SIZE IN DUST CLOUDS
16	VISIBILITY TRANSMISSIVITY
17	SAMPLING OF DEIMOS AND PHOBOS

FIGURE 19

BIOLOGY STUDY PROGRAM

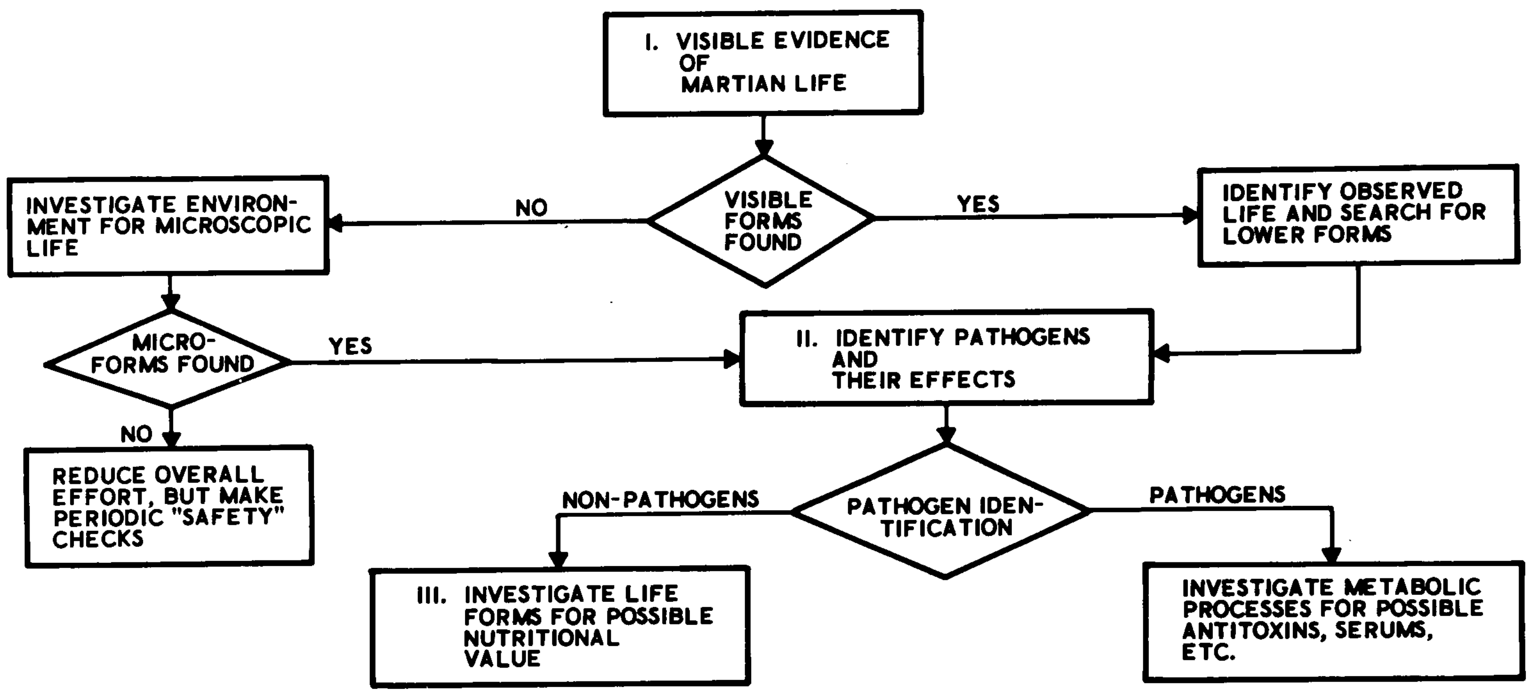
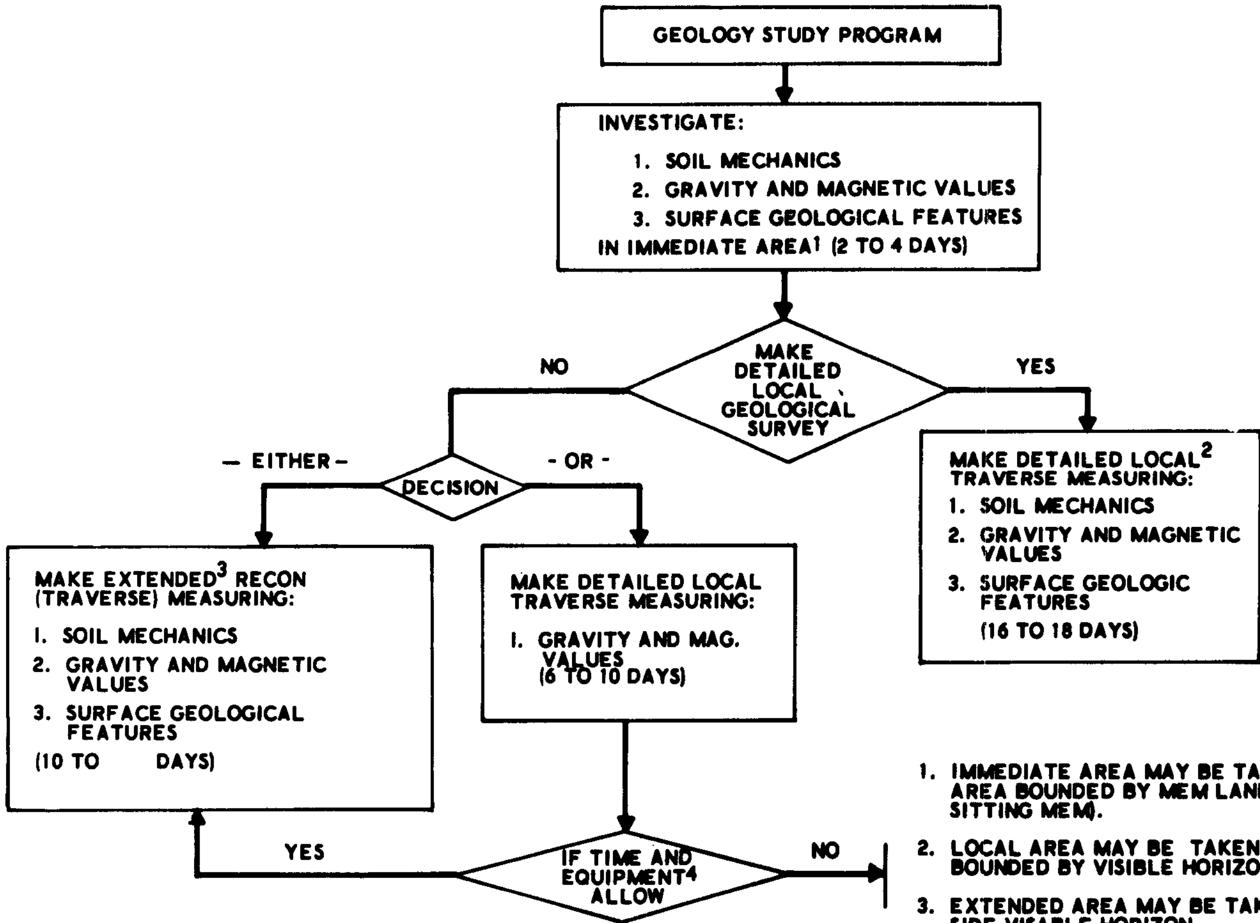


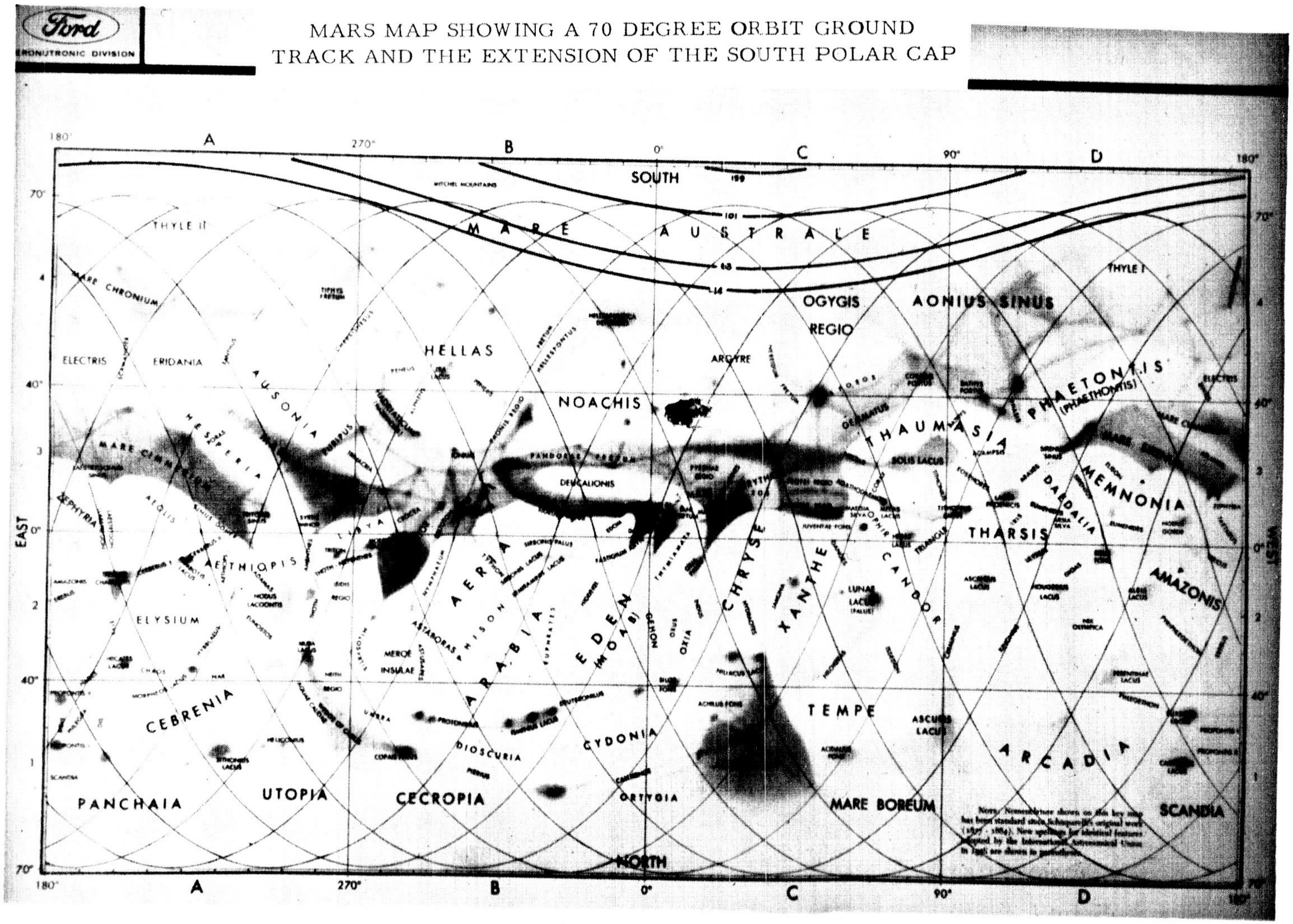
FIGURE 20



1. IMMEDIATE AREA MAY BE TAKEN TO MEAN APPROX. AREA BOUNDED BY MEM LANDING GEAR (A TAIL-SITTING MEM).
2. LOCAL AREA MAY BE TAKEN TO MEAN APPROX. AREA BOUNDED BY VISIBLE HORIZON.
3. EXTENDED AREA MAY BE TAKEN TO MEAN AREA OUT-SIDE VISABLE HORIZON.
4. EXTENDED TRAVERSING REQUIRES A SURFACE VEHICLE.

FIGURE 21

MARS MAP SHOWING A 70 DEGREE ORBIT GROUND TRACK AND THE EXTENSION OF THE SOUTH POLAR CAP



AERONUTRONIC DIVISION

FIGURE 22

CAPTAIN-ASTRONAUT-SCIENTIFIC AIDE

FLIGHT DUTIES

- COMPLETE AUTHORITY DURING THE FLIGHT PHASES
- DETERMINATION OF SURFACE STAY TIME
- DECISION TO ABORT
- CHOICE OF LANDING SITE
- SAFETY AND WELFARE OF CREW
- ALLOCATION OF DUTIES, OUTSIDE THOSE CONNECTED WITH STUDY PROGRAMS
- FLIGHT CONTROL OF MEM

SURFACE DUTIES

- SCIENTIFIC AIDE AND ASSISTANT
- SHARE IN HOUSEKEEPING
- SYSTEM MONITORING AND MAINTENANCE

FIGURE 23

AERONUTRONIC DIVISION

FIRST OFFICER-SCIENTIST-ASTRONAUT

FLIGHT DUTIES

- SUPPORT CAPTAIN IN FLIGHT TASKS
- NAVIGATION
- WEATHER
- ADVISE CAPTAIN AS TO LANDING SITE SUITABILITY

SURFACE DUTIES

- SUPERVISE GEOLOGY/METEOROLOGY EXPERIMENTATION
- AID BIOLOGIST WHERE NEEDED
- SHARE IN HOUSEKEEPING

Ford
AERONUTRONIC DIVISION

FIGURE 24

SECOND OFFICER-SCIENTIST ASTRONAUT

FLIGHT OPERATIONS

- MONITOR MEM SYSTEMS
- ESTABLISH/MAINTAIN LIFE SUPPORT ENVIRONMENT
- MONITOR PHYSIOLOGICAL STRESSES

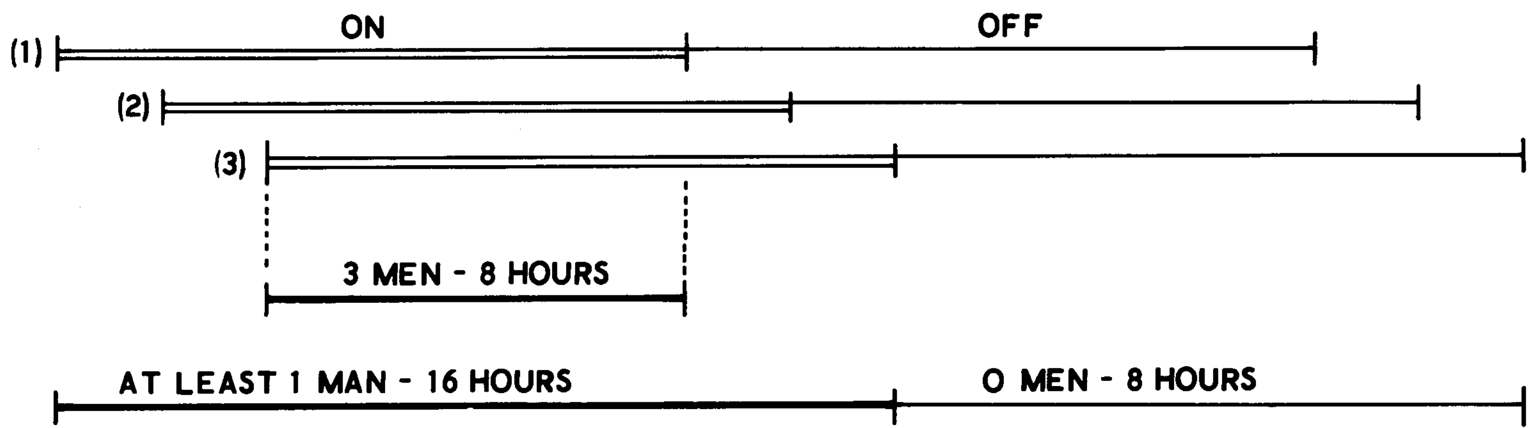
SURFACE OPERATIONS

- CONDUCT BIOLOGICAL STUDY PROGRAM
- GUARD CREW'S HEALTH
- ADVISE CAPTAIN CONCERNING CREW'S HEALTH
- AID GEOLOGIST WHERE REQUIRED
- SHARE IN HOUSEKEEPING

Ford AERONUTRONIC DIVISION

FIGURE 25

WORK - REST CYCLE



(1) CAPTAIN

(2) FIRST OFFICER

(3) SECOND OFFICER

DURING EVERY 24 HOURS:

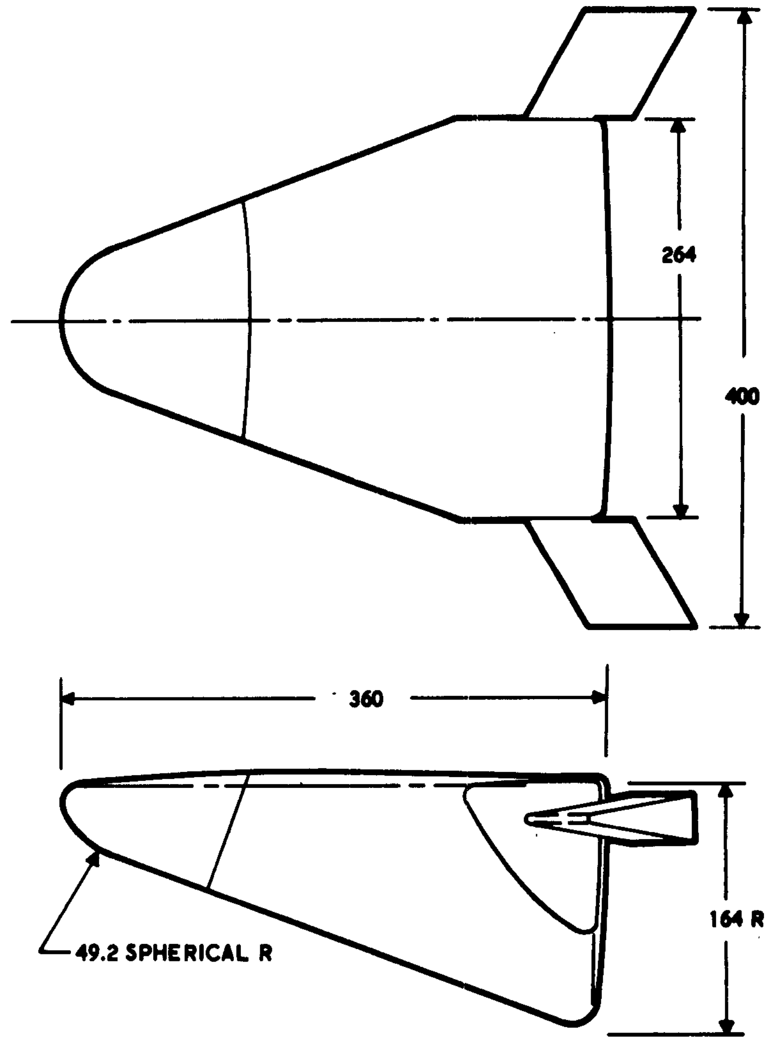
16 MAN HOURS OUTSIDE

20 MAN HOURS INSIDE

36 MAN HOURS ON DUTY

FIGURE 26

GENERAL ARRANGEMENT - TAIL SITTER VEHICLE



AERONUTRONIC DIVISION

FIGURE 27

INBOARD PROFILE MEM CONFIGURATION D9-1

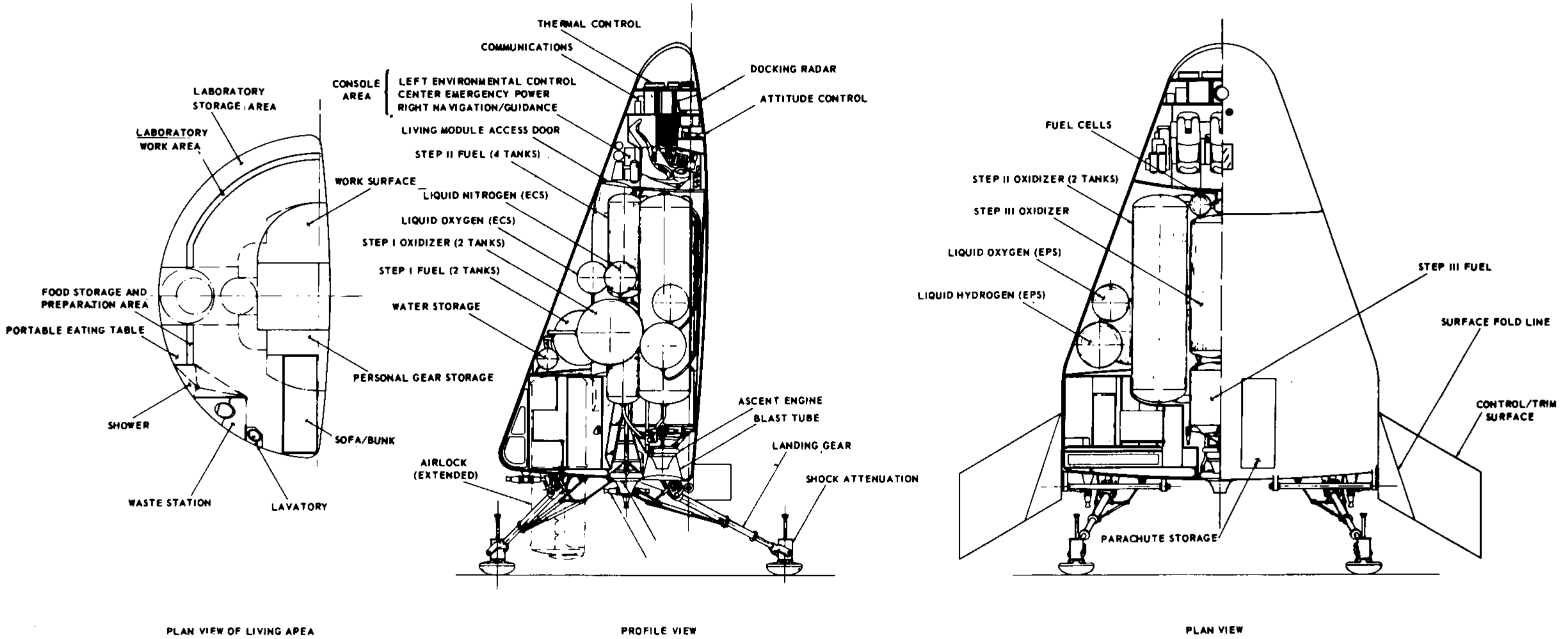


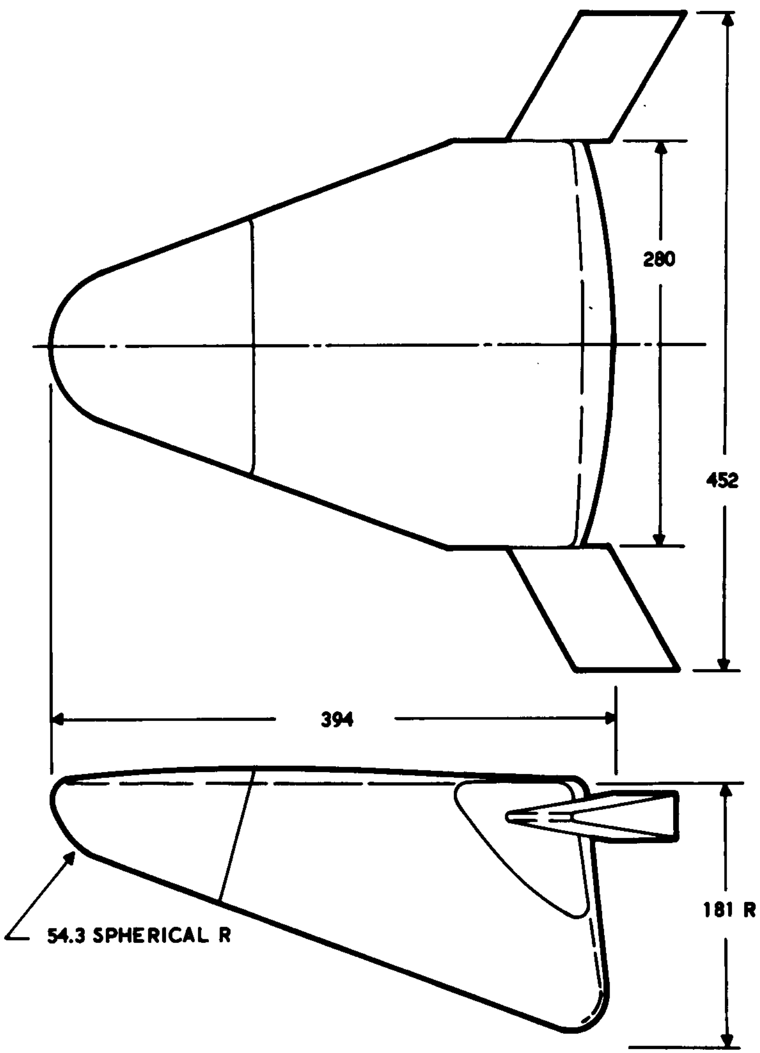
FIGURE 28

MEM TAIL SITTER WEIGHT SUMMARY

ITEM	NOMINAL VEHICLE		MINIMUM VEHICLE	
	LANDING	ASCENT	LANDING	ASCENT
STRUCTURE	9,006	2,230	8,287	2,028
CREW SYSTEMS	663	943	610	913
THERMAL CONTROL	35	157	33	143
AIR CONDITIONING	1,142	159	1,102	153
INSTRUMENTS/DISPLAYS	163	100	151	93
ELECTRICAL POWER	2,159	964	2,017	897
COMMUNICATIONS	298	194	275	180
GUIDANCE/NAVIGATION-STABILITY/CONTROL	–	346	–	330
REACTION CONTROL	441	634	420	610
FIN CONTROL SYSTEM	200	–	190	–
PROPULSION SYSTEM	1,652	3,087	1,421	2,426
SCIENTIFIC PAYLOAD	1,214	800	1,130	750
USEABLE PROPELLANT	11,831	28,646	10,104	23,807
SUBTOTAL	28,804	38,160	25,740	32,330
GROSS WEIGHT		66,964		58,070

FIGURE 29

GENERAL ARRANGEMENT - CANTED LANDING VEHICLE



Ford
AERONUTRONIC DIVISION

FIGURE 30

INBOARD PROFILE MEM CONFIGURATION D8-I

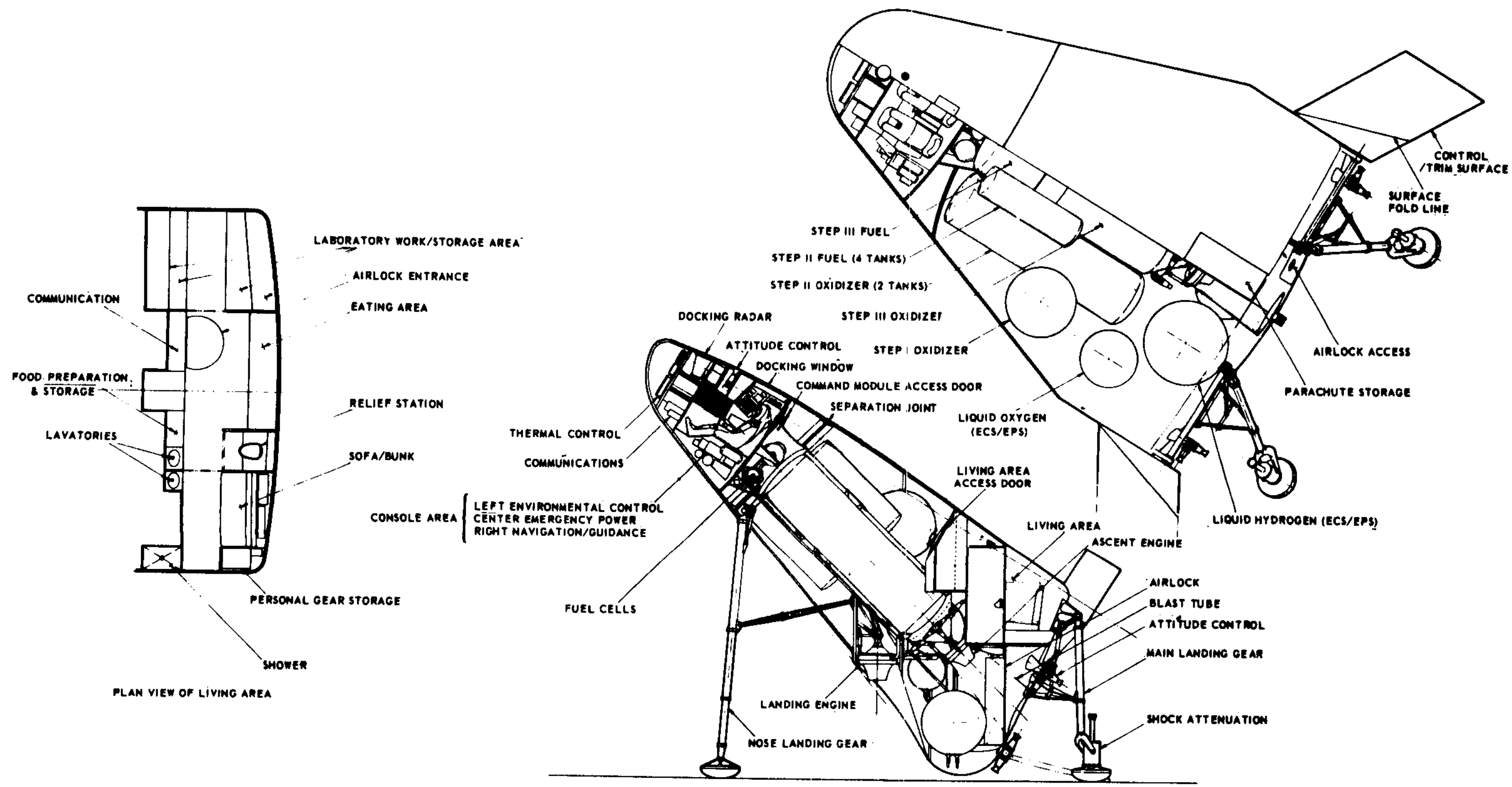


FIGURE 31

SUMMARY WEIGHT ENVELOPE

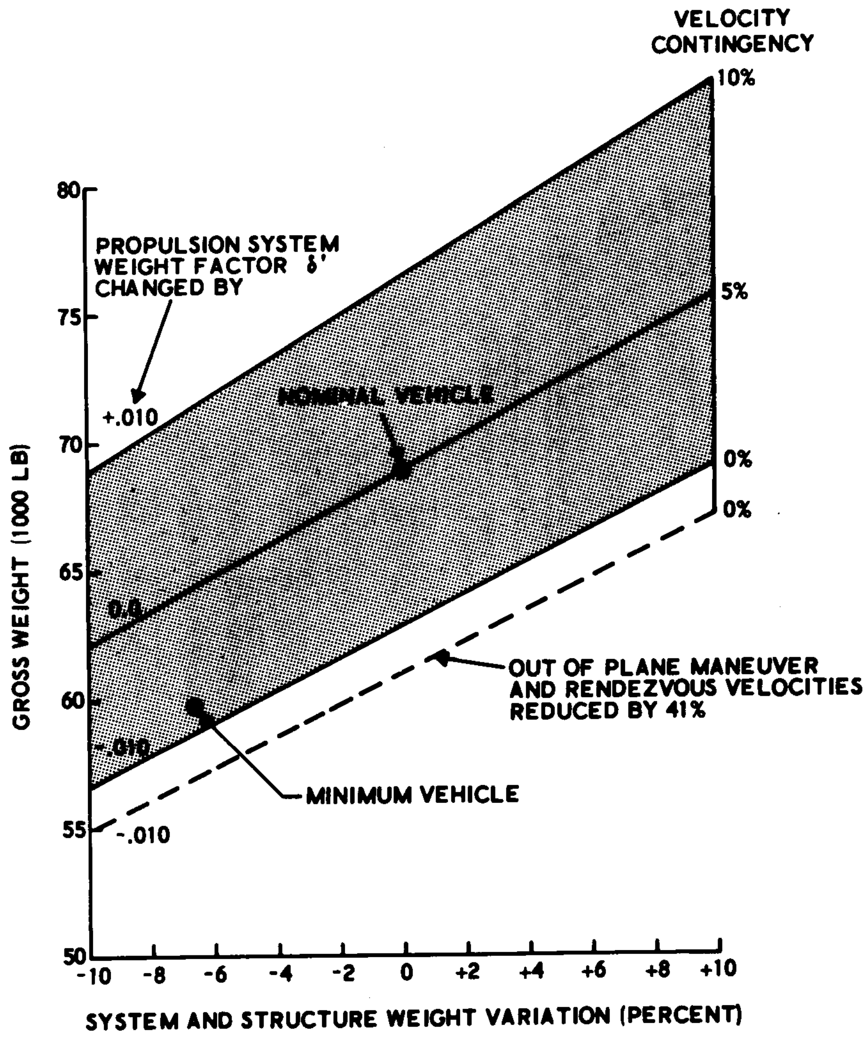
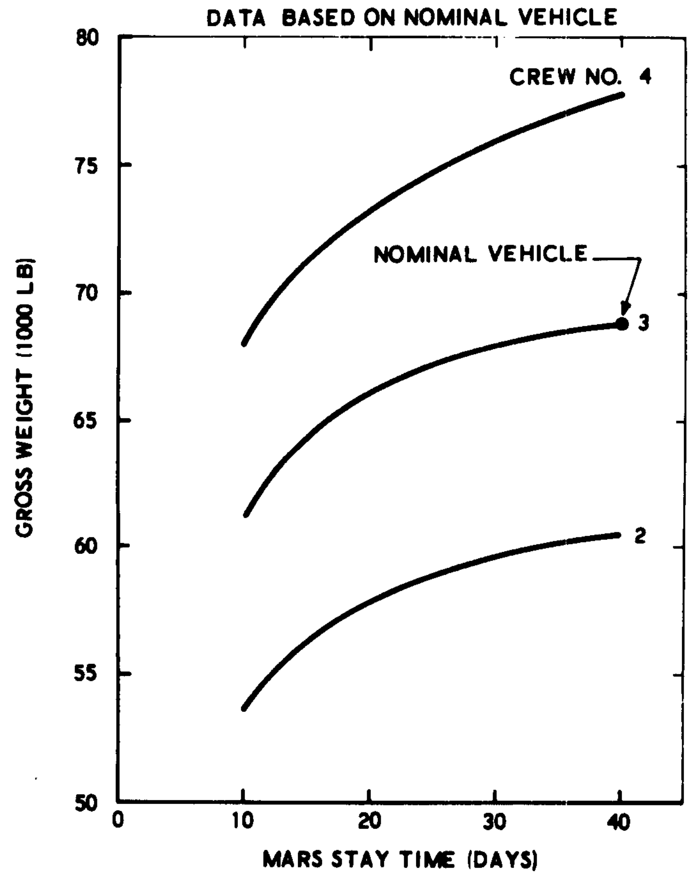


FIGURE 32

GROSS WEIGHT VERSUS MARS STAY TIME AND CREW NUMBER



AERONUTRONIC DIVISION

FIGURE 33

GROSS WEIGHT VERSUS SPECIFIC IMPULSE

DATA BASED ON NOMINAL VEHICLE

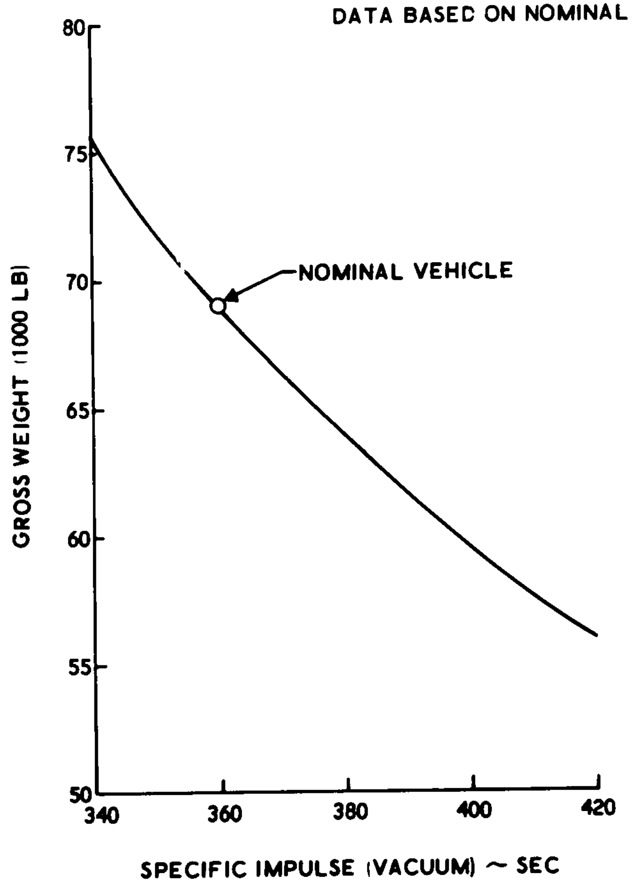


FIGURE 34

GROSS WEIGHT VERSUS NUMBER OF ASCENT STAGES

- DATA BASED ON NOMINAL VEHICLE
- ONLY PROPELLANT TANKAGE IS STAGED

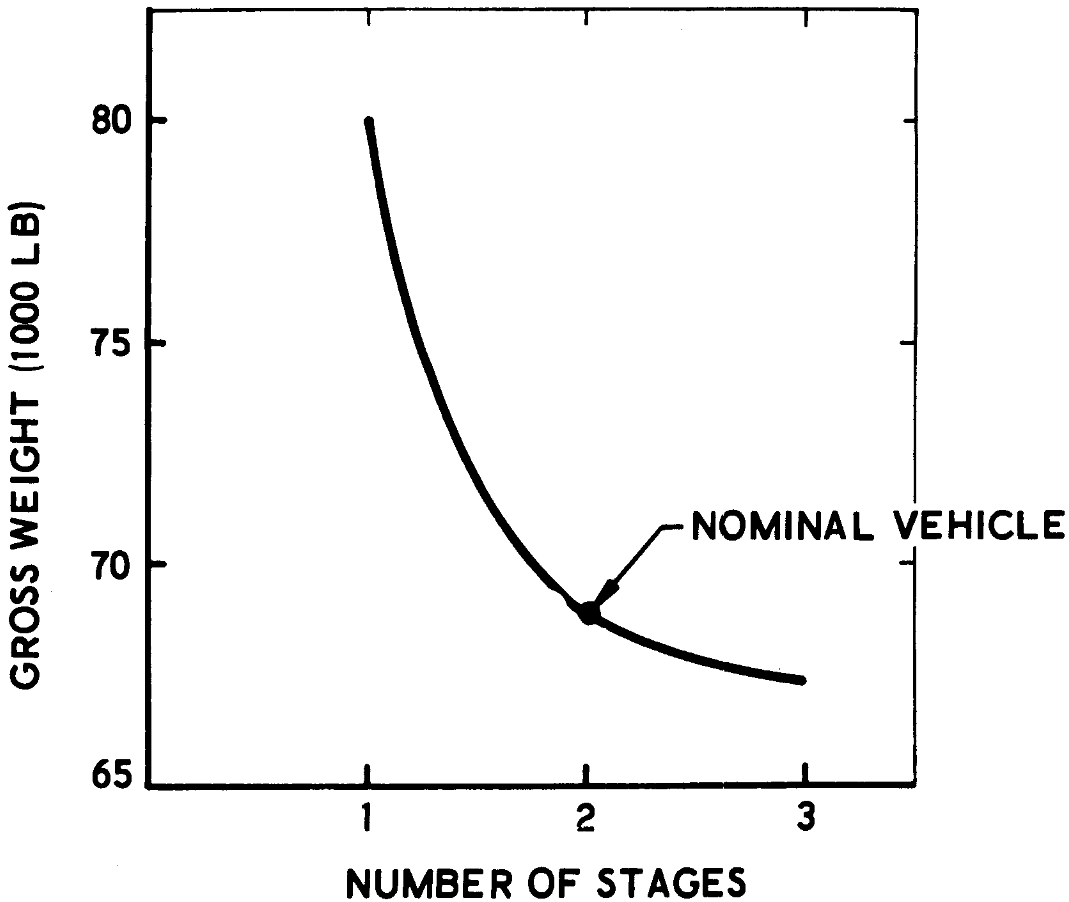


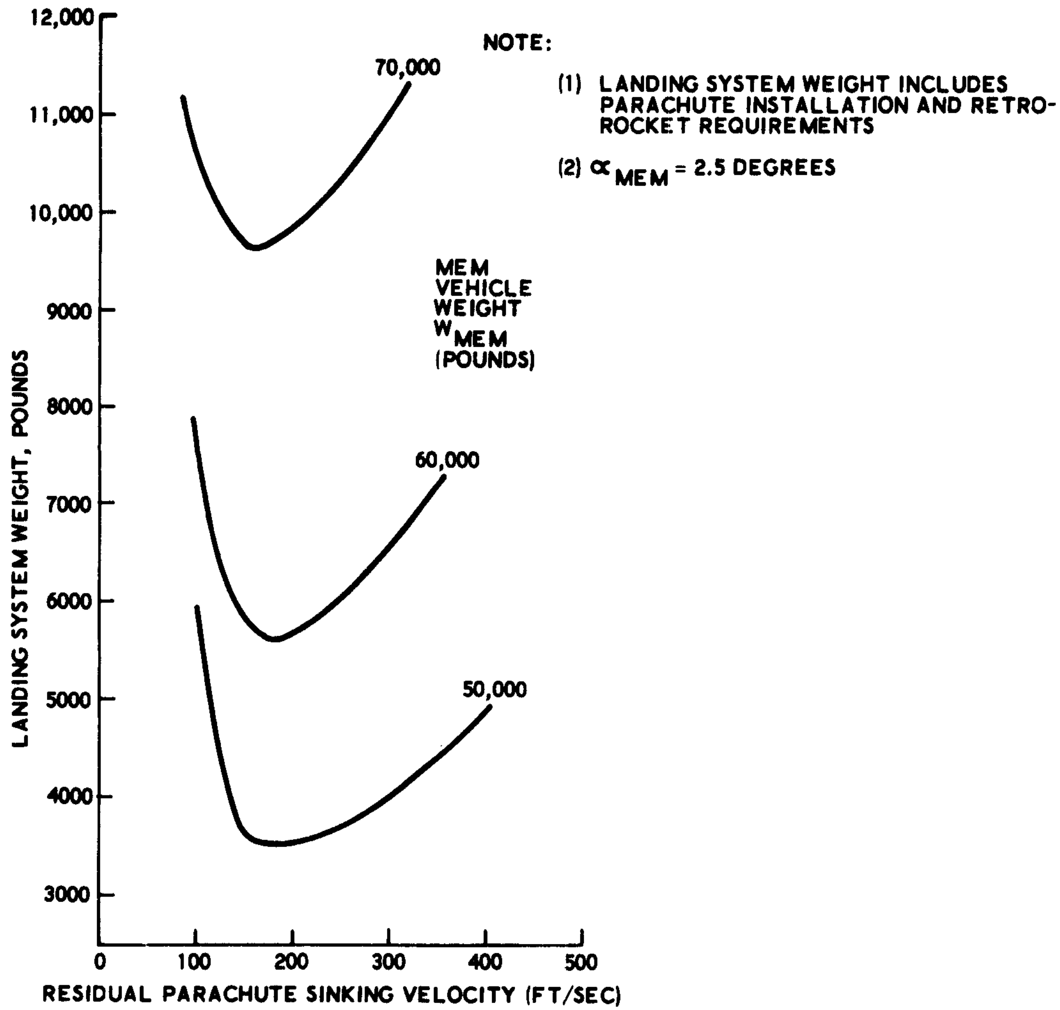
FIGURE 35

MEM BASIC AERODYNAMIC DATA TABLE

PARAMETER	NEWTONIAN THEORY	TEST DATA
$(L/D)_{MAX}$ (TRIMMED)	1.32	1.00
C_L AT $(L/D)_{MAX}$ (TRIMMED)	0.198	0.365
C_D AT $(L/D)_{MAX}$ (TRIMMED)	0.150	0.362
α AT $(L/D)_{MAX}$ (TRIMMED)	-2.5°	+2.0°
$C_{L_{MAX}}$ (TRIMMED)	0.672	0.649
C_D AT $C_{L_{MAX}}$ (TRIMMED)	1.295	1.249
α AT $C_{L_{MAX}}$ (TRIMMED)	23°	25°

FIGURE 36

MEM DECELERATION AND RETRO SYSTEM OPTIMIZATION



AERONUTRONIC DIVISION

FIGURE 37

RADIATIVE HEATING TESTS IN CO_2-N_2 ATMOSPHERES AT AMES

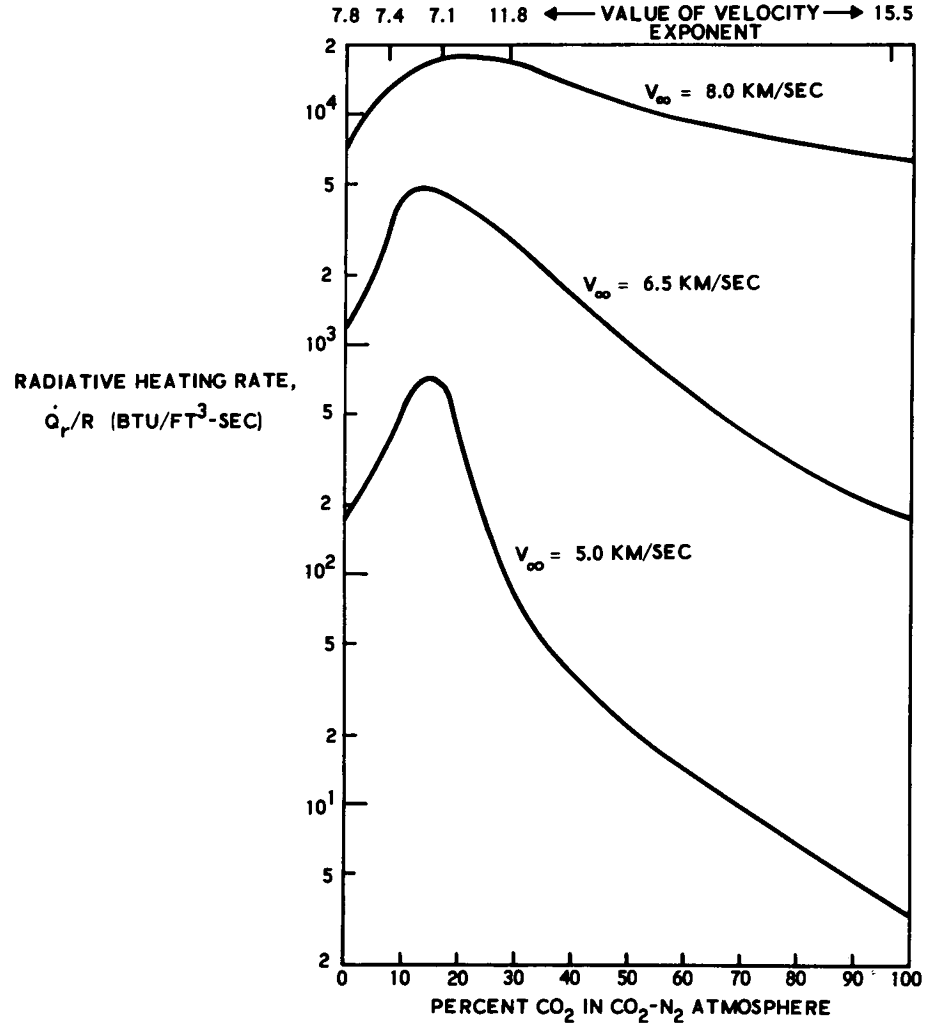


FIGURE 38

HEAT TRANSFER DISTRIBUTION FROM STAGNATION POINT

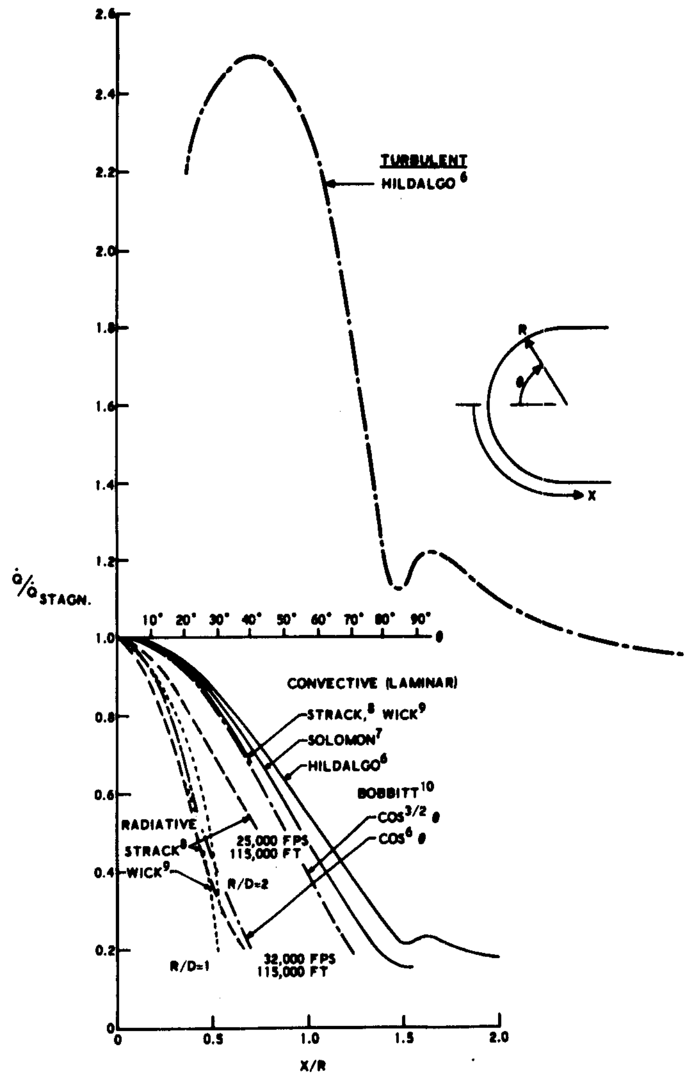


FIGURE 39

PEAK TEMPERATURE DISTRIBUTION ON VEHICLE AT $\alpha = 30°$

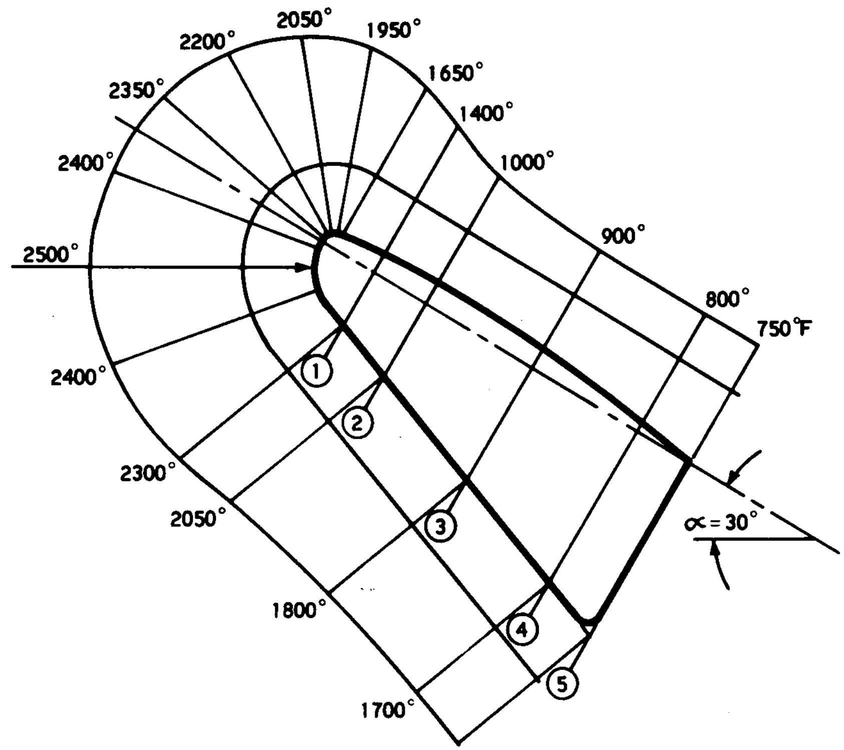
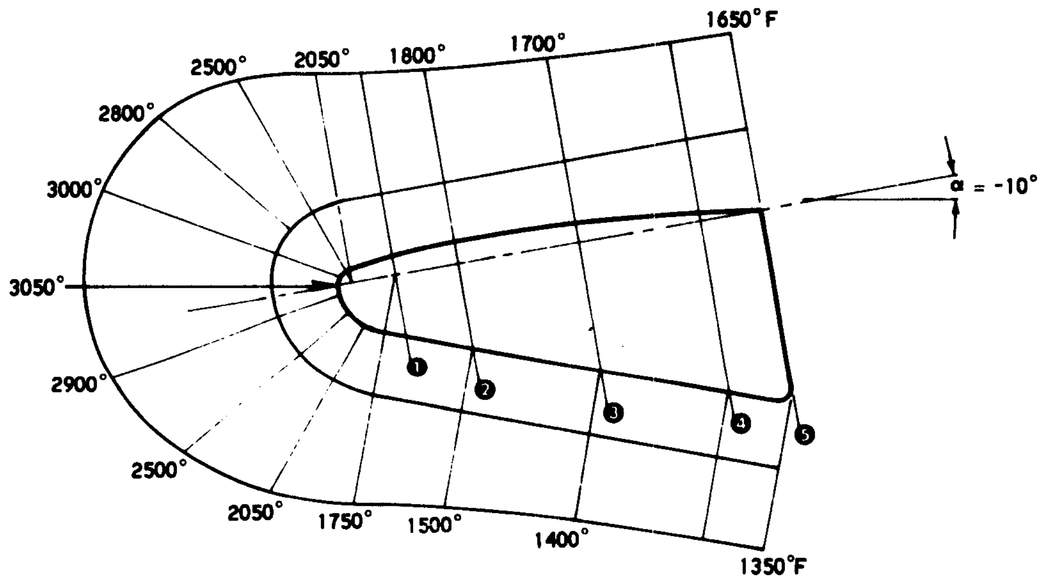


FIGURE 40

PEAK TEMPERATURE DISTRIBUTION ON VEHICLE AT $\alpha = -10°$



AERONUTRONIC DIVISION

FIGURE 41

ESTIMATED ENTRY HEATING (MAXIMUM RANGE)

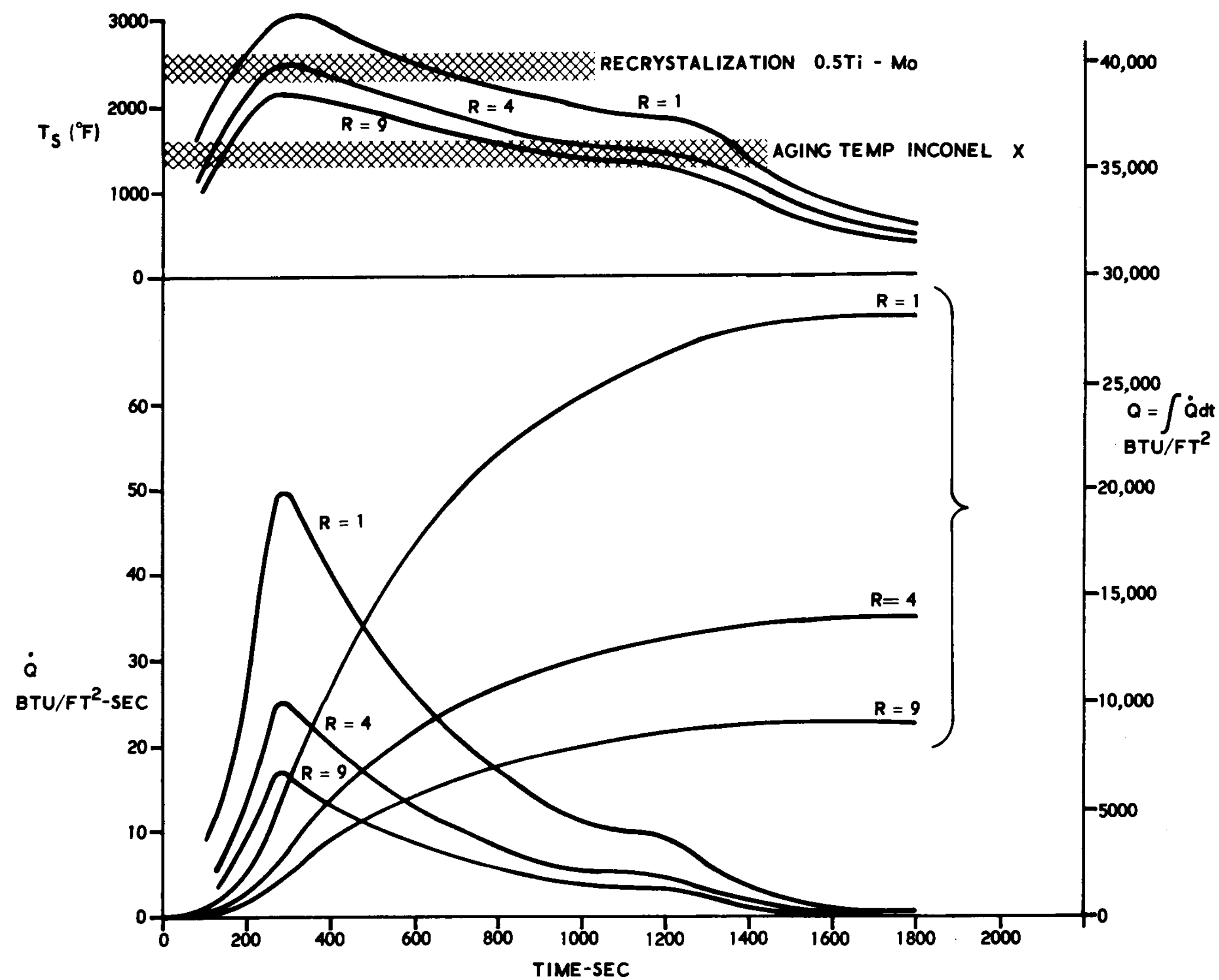
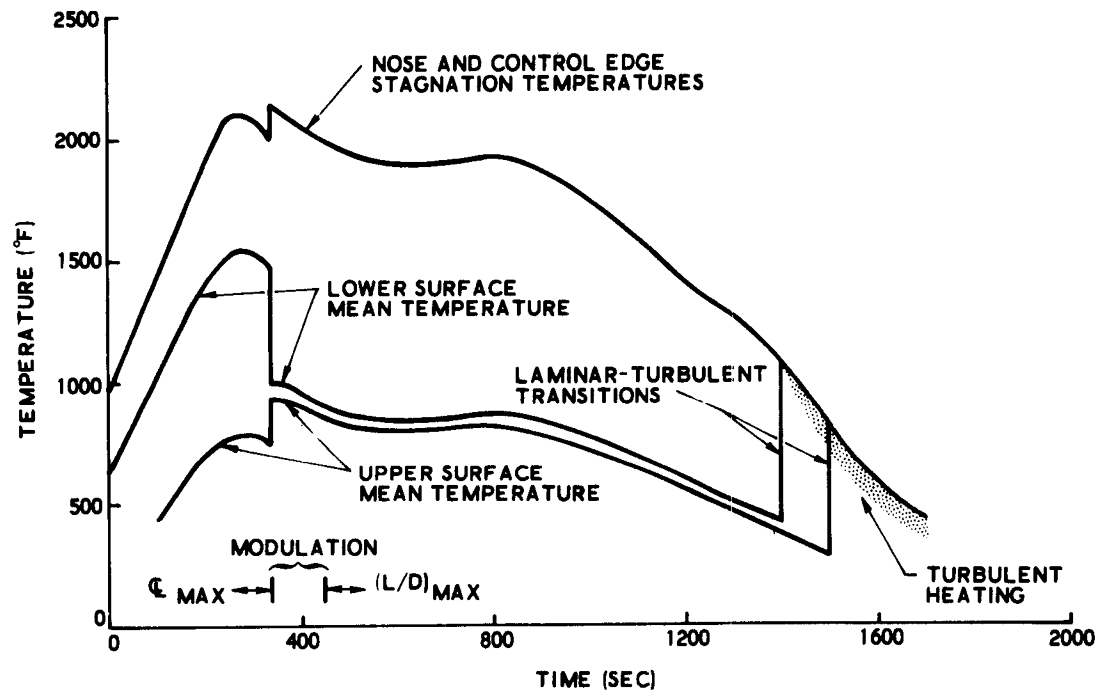


FIGURE 42

TEMPERATURE TIME HISTORIES FOR MODULATED ENTRY

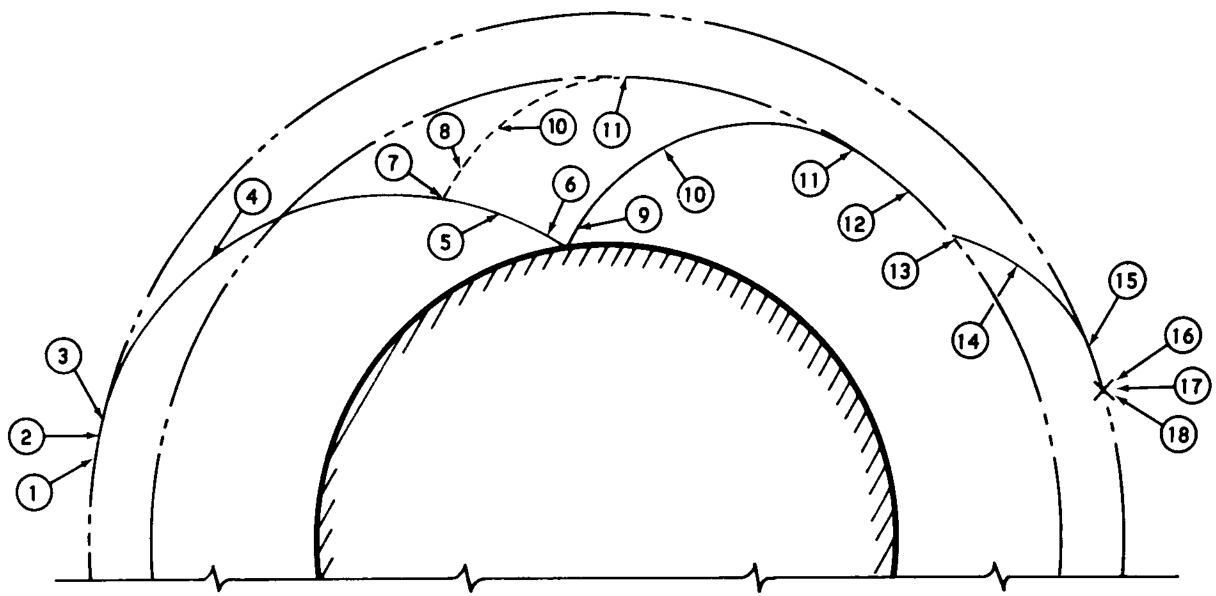


Ford AERONUTRONIC DIVISION

FIGURE 43

MEM PROPULSION SEQUENCES

1. MEM-MMM SEPARATION
2. ORIENTATION
3. DE-ORBIT
4. ORIENTATION
5. ENTRY ATTITUDE CONTROL
6. RETRO AND HOVER
7. ABORT SEPARATION
8. ABORT ASCENT
9. NORMAL ASCENT
10. ORIENTATION
11. INTERMEDIATE ORBIT INJECTION
12. ORIENTATION
13. TRANSFER ORBIT INJECTION
14. ORIENTATION
15. RENDEZVOUS ORBIT INJECTION
16. RENDEZVOUS
17. DOCKING
18. MEM-MMM SEPARATION



AERONUTRONIC DIVISION

FIGURE 44

MEM MAIN PROPULSION INCREMENTS

ΔV INCREMENT	NOMINAL ΔV (FT/SEC)	ENVIRONMENTAL AND FUNCTIONAL CONSTRAINTS
1. DEORBIT	550	VACUUM CONDITIONS, ZERO GRAVITY START
2. RETRO	870	AMBIENT PRESSURE 1.25 PSIA
HOVER	740	ACCELERATION LEVEL = 1.5 g♂
CONTINGENCY	110	THRUST: THROTTLEABLE 1 g♂ > T > 1.5 g♂
3. ABORT ASCENT	-	NORMAL ASCENT CAPABILITY UTILIZED
4. NORMAL ASCENT	15,350	AMBIENT PRESSURE=1.25 PSIA TO VACUUM CONDITIONS
PLANE CHANGE	440	ACCELERATION LEVEL=ZERO TO 2.5 g♂
CONTINGENCY	810	THRUST: NO THROTTLING, MILTIPLE START
5. INITIAL ORBIT INJECTION	500	AMBIENT PRESSURE=VACUUM CONDITIONS
6. TRANSFER ORBIT INJECTION		ACCELERATION LEVEL=ZERO GRAVITY START
7. FINAL ORBIT INJECTION	———	
8. RENDEZVOUS		
TOTAL CHARACTERISTIC VELOCITY CHANGE	19,370	

FIGURE 45

DELIVERED SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE CONSIDERING THE POINT OF SEPARATION AS LIMITING THE EXPANSION AREA RATIO OF THE BELL NOZZLE

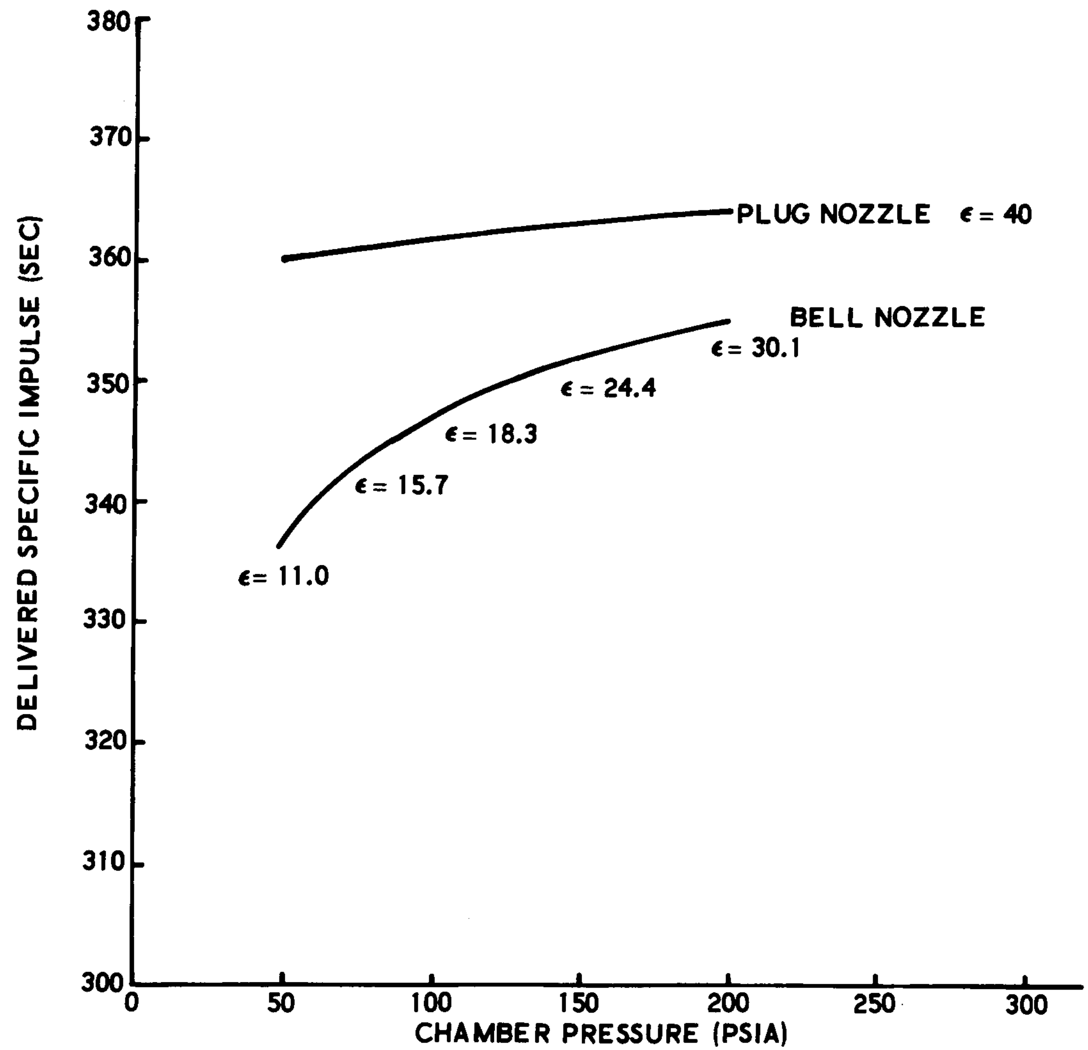


FIGURE 46

PLACEMENT OF MEM DESCENT VEHICLE RCS ENGINES

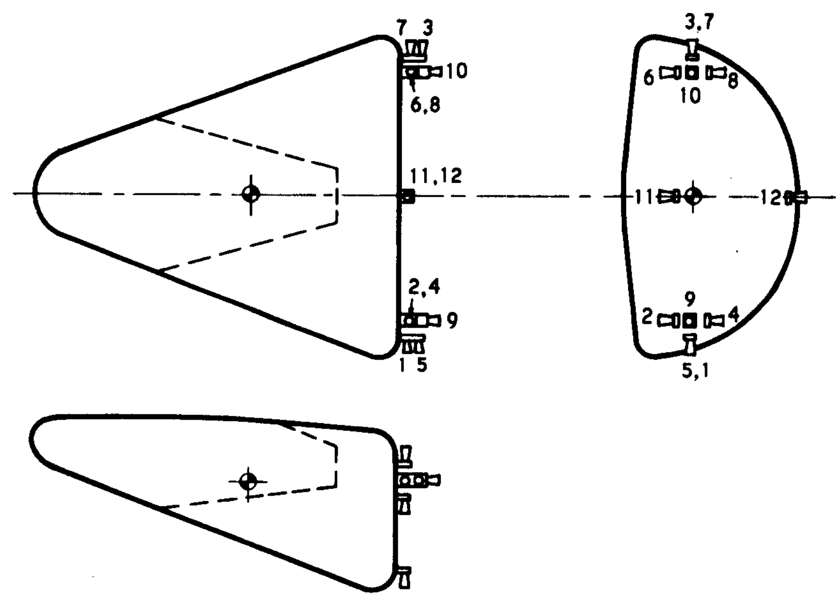


FIGURE 47

PLACEMENT OF MEM ASCENT VEHICLE RCS ENGINES

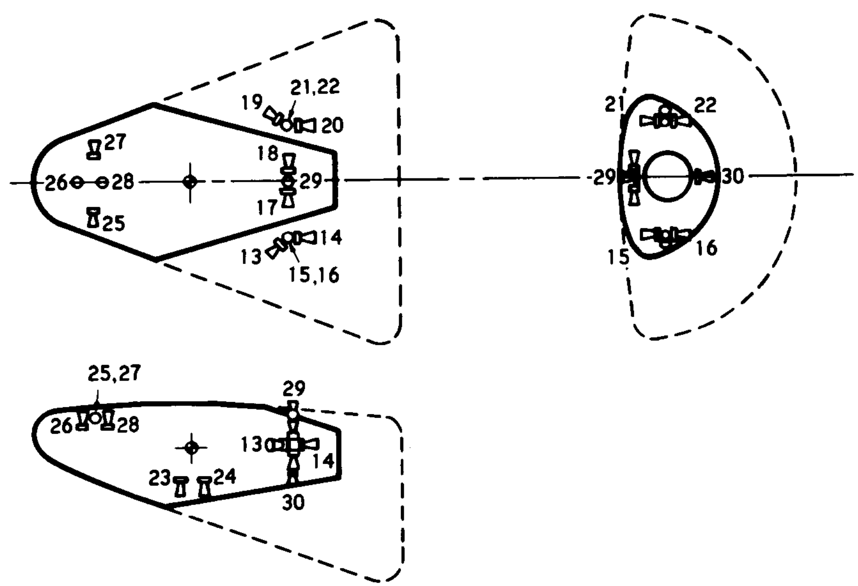


FIGURE 48

ELECTRICAL POWER DUTY CYCLE

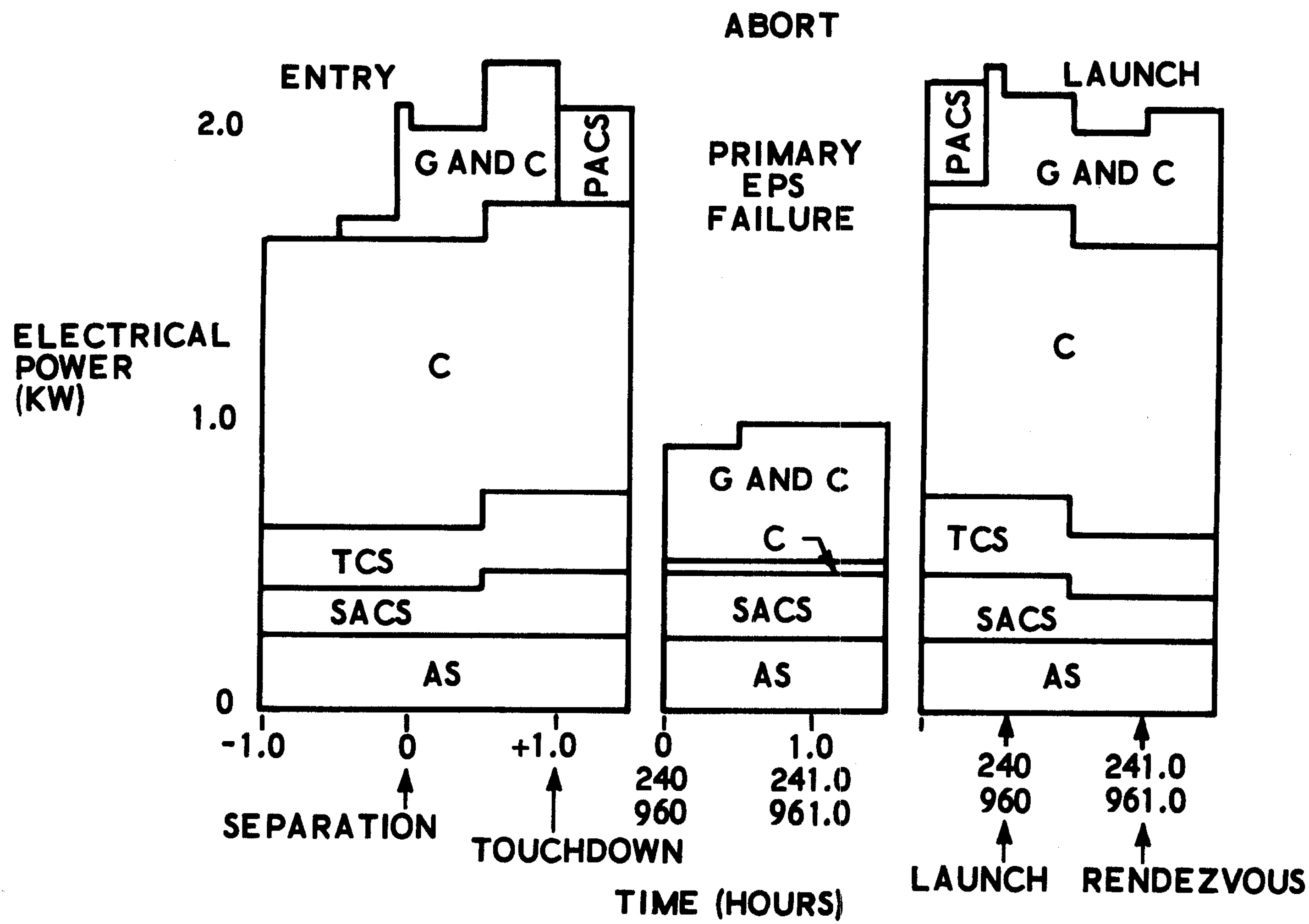


FIGURE 49

REQUIRED MEM SUBSYSTEMS

- ATMOSPHERIC CONTROL
  - O_2 AND N_2 SUPPLY
  - CO_2 CONTROL
  - HUMIDITY CONTROL
  - TRACE CONTAMINANT CONTROL
- THERMAL CONTROL
- PRESSURE SUITS
- BACKPACK ENVIRONMENTAL CONTROL SYSTEM
- WATER MANAGEMENT
- WASTE MANAGEMENT AND HYGIENE
- LOCK SYSTEM
- CREW SYSTEMS
  - TIMING SYSTEM
  - ILLUMINATION
  - DATA COMPUTER
  - FIRE DETECTION AND CONTROL
  - REPAIR EQUIPMENT
  - FIRST AID EQUIPMENT
  - FOOD
- BACKPACK REGENERATION EQUIPMENT
- BATTERY CHARGER

AERONUTRONIC DIVISION

FIGURE 50

PRIMARY ATMOSPHERIC CONTROL SYSTEM

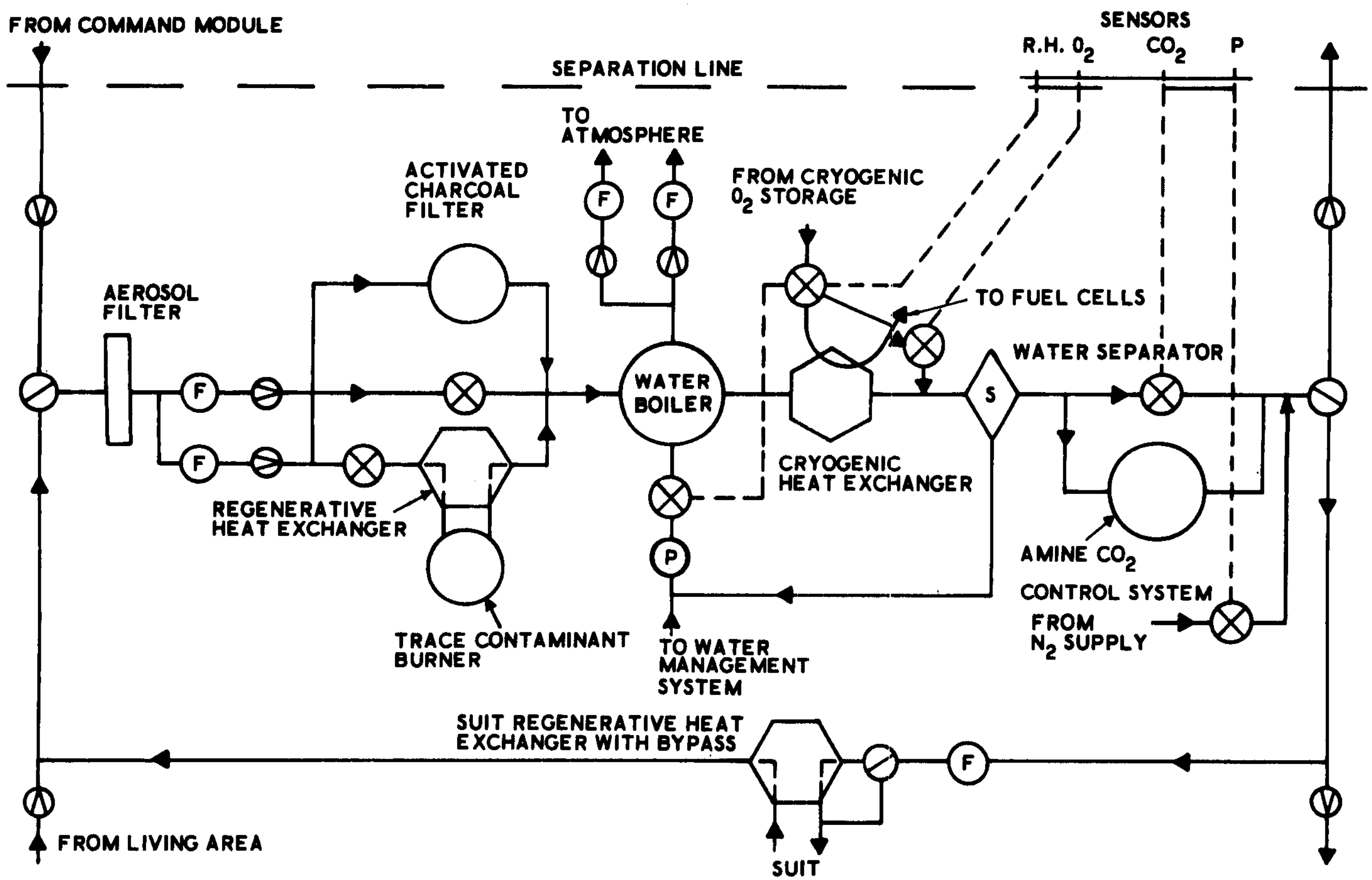


FIGURE 51

LIFE SUPPORT AND THERMAL CONTROL SYSTEMS

USEABLE LOAD TRADE-OFF

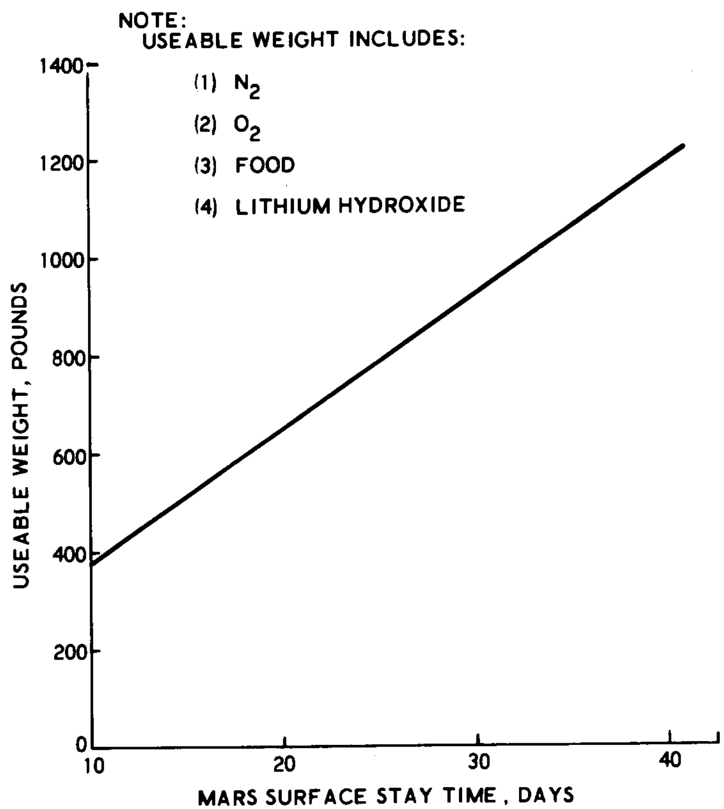


FIGURE 52

TOTAL POSITION ERRORS

AT INITIATION OF LANDING MANEUVER

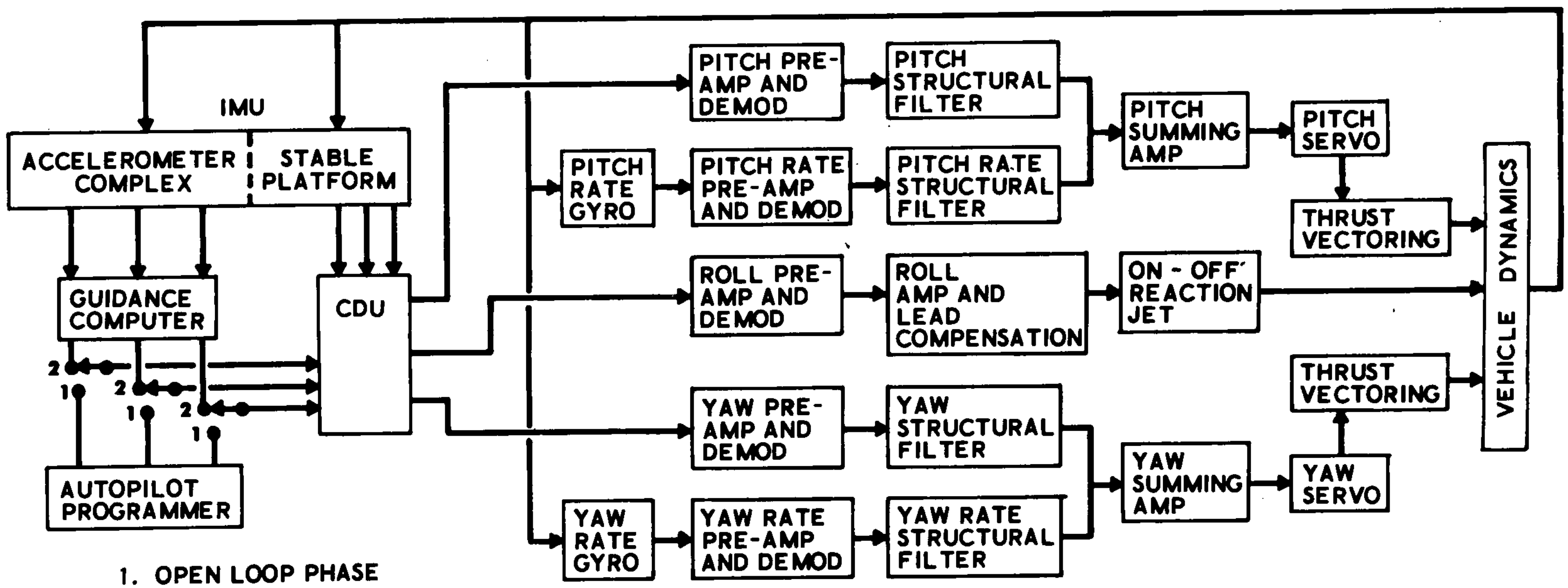
	1σ (KM)
DOWNRANGE	70.00
NORMAL	15.80

INITIATION OF RENDEZVOUS MANEUVER

	1σ (KM)
DOWNRANGE	20.40
RADIAL	5.80
NORMAL	5.50

FIGURE 53

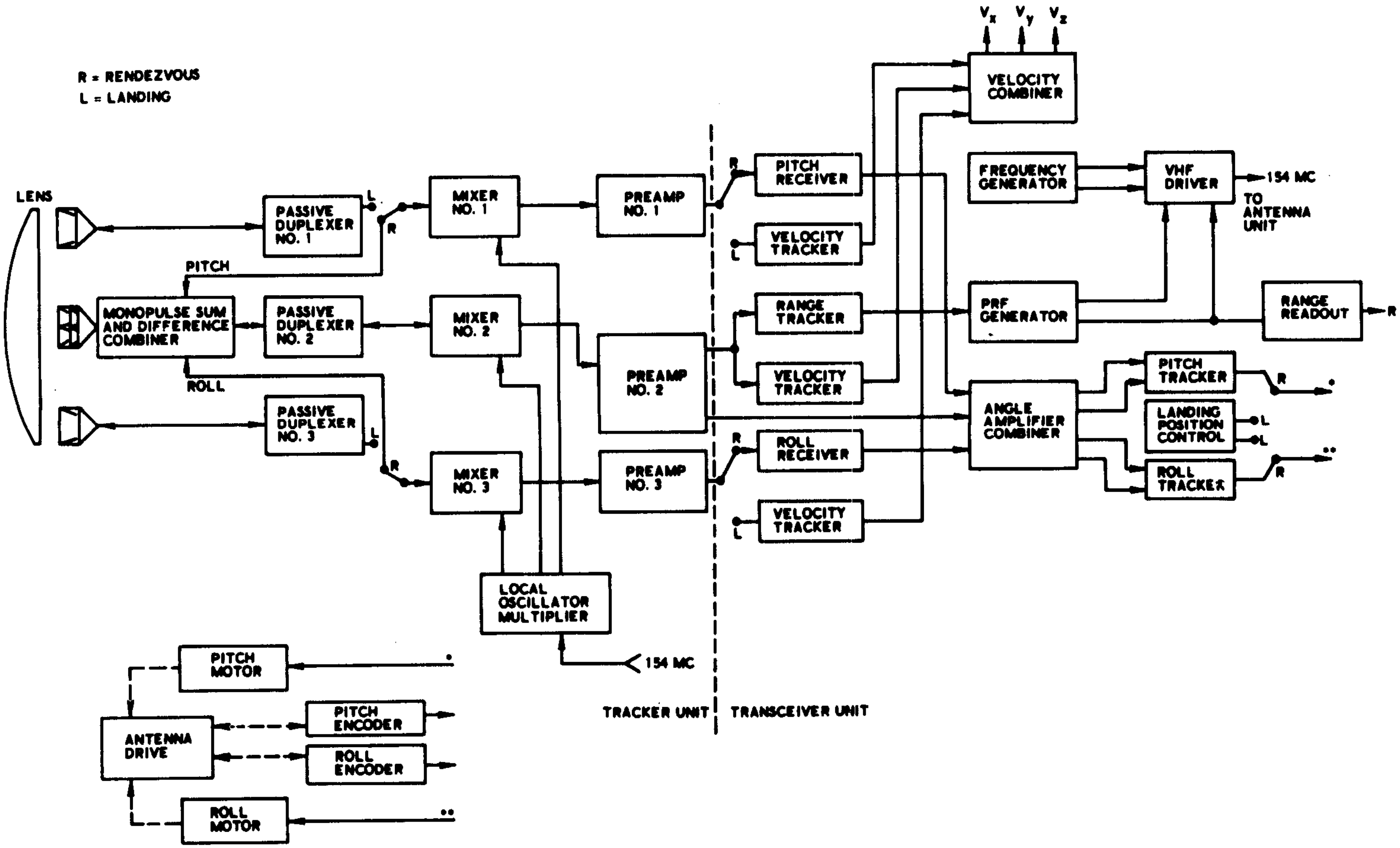
GUIDANCE AND CONTROL FUNCTIONAL SCHEMATIC
MEM THRUSTING PHASE



1. OPEN LOOP PHASE
2. GUIDANCE PHASE

FIGURE 54

SCHEMATIC, MEM RADAR SYSTEM



AERONUTRONIC DIVISION

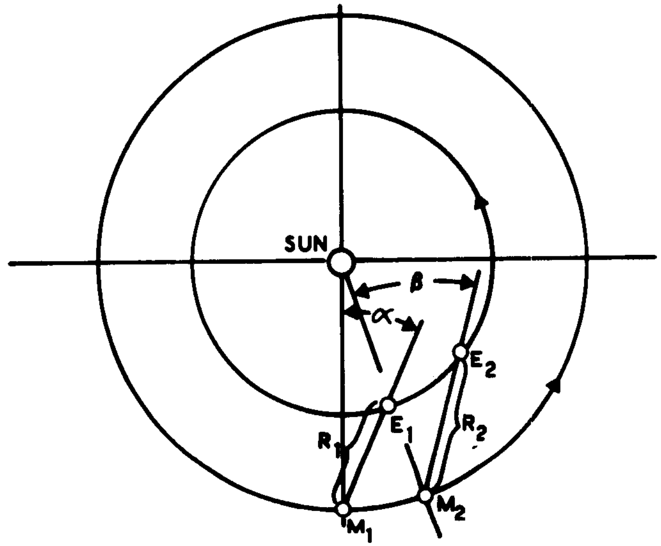
FIGURE 55

AVERAGE DAILY DATA LOAD

APPROX AMOUNT OF DATA GATHERED PER DAY	BANDWIDTH-TIME PRODUCT ANALOG (C/S^2)	DIGITAL TECHNIQUES BY 1970	TOTAL BITS PER DAY
50 SHEETS OF DATA TO BE READ BY PAGE READER	2.0×10^6	IF STRICTLY WRITTEN INFORMATION (NO SKETCHES OR DRAWINGS), CHARACTER READER CAN BE USED	1.5×10^6
50 SLIDES, SAMPLES, OR PHOTOGRAPHIC FACSIMILE	10.0×10^6	SAMPLE AT 4 BITS PER CYCLE, USING DELTA MODULATION OR EQUIVALENT	40.0×10^6
120 MINUTES OF VOICE MESSAGES	22.0×10^6	REDUCTION BY 5 BECAUSE OF 600 BITS/SEC VOCODER	4.6×10^6
STATUS REPORTING OR PRE-RECORDED VOICE	0.2×10^6	INSIGNIFICANT AMOUNT IF DIGITIZED	
SERVICE SIGNALS, NON-UTILIZED INFORMATION SPACE, OTHER CONTINGENCIES	1.0×10^6	CONVERT 2:1	0.5×10^6
	DIGITAL (BITS)		
6 WEATHER STATIONS x 24 INTERROGATIONS AT 100 BITS EACH	1,500	ROUNDED-OFF TO	0.2×10^6
RADIO SONDE $\left[\frac{12 \times 60}{5}\right]$ x 50 BITS, 3 TIMES	22,000		
ON-BOARD TELEMETRY	140,000		
		TOTAL NO. OF BITS/DAY $\approx$	5×10^7

FIGURE 56

EARTH - MARS - SUN GEOMETRY



M_1 = MARS POSITION AT ARRIVAL AT MARS

M_2 = MARS POSITION AT DEPARTURE FROM MARS

E_1 = EARTH POSITION AT ARRIVAL AT MARS

E_2 = EARTH POSITION AT DEPARTURE FROM MARS

R_1 = EARTH-MARS RANGE AT ARRIVAL AT MARS

R_2 = EARTH-MARS RANGE AT DEPARTURE FROM MARS

LAUNCH DATE FROM EARTH	ARRIVAL DATE AT MARS	DEPARTURE DATE FROM MARS	α (°)	β (°)	R_1 (MI	R_2 x 10^7)
16 MAY - 1971	14 SEPT - 1971	24 SEPT - 1971	35.5	39.0	4.7	5.2
3 AUG - 1973	2 DEC - 1973	12 DEC - 1973	30.0	36.0	3.6	5.5
5 AUG - 1973	3 DEC - 1973	12 JAN - 1974	30.0	38.0	5.8	9.7
8 SEPT - 1975	16 FEB - 1976	26 FEB - 1976	35.0	36.5	9.2	10.2

FIGURE 57

VISIBILITY OF EARTH FROM MARS, 1971 - 1976

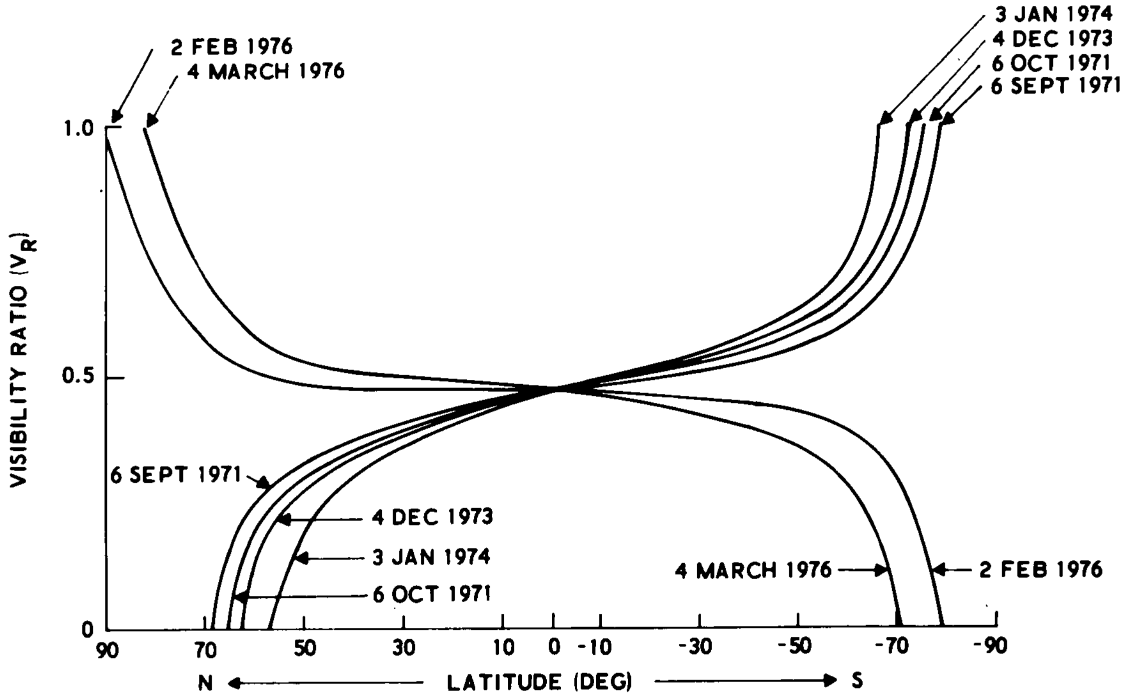
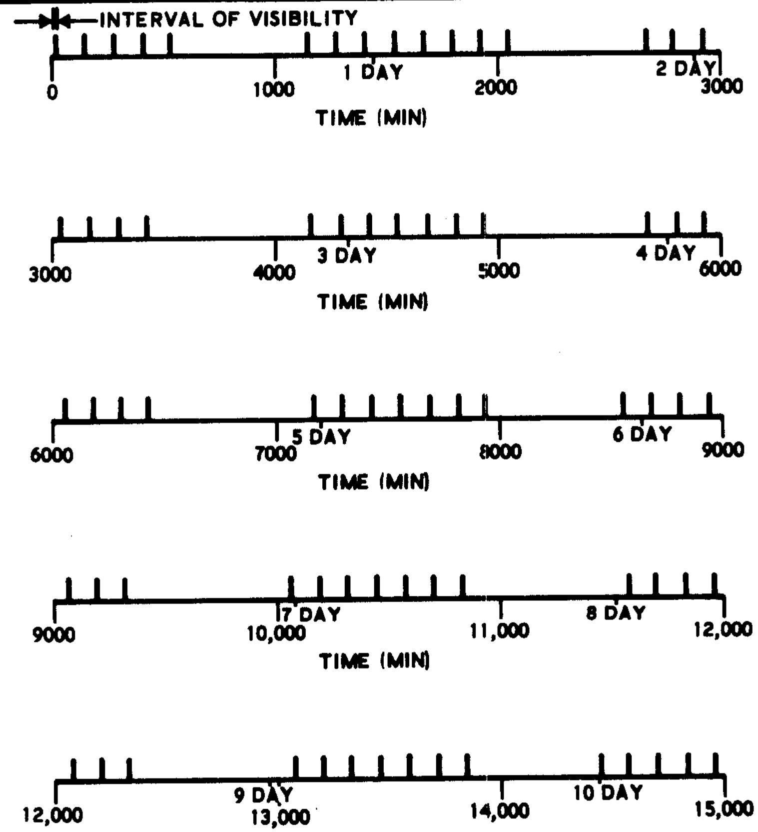


FIGURE 58

VISIBILITY OF ORBITING MARS MISSION MODULE FROM MEM LANDING SITE



AVERAGE DURATION OF VISIBLE PERIOD = 15.7 MIN

SATELLITE IS VISIBLE 7.7% OF TIME

NOTE:

(1) CIRCULAR ORBIT h = 550 KM

(2) ORBIT INCLINATION = 70°

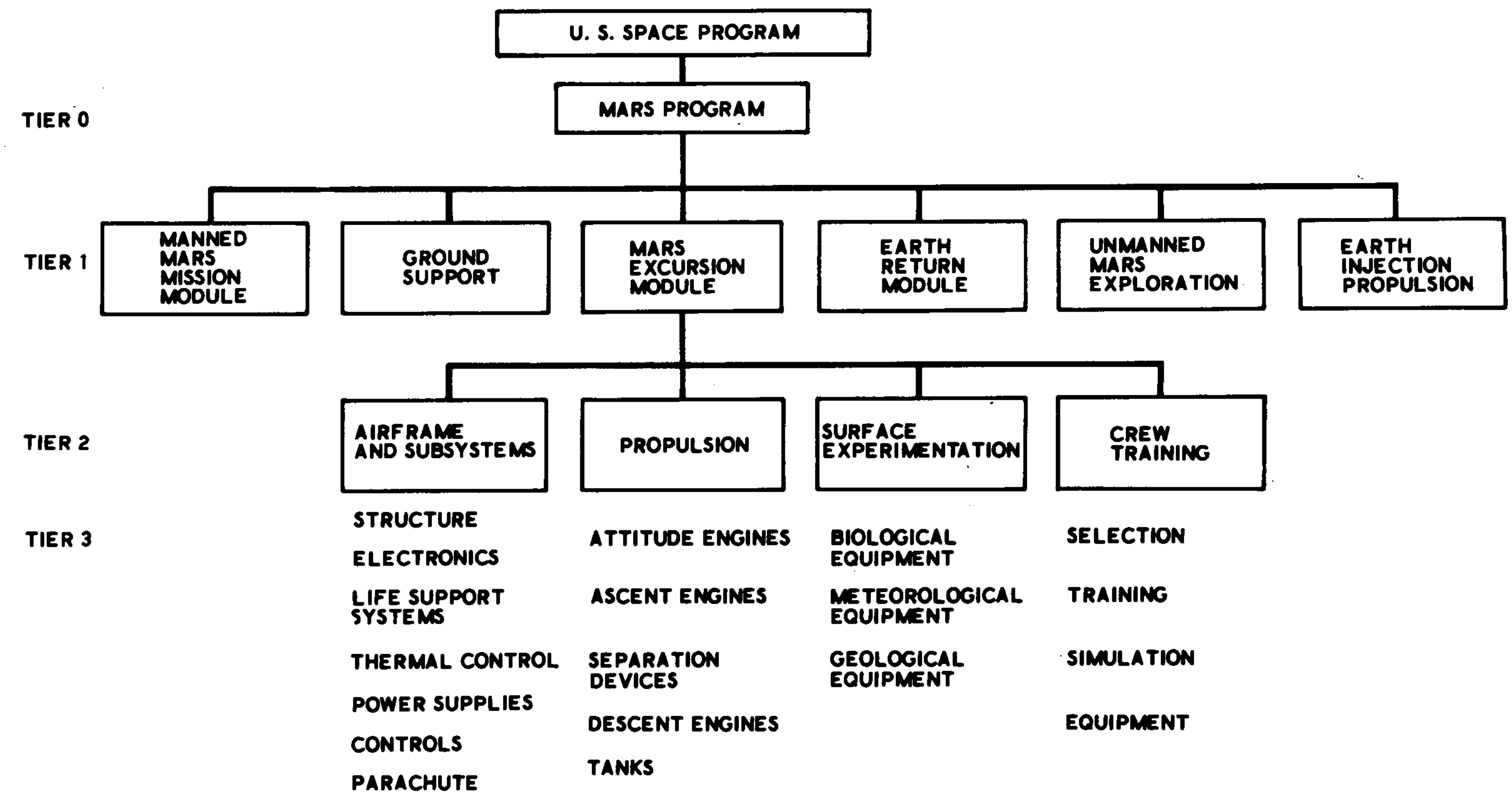
(3) LANDING SITE LATITUDE = 70°

(4) SATELLITE IS CONSIDERED TO BE VISIBLE WHEN IT IS 5° OR MORE ABOVE THE HORIZON

AERONUTRONIC DIVISION

FIGURE 59

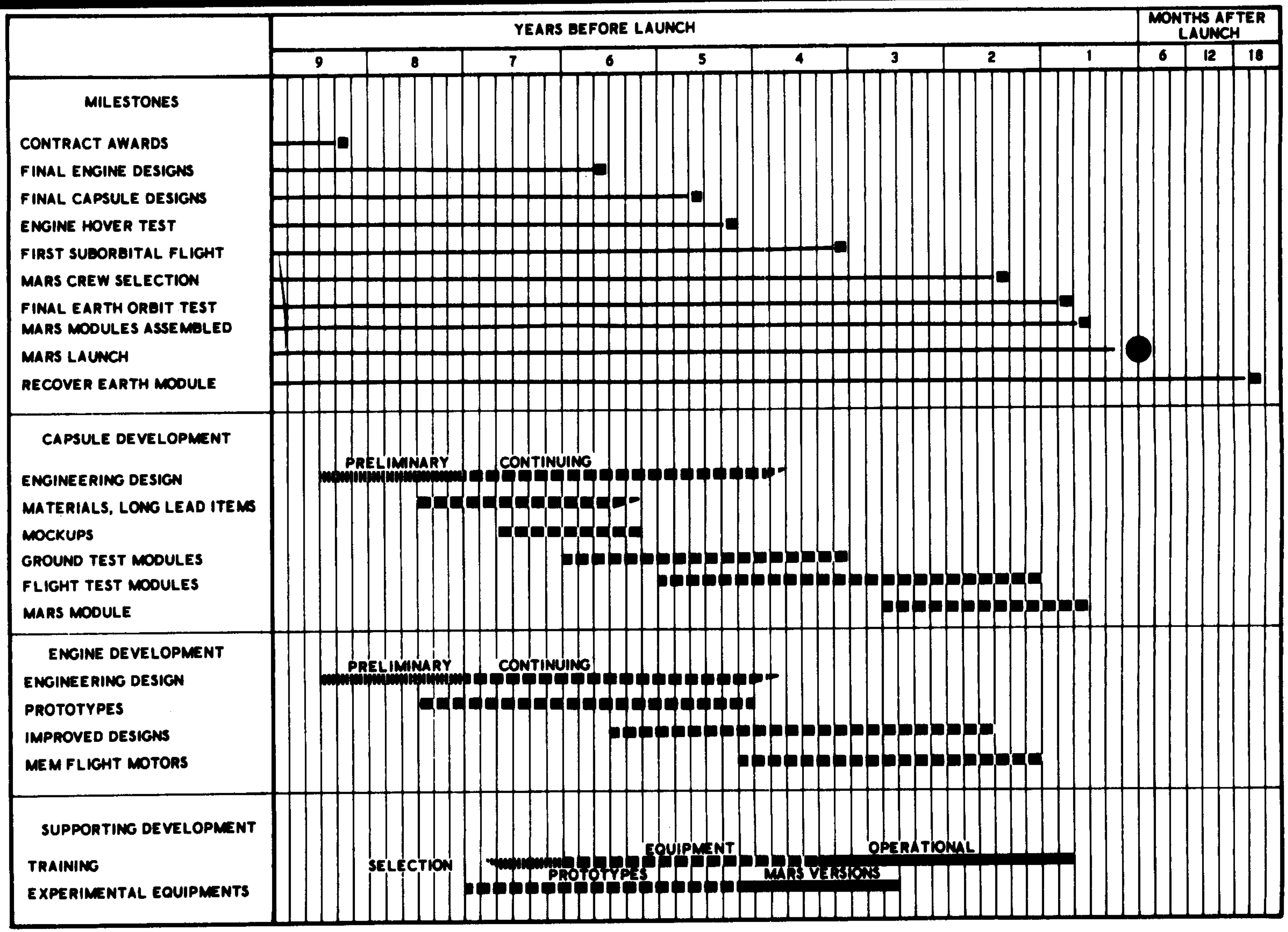
MARS MISSION DEVELOPMENT PROGRAM



Ford
AERONUTRONIC DIVISION

FIGURE 60

MARS EXCURSION MODULE DEVELOPMENT SCHEDULE



AERONUTRONIC DIVISION

FIGURE 61

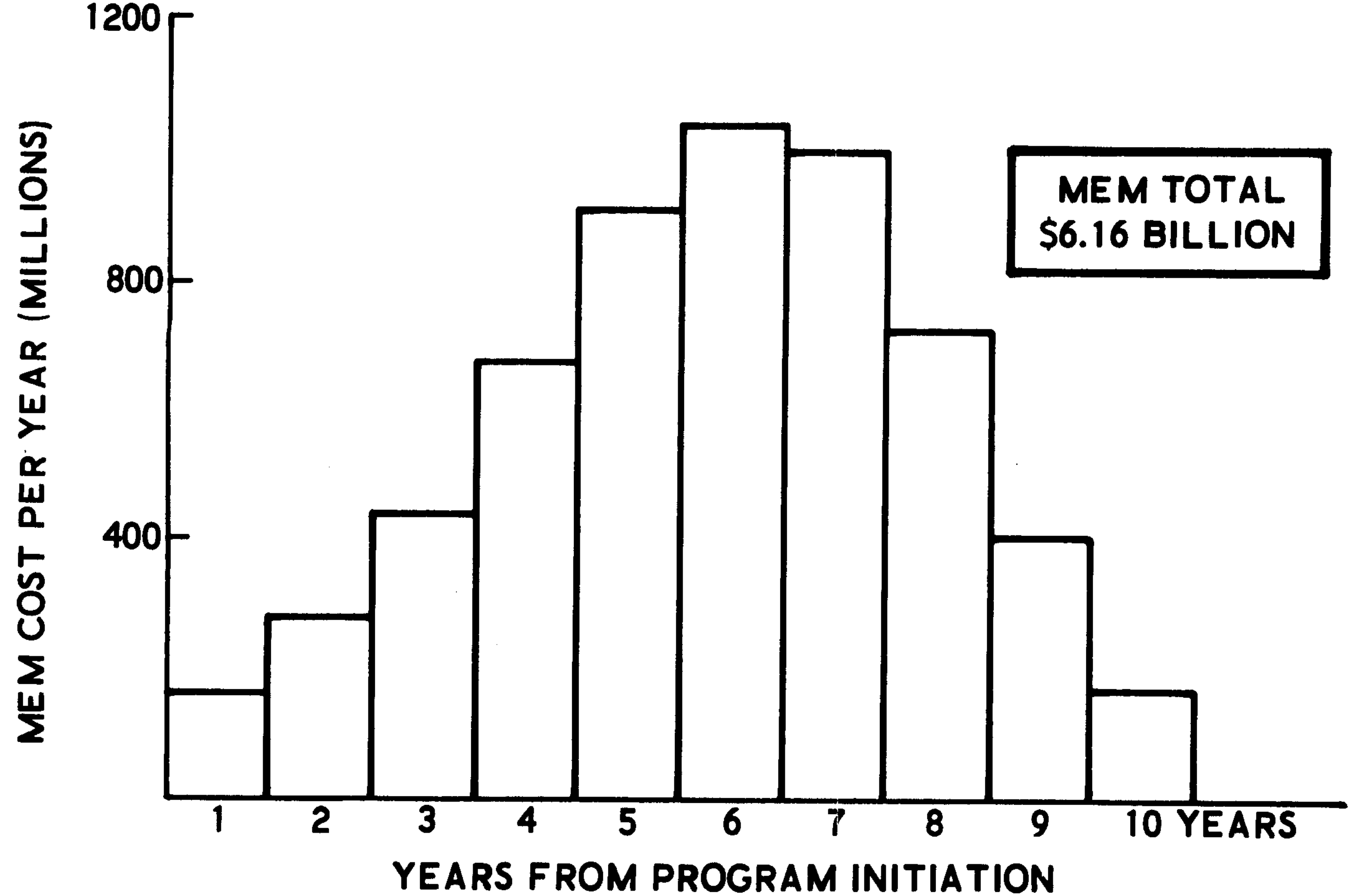
MARS EXCURSION MODULE COST ANALYSIS

		COST (BILLIONS $)
• FLIGHT HARDWARE		
AIRFRAME AND PROPULSION		2.26
EXPERIMENTAL EQUIPMENTS		.02
	TOTAL	2.28
• TOTAL PROGRAM COST		
	37% (TOTAL COST) =	2.28
	TOTAL COST =	6.16
• PROGRAM COST BREAKDOWN		
ENGINEERING		2.65
SUPPORTING RESEARCH		0.19
SYSTEM STUDIES		0.12
FLIGHT HARDWARE		2.28
GROUND SUPPORT (INCLUDING BOOSTERS)		0.19
FACILITIES		0.55
MISSION AND PAYLOAD INTEGRATION		0.18
	TOTAL	6.16
• COST FACTORS		
MEM VEHICLE		$3000/Pound
EXPERIMENTAL EQUIPMENT		$ 950/Pound

Ford AERONUTRONIC DIVISION

FIGURE 62

MARS EXCURSION MODULE FUNDING PROFILE



26990

PART 13

PRELIMINARY DESIGN OF A MARS-MISSION EARTH REENTRY MODULE

by

D. J. Shapland
Lockheed Missiles and Space Company
Contract No. NAS9-1702

NOTATION

A	reference area
C_L	lift coefficient
C_D	drag coefficient
D	drag force
E	modulus of elasticity
F_{tu}	ultimate tensile strength
G	normalized acceleration
L	length
p_{CR}	critical pressure
q	heat-transfer rate
q_s	heat-transfer rate at stagnation point
R	radius
Re	Reynolds number
Re_c	transition Reynolds number
r_N	nose radius
r_B	radius of circular cross section (at forecone/aftcone junction)
S	distance from stagnation point (along body contour)
T	time
t	effective thickness
V_E	reentry velocity

V_H	horizontal velocity
V_V	vertical velocity
y	distance from stagnation point (normal to body)
W	weight of vehicle
α	angle of attack
δ	cone rake angle or shock-layer thickness
ϵ	emissivity
θ	sweep back angle
ρ_o	stagnation density
ρ_{SL}	sea level density
σ	actual stress or standard deviation
ϕ	roll angle

Section 1
INTRODUCTION

1.1 STUDY OBJECTIVES

The overall objective of the study was the preliminary design of an Earth reentry vehicle for use at the termination of a manned Mars mission departing Earth in the years 1971, 73, 75. Reentry speeds are in the range of 45,000 ft/sec to 65,000 ft/sec. Except for checkouts of systems and components, the reentry module is not used during the mission; thus the main study responsibility covered the phase from separation from the Mars Mission Module (MMM) through reentry to ground landing. Subsidiary objectives of the study are as follows:

- Establish reentry criteria and perform configuration studies
- Investigate vehicle aerodynamics and trajectory dynamics; establish operational profiles
- Study heating environment and define heat-protection system
- Design major structure
- Define essential subsystems, i.e., controls, life support, power supplies, navigation and guidance, and communications
- Establish integrated design of reentry module
- Insure compatibility with MMM
- Define problem areas and recommend subjects for further study

The design criteria generated during the study, the problem areas defined, and the recommendations made are expected to serve as a basis for future work.

1.2 STUDY GUIDELINES

The following guidelines were established by the Manned Spacecraft Center.

- Mission years: 1971, 73, 75
- Module to be used during Earth reentry phase only
- Maximum reentry velocity: 65,000 ft/sec
- No retro-propulsion to be used
- Maximum deceleration limit: 10 G
- Abort mode to be considered in structural design
- Normal crew: four men; emergency crew: six men
- Land or sea recovery; 3-day survival capability
- Scientific payload: 800 lb
- Module target weight: 15,000 lb
- State-of-the-art technology to be employed

Details of specific missions for the opposition years 1971, 73, 75 were provided. Associated reentry speeds are shown on a plot (Fig. 1-1) of Earth reentry velocity versus Earth departure date, for a minimum mass on Earth parking orbit criterion. The spread in the results reflects the choice of chemical or nuclear propulsion. Since the reentry speeds (and consequently, the conditions that must be satisfied in the vehicle design) vary with the mission chosen, an optimized design to satisfy all missions is not practical. The approach taken was to design for the maximum speed of 65,000 ft/sec, since this represented the most stringent conditions in all areas. However, the less severe requirements associated with the lower reentry speeds will be apparent from the study.

1.3 TYPICAL REENTRY TRAJECTORY

The general reentry concept is shown diagrammatically in Fig. 1-2 in the form of a typical reentry trajectory for the Mars-Mission Earth Reentry Module (MMERM).

The reentry module will be stowed in the MMM approximately in the position shown in the diagram. Before separation, a checkout period of about six hours is required to

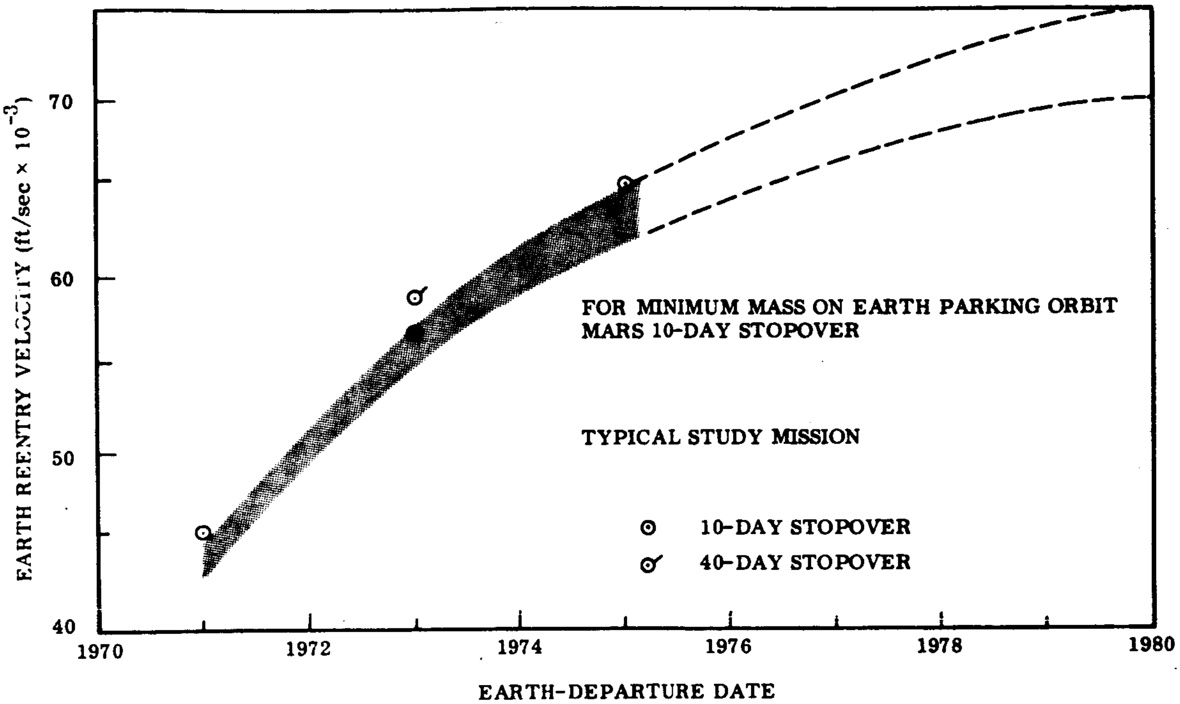


Fig. 1-1 Earth-Reentry Velocities from Mars Missions

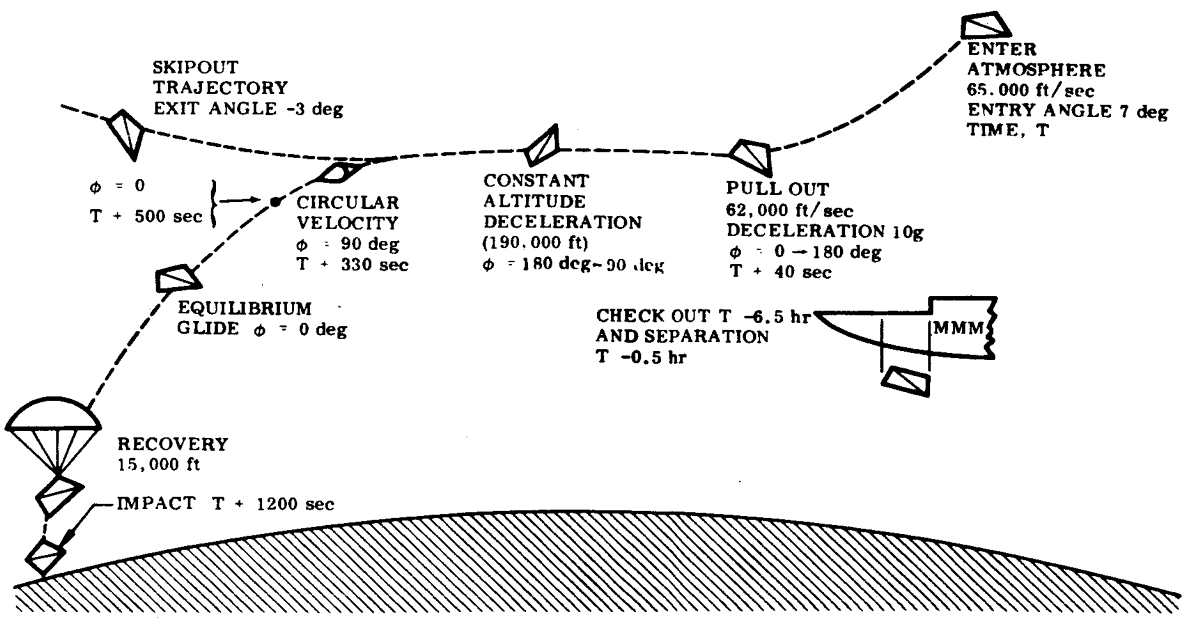


Fig. 1-2 Reentry Trajectory (Diagrammatic)

ensure that all MMERM systems are operating satisfactorily. During this period, the MMERM will be extended from the mission module to provide access to the celestial sphere so that the on-board inertial platforms can be erected accurately. About half an hour before the reentry module separates from the main spacecraft, the final velocity change will be made by means of MMM propulsion. This final correction must be accurate enough to enable MMERM to enter the Earth's atmosphere within very narrow limits – the reentry corridor. This corridor is bounded on the lower side (undershoot) by the 10-G deceleration tolerance limit of the crew and on the upper side (overshoot) by the requirement that the atmosphere be dense enough to ensure sufficient negative lift to counteract the centrifugal force generated during deceleration at constant altitude.

After separation, a vacuum trajectory is followed, and the vehicle is oriented in the correct reentry attitude. The reentry angles range between 6.9 and 7.9 deg for the overshoot and undershoot boundaries. The vehicle pulls out at about 200,000 ft. At this point, maximum deceleration and heating are encountered. A period of constant-altitude deceleration follows during which the vehicle is rolled to direct the lift vector toward the Earth. This negative lift keeps the vehicle within the atmosphere. The roll angle ϕ is gradually reduced from 180 to 90 deg – at which point circular velocity is reached. From then on, a nominal direct reentry trajectory may be followed during which positive lift is applied and the roll angle is reduced to zero; an equilibrium glide path is then maintained. If the roll angle is held at some value between 0 and 90 deg, greater cross range may be obtained, but at the expense of downrange.

A skip maneuver may be initiated if a long-range trajectory is desired. Up to 10,000-nm range may be obtained, depending on the exit conditions. The skip trajectory is followed by a second reentry, for which the heating and deceleration forces are much less severe than are those encountered during initial pull-out.

Conventional parachute recovery is assumed, since adequate range control is provided by using the relevant roll-control program.

Section 2
VEHICLE SELECTION AND PERFORMANCE

The blunt-compact-body approach has been used in Projects Mercury and Apollo. The same logic underlying this approach applies this principle (with suitable shape modifications) to developing an Earth reentry-vehicle configuration for return from planetary missions.

2.1 CHOICE OF CONFIGURATION

The basic requirements for a high-speed reentry configuration are as follows:

- Low W/C_LA for deceleration at relatively low density
- Adequate L/D value to meet reentry-corridor and trajectory-control requirements
- Lowest weight compatible with volume requirements
- Efficient heat-protection system; an ablator is essential.

At the reentry speeds considered, radiation heat transfer will be of major importance. From this point of view, a blunt shape is undesirable. However, convective heating cannot be ignored, and convective heating rates are reduced by increasing bluntness. A compromise indicates a conical shape that retains a degree of bluntness (low W/C_DA) consistent with lifting requirements. The crew must be positioned in the vehicle such that the resultant force vector acts transverse. An offset C. G. is required for trim control. An aerodynamically stable body is desired to maintain the proper acceleration orientation, achieve maximum drag, and maintain lift capability.

These general considerations, backed up by theoretical analyses, led to the selection of of the configuration shown in Fig. 2-1. The reference configuration consists of two portions of two cones. The forebody (main heat shield) is a blunted circular cone,

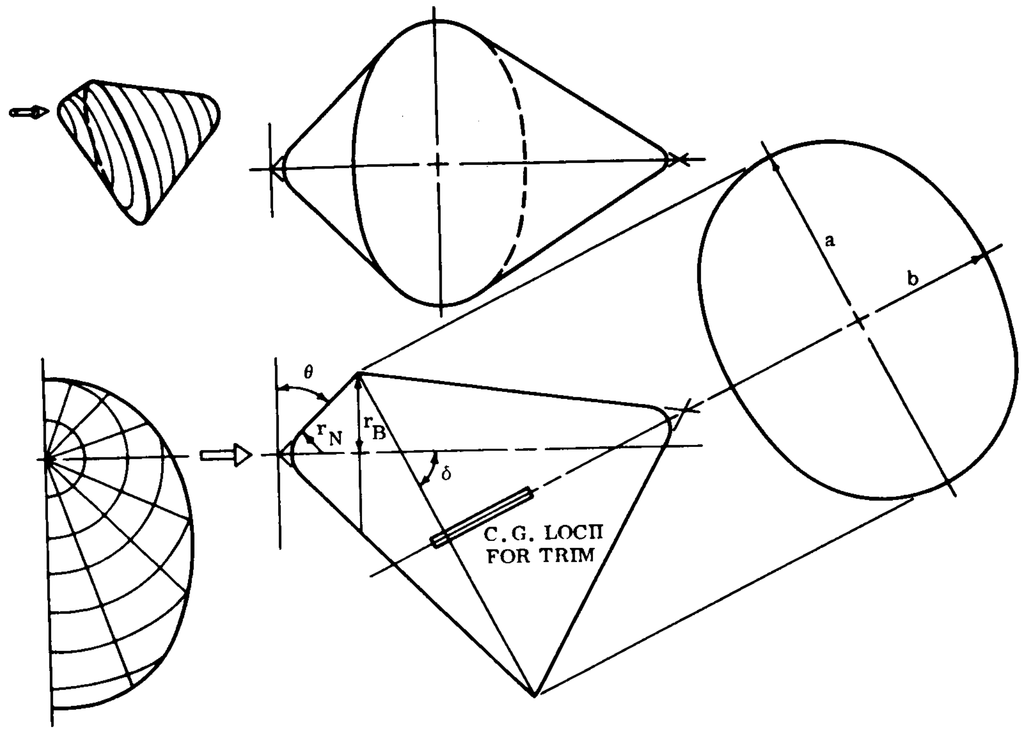


Fig. 2-1 Reference-Configuration Geometry

raked off at an angle δ to yield the desired L/D ratio. The afterbody is a right elliptic cone fitted to the cross section of the forebody. The vehicle trims at zero angle of attack (relative to the forecone center line), which greatly facilitates flow-field analyses. Geometry variations result in a family of vehicles with L/D values ranging from 0.5 to 1.0. Considerations that led to the selection of the LMSC configuration are summarized in Table 2-1.

Half way through the study, a particular design configuration was chosen for detailed analysis and design integration. This configuration had a lift-to-drag ratio of 0.6, a half-cone angle of 35 deg (forecone), a drag parameter of 204 lb/ft^2, and a volume of 500 ft^3.

Table 2-1

REENTRY CONFIGURATION SELECTION

Criteria	Technique	Requirement
Corridor Width	10-nm Approach Guidance Accuracy	Low L/D
Landing Point Control		
Cross-Range	Approach Orbit Inclination	Low L/D
Down-Range	Skip Maneuver	Low L/D
Touchdown	Parachute	Low L/D
Trajectory Control	Trimmed Lift Roll Control	Reaction System
Crew Orientation	Transverse Loading	Minimum Size
Total Heat Load	High Altitude Minimum Flight Time	Small W/C_LA Low L/D
Heat-Transfer Calculation Reliability	Simply Defined Flow Field	Conical Forebody at Zero α.
Weight Optimization	Simple Geometry	Small number of parameters
Volume and Arrangement	Maximum Useful Space	Compact Shape

2.2 REENTRY CORRIDOR REQUIREMENTS

The available reentry corridor width for return from a Mars mission is small. The width is governed mainly by the value of the reentry velocity and the L/D capability of the vehicle. The L/D requirement to achieve a given corridor width increases rapdily with increasing reentry velocity, and a practical limit of about 1.0 results for reentry at 65,000 ft/sec. Figure 2-2 illustrates the corridor available as a function of L/D and reentry velocity for a 10-G undershoot boundary. The effect of W/C_LA is slight. Note that for a 65,000 ft/sec reentry speed, increasing the L/D value from 0.6 to 1.0 only increases the corridor width by about 2.5 nm. The available corridor width is, therefore, limited to about 11 nm (L/D = 1.0) for the 65,000 ft/sec reentry; consequently approach-guidance requirements are very stringent. If the acceleration limit on the lower boundary can be increased, the corridor could be opened-up considerably – approximately 1 nm per G.

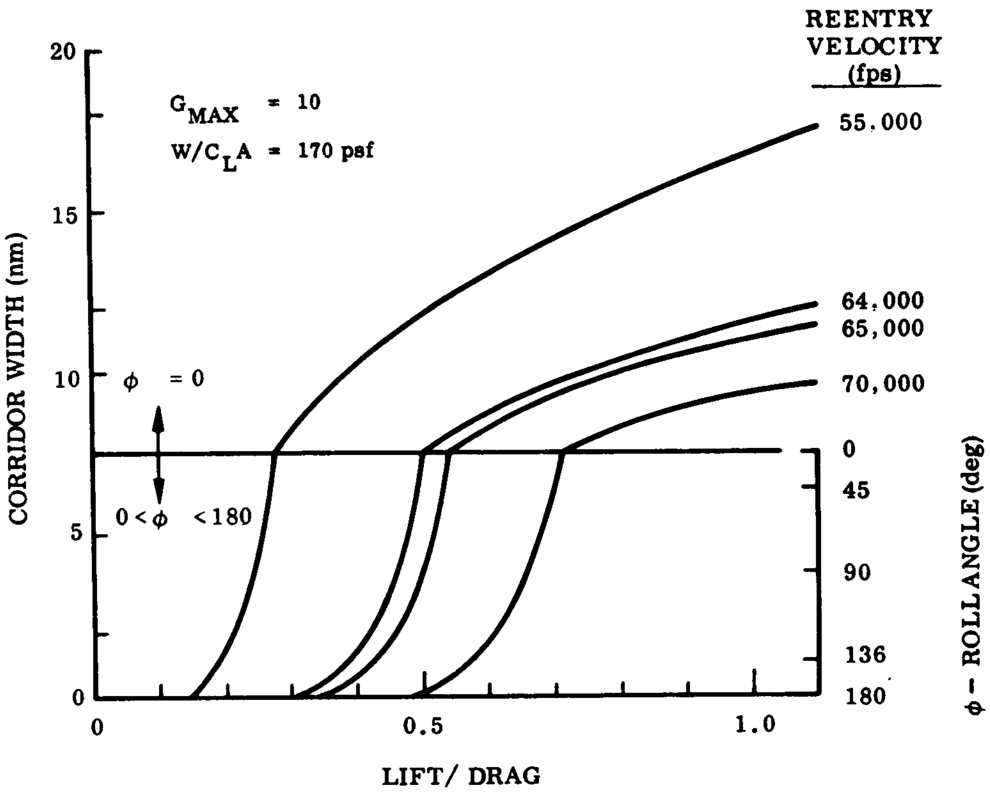


Fig. 2-2 Reentry Aerodynamic Requirements

2.3 CONFIGURATION AERODYNAMICS

Large variations in aerodynamic characteristics can be obtained for the family of shapes by varying the geometry, as shown in Fig. 2-3. The influence of sweep-back angle θ and slice angle δ are illustrated. The effect of bluntness ratio r_N/r_B is not very significant. The drag coefficient C_D is based on the circular cross-section area at the upper junction of forecone and after-body. The attainable $W/C_L A$ value is limited by the fact that as θ or δ are decreased, the geometry tends to a flat plate. For $L/D = 0.6$ and a reference area of 123 ft^2, the limiting value of $W/C_L A$ is about 160 lb/ft^2 at a sweep-back angle of about 30 deg. The static stability of the configuration family is good over a wide range of C. G. positions for the smaller sweep-back angles. Higher sweep-back shapes impose more stringent C. G. requirements for satisfactory lateral stability.

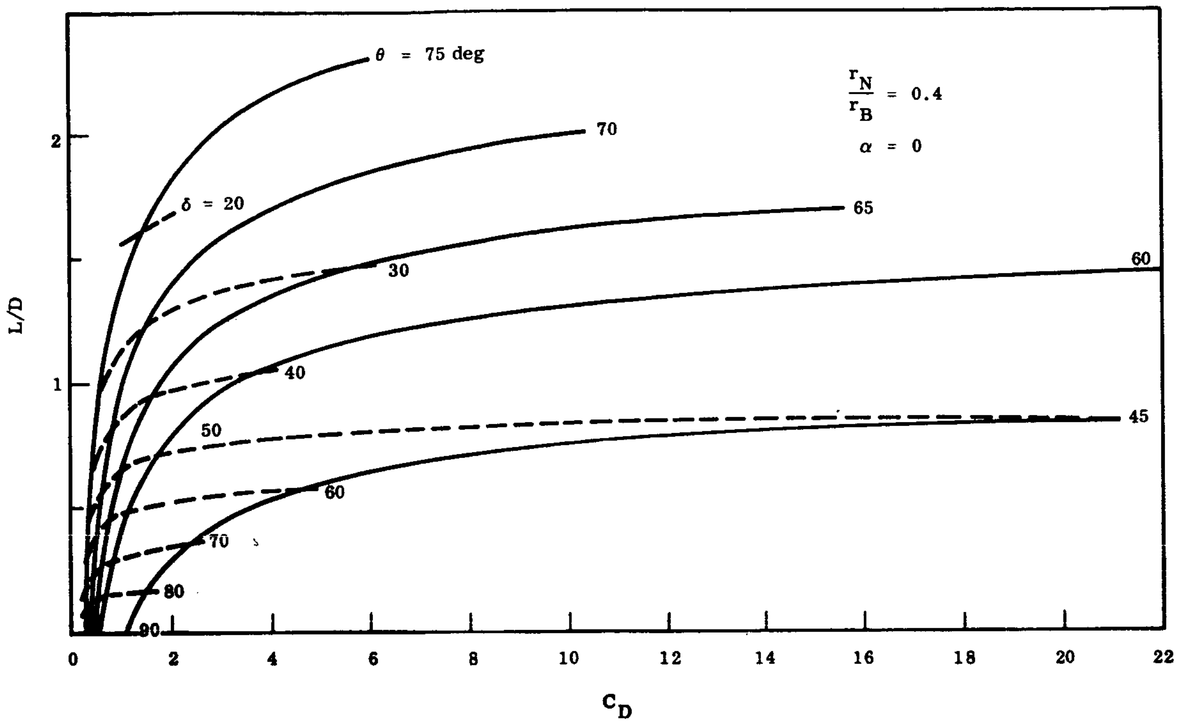


Fig. 2-3 Trim Aerodynamics Characteristics

Maneuvers within the corridor are achieved by rolling the vehicle about the velocity vector by means of a reaction-control system. Such a system makes it possible to direct the lift vector as required. To prevent any significant degradation of the corridor width, the roll rate at pullout (0 to 180 deg) should not be less than 40 deg/sec. Excursions in pitch and yaw due to C. G. shift can involve large moments which result in large fuel requirements. Consequently, internal mass balancing is proposed to maintain trim.

2.4 RANGE-MANEUVER CAPABILITY

Cross-range control is achieved by selecting a suitable roll-angle program. A skip maneuver may be used to increase the longitudinal range. Figure 2-4 illustrates a typical footprint for direct reentry. For reentry along the upper boundary, the nominal

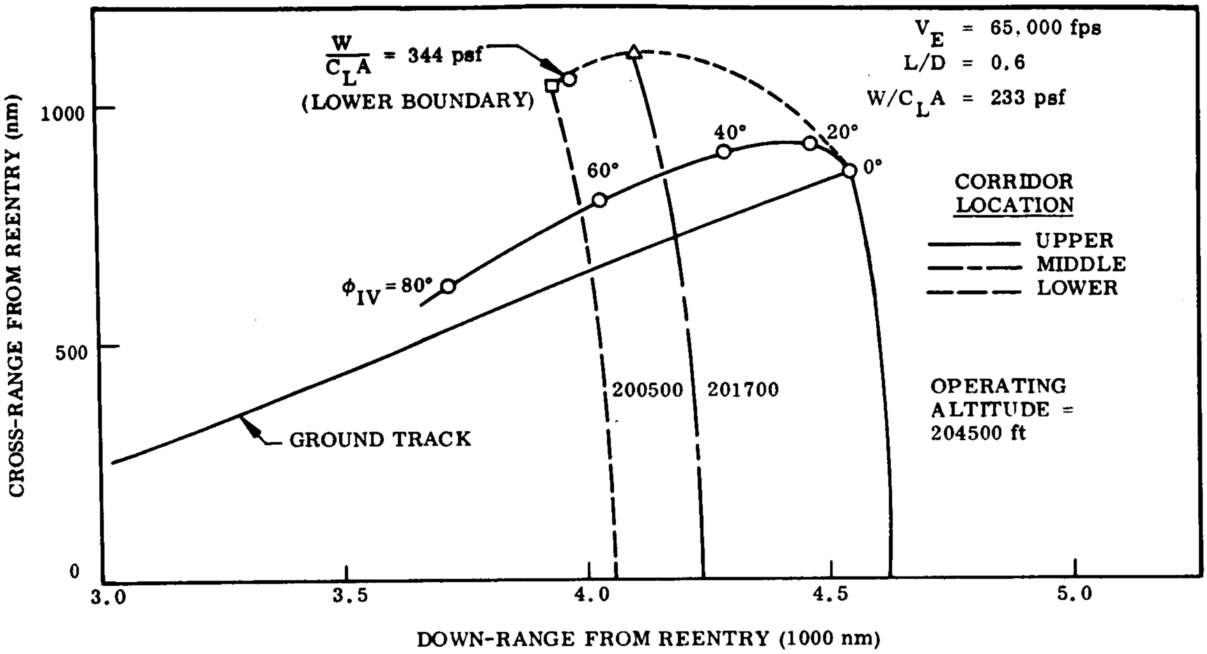


Fig. 2-4 Direct Reentry Maneuver Capability

roll program $\phi_{IV} = 0$ gives a down-range capability of about 4,500 nm and a cross-range capability of some ±800 nm. If the roll angle is held at some value between 0 and 90 deg (after circular velocity has been reached), additional cross- and down-range combinations are possible. Adjusting the operating altitude also alters the pattern, as shown in the figure. Variations in the lift parameter produce only minor changes in the maneuver capability.

A skip trajectory may be initiated to increase down range. The maximum skip range possible is about 10,000 nm. Second reentry provides a further cross-range capability of about ±200 nm.

As these curves in the illustration demonstrate, adequate landing-point control to account for guidance errors accumulated during the initial reentry and skip phases of the trajectory is available.

2.5 OPERATIONAL TRAJECTORY

To illustrate the ability of the reentry system to satisfy mission requirements, an exact Mars to Earth transfer trajectory was calculated for the 1975 opposition. The ground track is shown in Fig. 2-5. Reentry occurs at the point P. A direct reentry trajectory would not land the vehicle at a suitable site; however, a short skip will result in a landing at Edwards Air Force Base. The approach orbit inclination controls the skip direction, but modifications to the heading can be made by means of the cross-range control at first reentry. (This capability is used mainly to correct for guidance errors.) The reentry pattern can be rotated about the subpoint by changing the orbit inclination so that Woomera may also be reached by employing a short skip trajectory.

Mass changes due to ablation will alter the trajectories slightly. The main effect is to reduce cross-range capability by about 20 nm and down-range capability by about 300 nm.

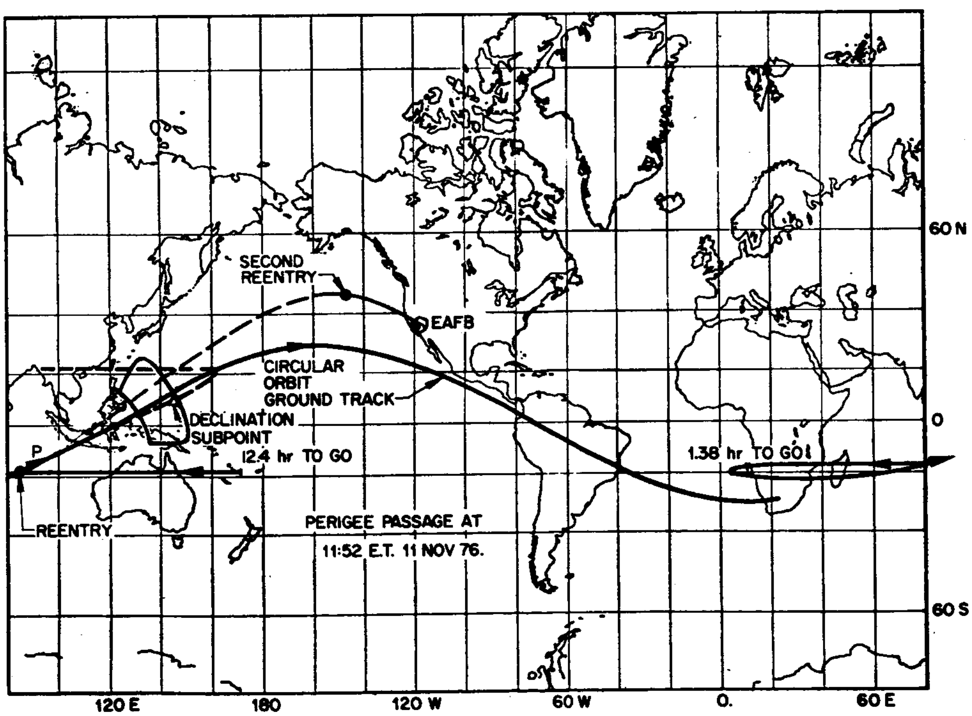


Fig. 2-5 Earth-Return Trajectory (8 Sep 75 Mission)

Section 3

REENTRY HEATING AND THERMAL PROTECTION

This section deals with the thermal environment encountered by the MMERM configuration as it enters the Earth's atmosphere along the proposed trajectories.

Prior to this study, relatively little consideration has been given to the thermal conditions associated with such extreme flight velocities. Consequently, some effort was devoted to the development of suitable analysis techniques and to the extension of the thermodynamic properties of equilibrium air to the temperature range of interest. All the results summarized here are for reentry at 65,000 ft/sec. The study showed that reentry at this speed is thermally feasible with an efficient ablation heat-protection system. The heat shield weight is one third of the total vehicle weight.

Typical shock-layer conditions at the stagnation point are given in Fig. 3-1 in terms of density ratio and temperature. The first 200 seconds (this represents the main heating period) from reentry are covered by the figure. Note that the temperatures approach a peak value of 40,000°R.

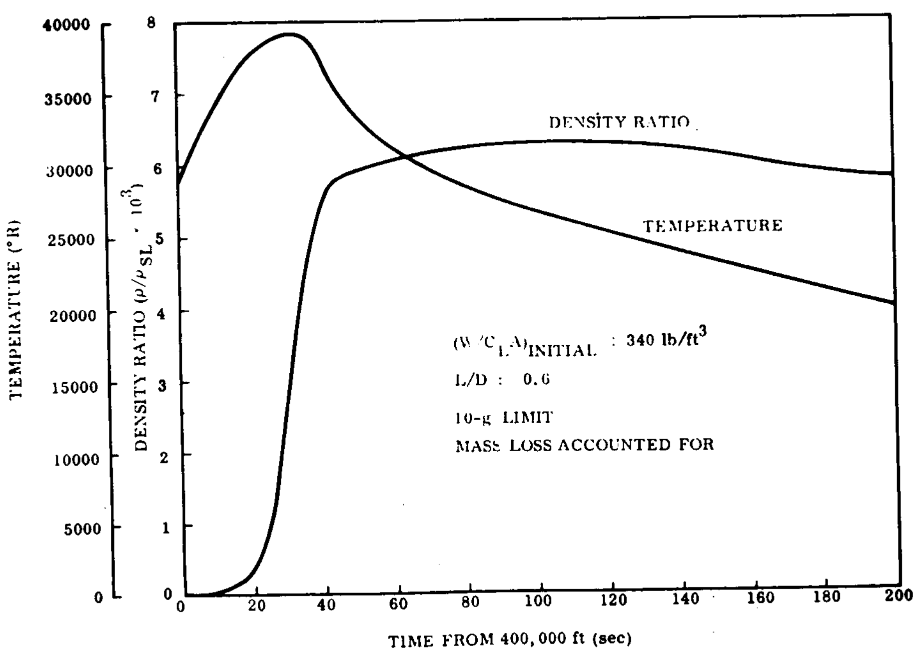


Fig. 3-1 Temperature and Density in the Shock-Layer at the Stagnation Point

3.1 RADIATIVE PROPERTIES OF AIR

At the high speeds considered here, radiative emission from the shock layer will be intense. Thus, a realistic description of the air emissivity is essential. Various data are available, but they show poor agreement, and order-of-magnitude differences are apparent. The available data were reviewed, and the basic assumptions were checked. Figure 3-2 illustrates the uncertainty that exists (the results are for a typical density ratio). The early model was adopted before the results of Nardone, Breene et al. (Ref. 1) became available and was retained as a lower-bound estimate. Breene's values are considerably higher in the range 15,000 to 25,000°R, mainly as a result of the inclusion of contributions due to nitrogen and oxygen deionization at wavelengths between 0.05 and 0.2μ. Independent checks at LMSC showed that the

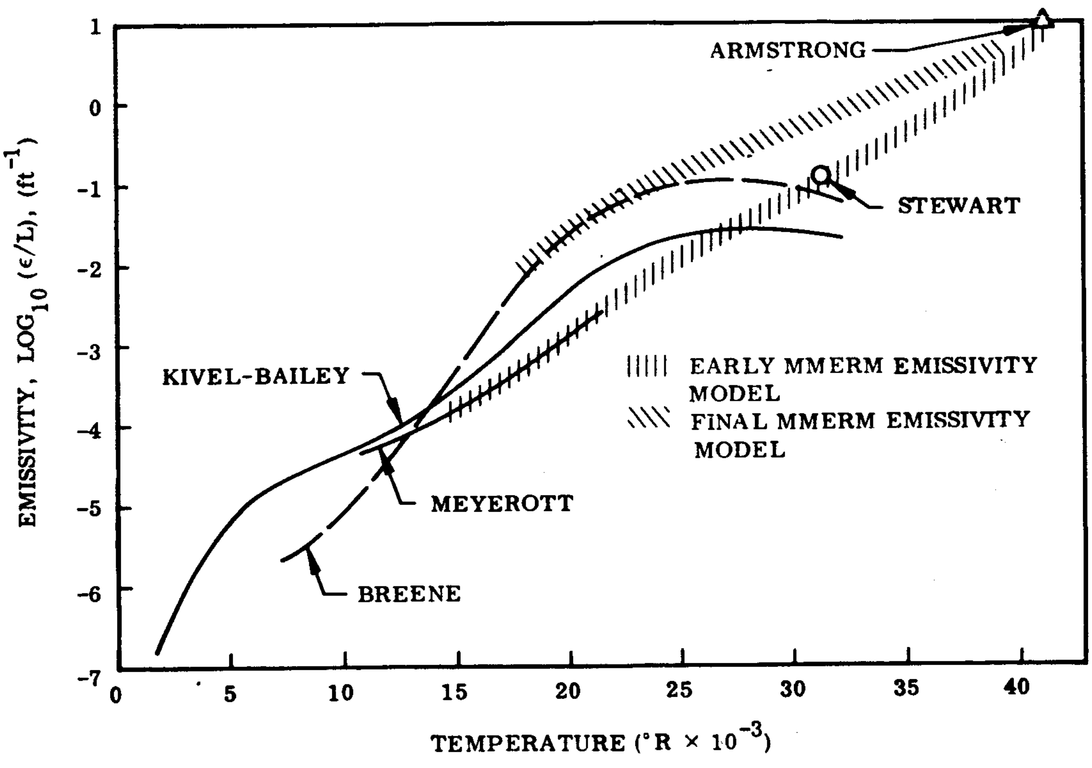


Fig. 3-2 Air-Emissivity Data for $\rho/\rho_{SL} = 10^{-3}$

results were realistic. An upper-bound model was therefore defined which included the atomic-line contribution calculated by Armstrong (Ref. 2) for temperatures above 25,000°R. The work of Kivel and Bailey (Ref. 3), Meyerott (Ref. 4), and Stewart (Ref. 5) was also considered in preparing the models.

3.2 ENTROPY-LAYER EFFECTS

The entropy change across the normal shock (at the stagnation point) is much greater than that for the conical shock. Another uncertainty arises from this (vorticity) effect, since the convective heating level on the cone depends on the thickness of the boundary layer relative to the entropy layer. Figure 3-3 shows the Reynolds number per unit length on the cone for two distinct streamlines; one emanates from the normal shock at the nose and the other from the conical shock. A variation by a factor of 10 in the

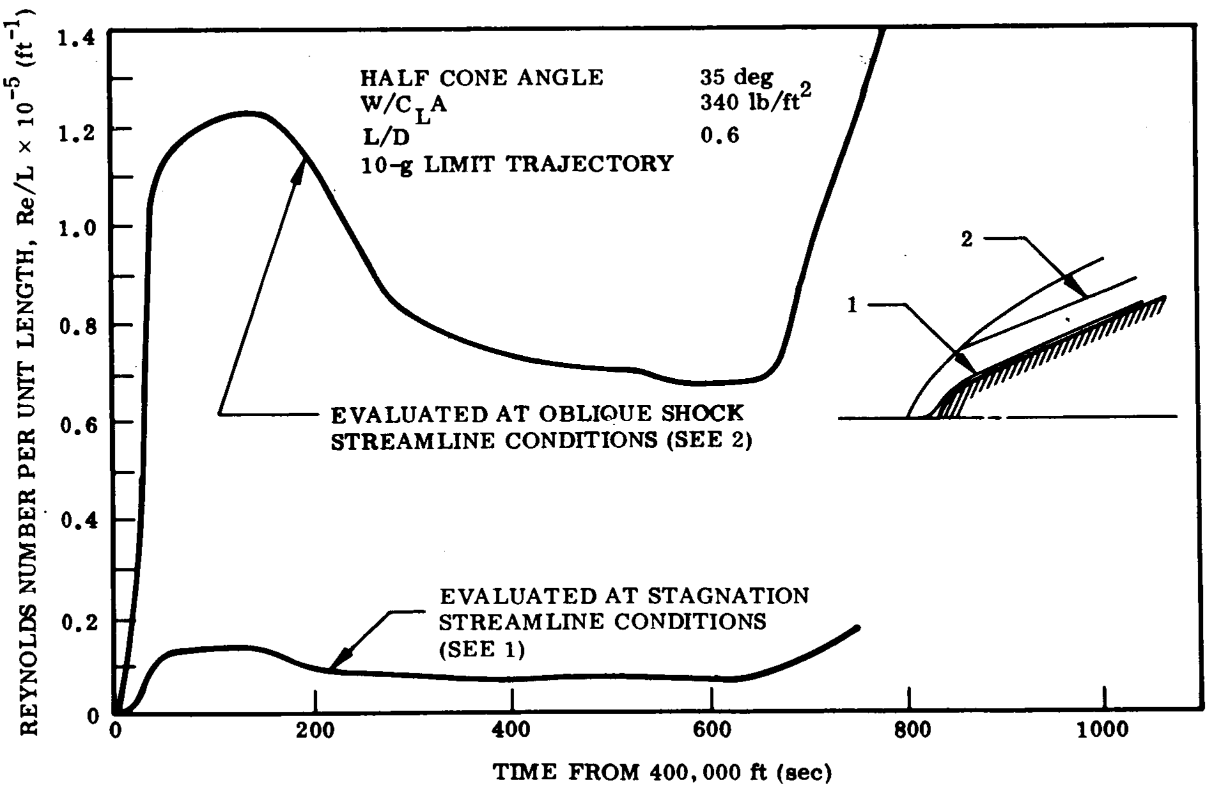


Fig. 3-3 Conical Forebody Reynolds-Number History

Reynolds number per foot is evident, depending on the source of the streamline. The extent of the vorticity interaction was investigated, and it was concluded that, over most of the body, boundary-layer phenomena are governed by flow emanating from the conical shock. This result has an important effect on transition conditions. For the transition criterion adopted ($Re_c = 2 \times 10^5$), the boundary layer will be turbulent over most of the surface.

3.3 RADIATION LOSS FROM SHOCK LAYER

At a speed of 65,000 ft/sec, the air in the shock layer cannot retain its energy, and radiative cooling of the gas is appreciable. The effect is illustrated in Fig. 3-4 for the stagnation point. At the shock, adiabatic and actual values are identical, since the air has had no time to radiate away any energy. As the air moves towards the

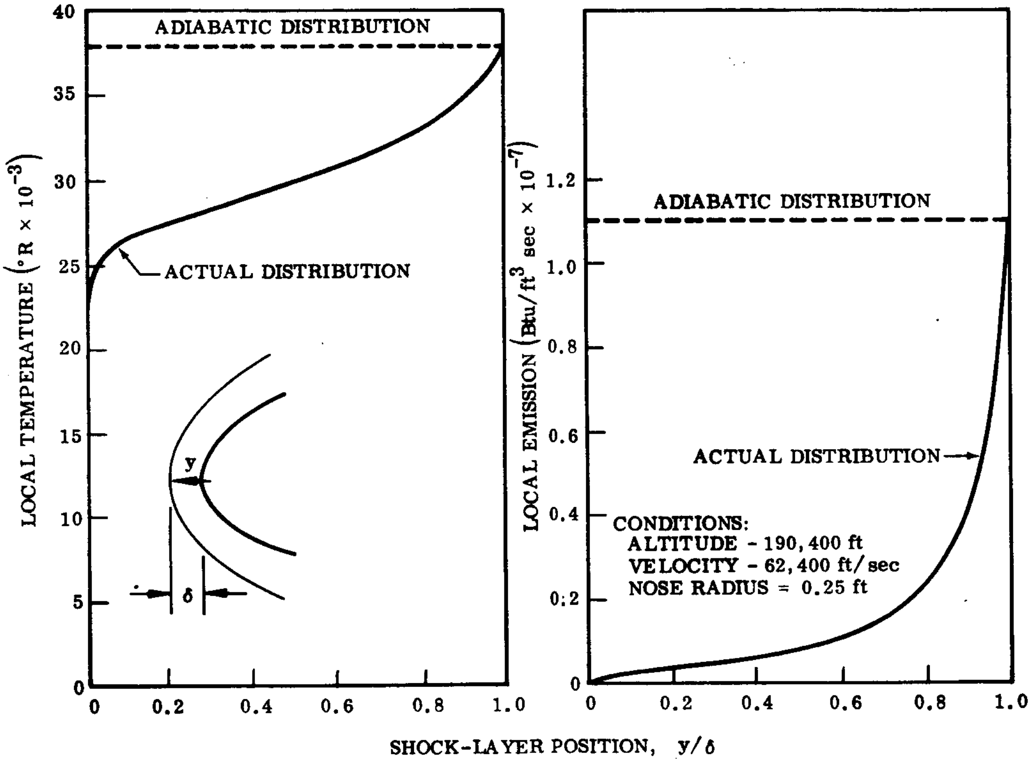


Fig. 3-4 Influence of Cooling on Temperature and Radiation Distributions through the Shock Layer at the Stagnation Point

surface, it cools, and its temperature and local emission drop rapidly. The integrated value of the radiation to the stagnation point is about 20 percent of that expected under adiabatic conditions.

Approximate, closed form, analytic methods for predicting the nonadiabatic radiative heat transfer to the body were developed during the study. The results are in good agreement with more rigorous techniques and much less time consuming.

3.4 RADIATIVE HEAT TRANSFER

As an example of the results of the radiation heat transfer calculations, the heating distribution on the conical forebody of a configuration having a half-cone angle of 45 deg is shown in Fig. 3-5. The trajectory conditions are for maximum radiative heating. The influence of radiation cooling is apparent. The radiative flux decreases away from the stagnation point and then rises again along the conical surface. Near the sphere-cone junction, the entropy layer radiates strongly; the layer becomes

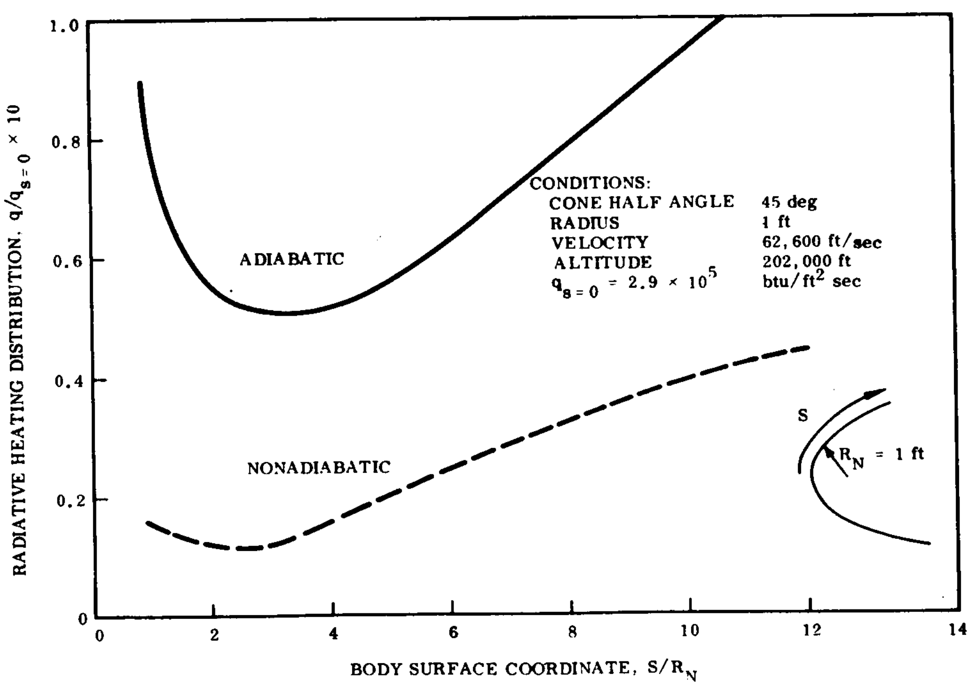


Fig. 3-5 Radiative-Heating Distribution on the Conical Forebody

thinner and cooler in passing rearward, and the radiation drops. However, the shock-layer thickness increases towards the rear, giving a subsequent increase in the heat transfer.

3.5 ABLATIVE HEAT-PROTECTION SYSTEM

Nylon-Phenolic (50-50 by weight) was selected as the state-of-the-art heat shield material because of its high heat absorption capability, fabricability, and the fact that its performance under the extreme conditions could be predicted by theoretical means.

A model describing the degradation of nylon-phenolic has been developed at LMSC. This model accounts for the complex processes of pyrolysis, char formation, effluence of gases through the material, and gas reaction and sublimation at the surface. Density and temperature distributions through the ablator are controlled by the chemical processes and environment. The model was extended to cover the range of interest of the MMERM vehicle. Normally, the surface recedes due to chemical erosion (reaction of boundary layer air and char). With extreme radiation heating rates, mass injection rates become sufficiently large to cause "boundary layer blow off"; convective heating and chemical erosion cease, and sublimation of the surface occurs.

3.6 RESULTS OF HEAT-TRANSFER ANALYSIS

Representative results are given in Figs. 3-6 through 3-8. The effects of radiation loss and mass injection were included in the calculations. Constant mass trajectories were used in the general analyses, but the influence of ablation mass loss on the trajectory was accounted for in all calculations pertaining to the design case. Conventional techniques were used for estimating the convective heating (e.g. Hoshizaki, Ref. 6). Nonequilibrium radiation was found to be negligible. It was assumed that boundary layer "blow-off" dominated when radiation-convection coupling effects might be significant.

Figure 3-6 illustrates the radiative and convective heating histories for a typical point on the cone (half angle 35 deg). The total heat transferred by convection far exceeds that transferred by radiation. The thermal degradation of the heat shield at this station is described by Fig. 3-7. The lower curve shows the surface recession history, while the upper curve indicates the depth to which the pyrolysis zone has penetrated. The final extent of the degradation is about 3 in; additional material is required for insulation purposes. When the heating rate is greatest, the char-layer thickness is relatively small; however, later in the trajectory, the char layer grows and reaches a maximum at second reentry.

Conditions at the nose are more extreme (Fig. 3-8). Radiation heat transfer is more significant, and recession rates as high as 0.3 in/sec are experienced. On the afterbody, the flow field is less well defined. Conservative estimates yielded an ablator thickness of 1 in. over the whole surface.

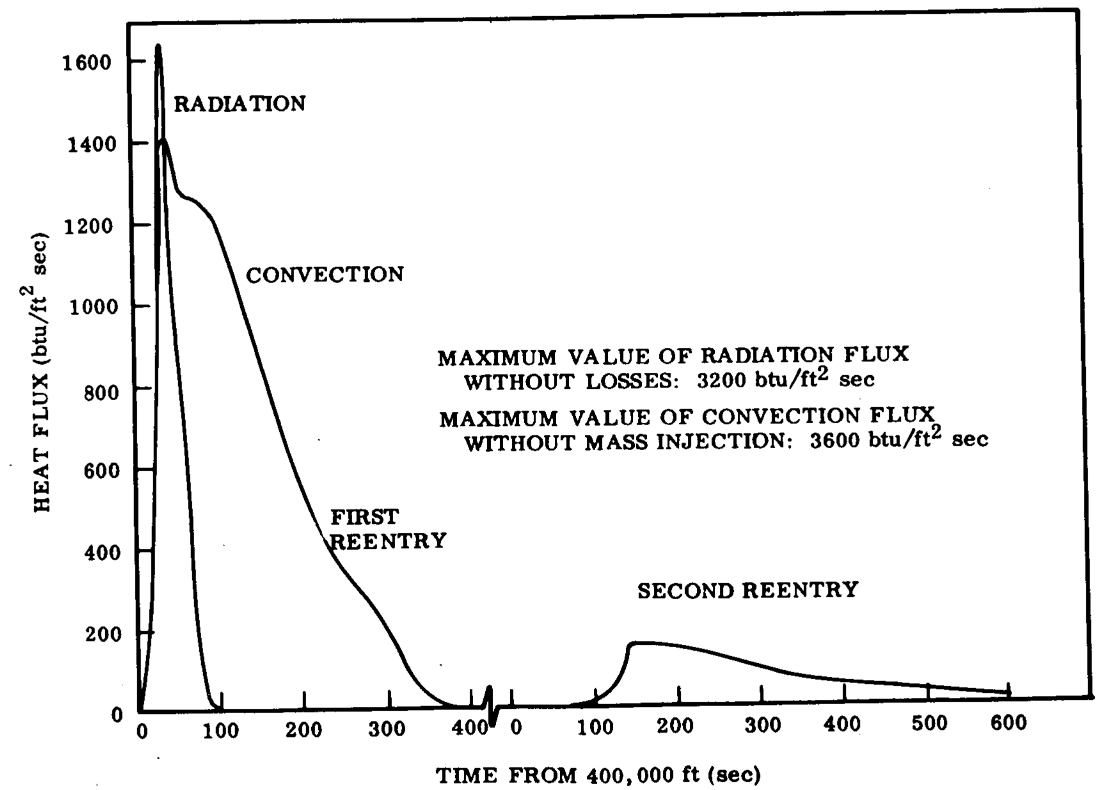


Fig. 3-6 Heating History of Cone 10 ft from Vehicle Tip

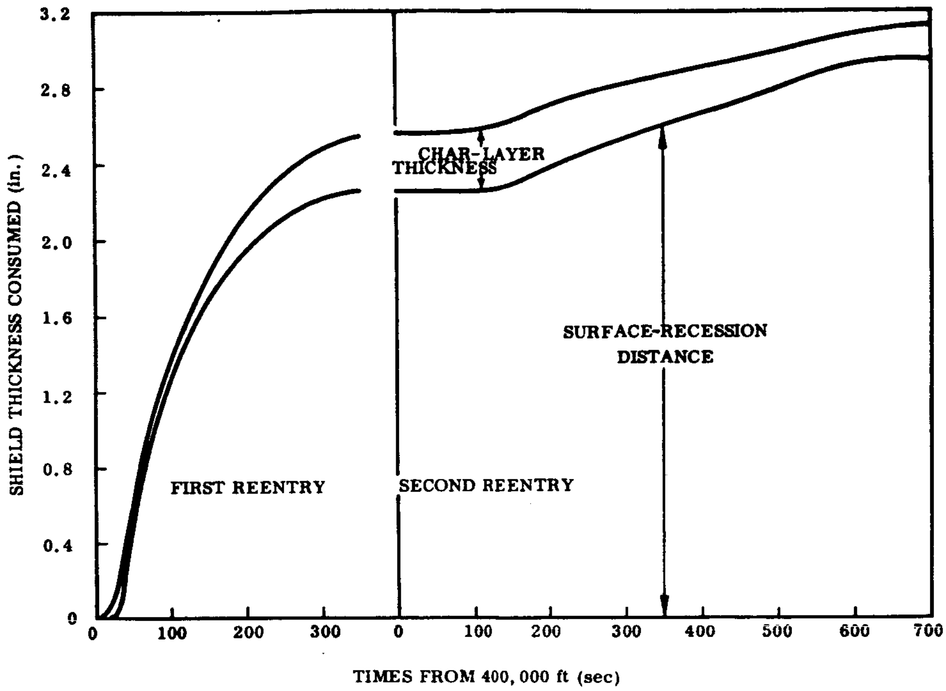


Fig. 3-7 Nylon Phenolic Ablation History - Surface Station: 10 ft

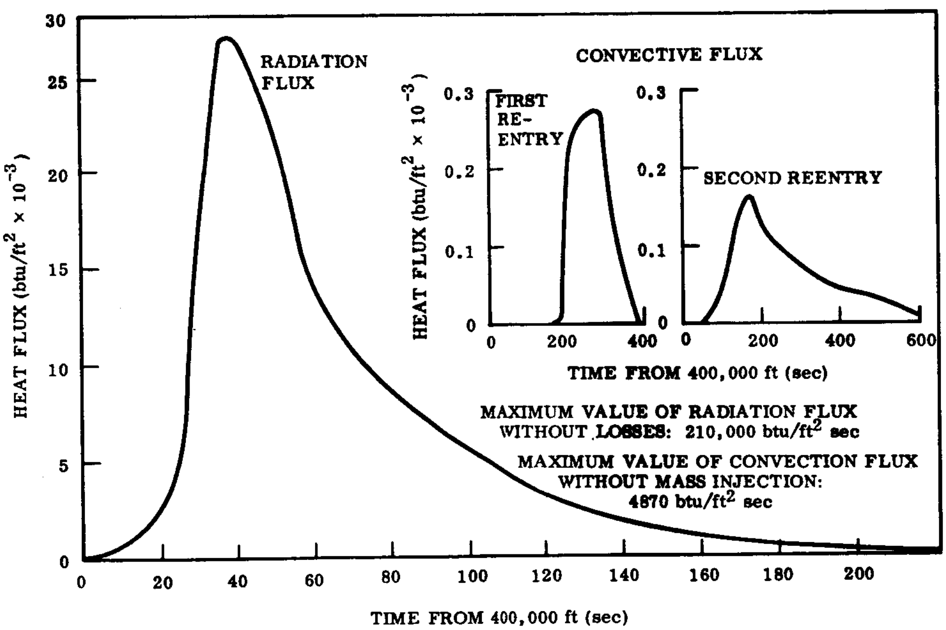


Fig. 3-8 Stagnation-Point Heating History

For the design case studied: L/D = 0.6, W/C_LA = 340 lb/ft^2, base area 131 ft^2, initial nose radius 3 in., the total heat shield weight is 5,540 lb. This is approximately one third of the total vehicle weight. Shield thickness at the nose is 15 in., and the final nose radius is about 1 ft. Over the conical section, an ablator thickness of 3.65 in. is required. (The local requirements vary somewhat, but for simplicity of manufacture, a uniform thickness is proposed). This heat-shield design would result in relatively low substructure temperatures – reaching about 500 to 550° F towards the end of a skip trajectory.

3.7 MINIMUM-WEIGHT CONFIGURATIONS

Total heat transfer to the vehicle is a strong function of vehicle geometry. In particular, the influence of the cone angle is particularly strong. This effect is shown in Fig. 3-9 for a vehicle with L/D = 0.6.

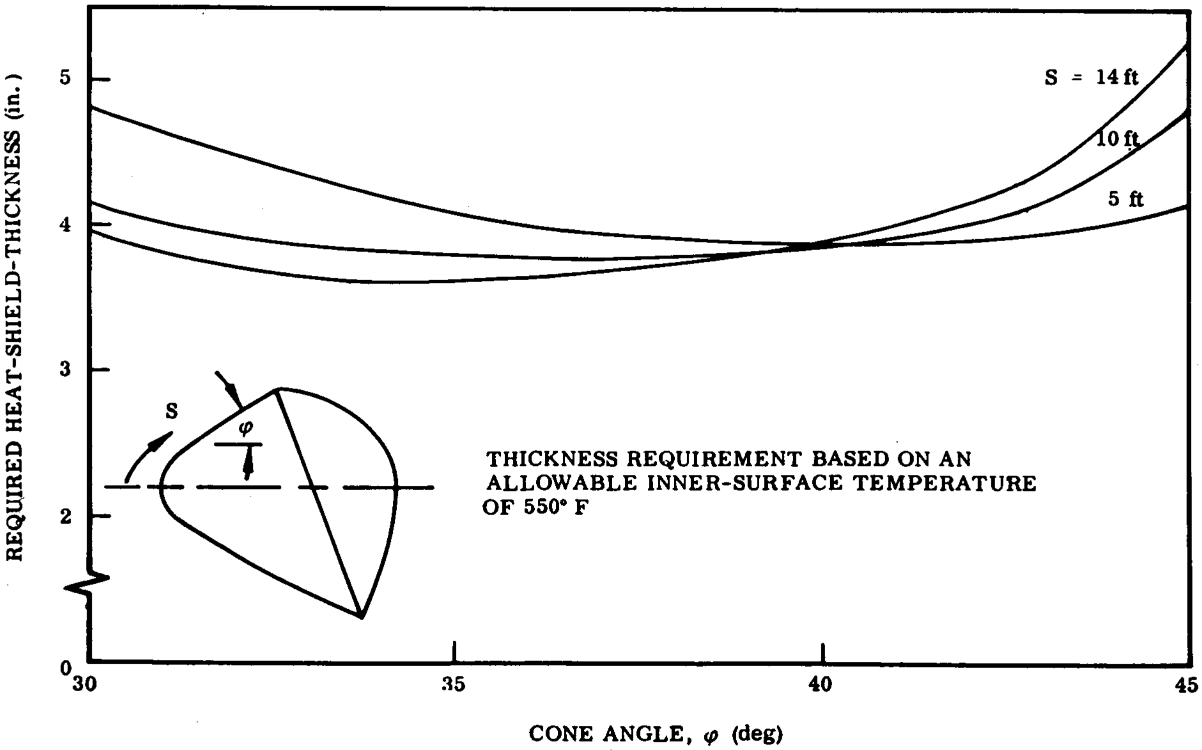


Fig. 3-9 Heat-Shield Thickness Requirements as a Function of Cone Angle

An analysis of the influence of vehicle bluntness on total vehicle weight must consider variations in surface area, structural weight factors, and trajectory (due to changes in $W/C_L A$). Results are presented in Fig. 3-10 for a fixed subsystem weight plus contingency of 7,737 lb (based on the design case) and the same volume (500 ft^3) and base area (131 ft^2). The four curves correspond to different assumptions regarding the heating environment, as is indicated on the figure. The results are for constant-mass trajectories. Such trajectories yield higher ablator weights than the operational trajectory.

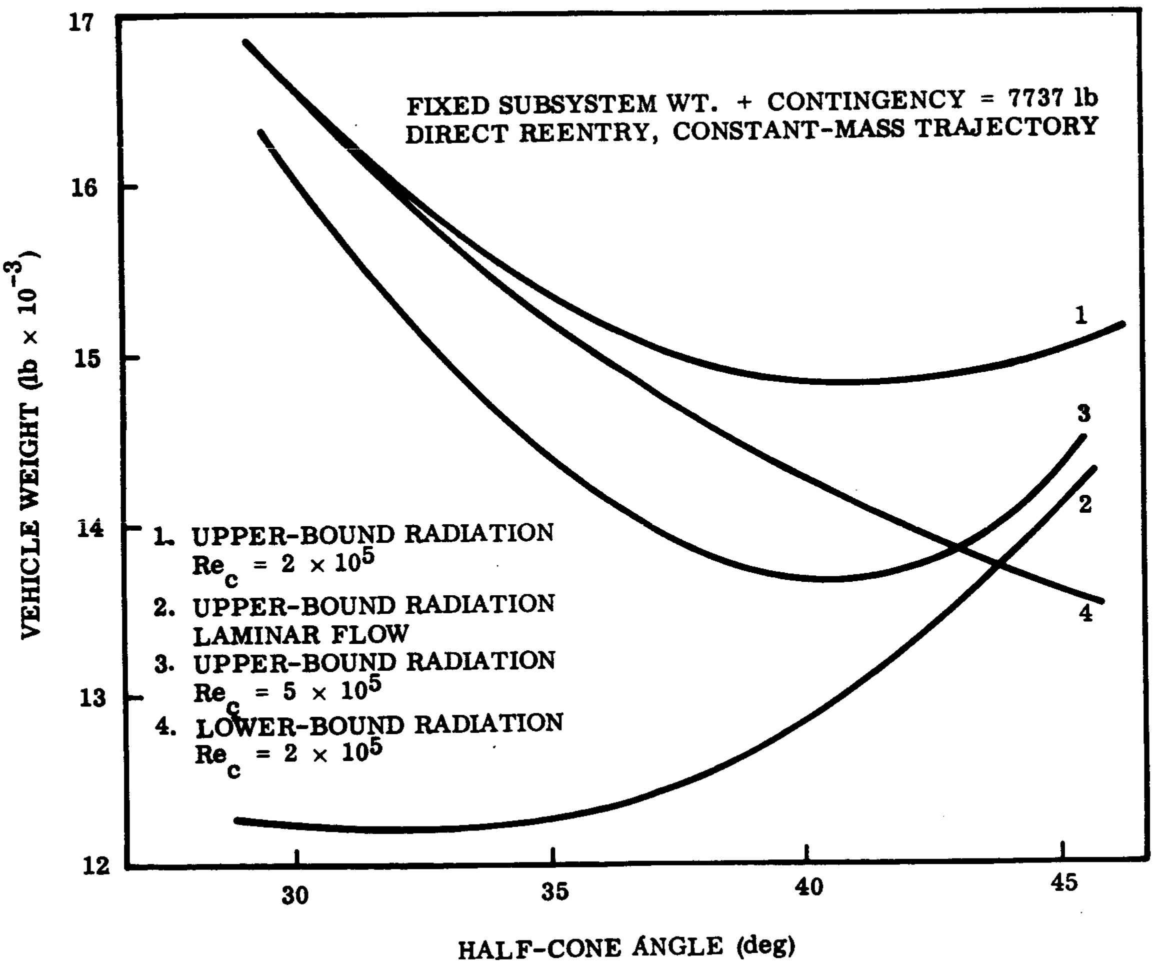


Fig. 3-10 Variation of Vehicle Weight with Cone Angle

The lowest weights are obtained if the boundary layer is laminar; further an optimum half-cone angle near 30 deg is indicated. If the boundary layer is turbulent over most of the vehicle ($Re_c = 2 \times 10^5$), the highest weights are obtained; this curve has a minimum at a half-cone angle of about 40 deg. The lower-bound radiation assumption moves the optimum half-cone angle to some value greater than 45 deg. Generally, increase in weight to the right of the minima is the result of increased radiative heat transfer, whereas the increase to the left of the minima reflects larger forecone areas and higher convective heating resulting from the lower pull-out altitudes associated with the higher $W/C_L A$ values. At this stage, the values indicated by Curve 1 of Fig. 3-10 are considered the most realistic. The influence of L/D on heat-shield requirements is not so great; for a half-cone angle of 35 deg the heat shield weight for the L/D = 1.0 vehicle is about 100 lb less than for the L/D = 0.6 configuration.

Section 4
STRUCTURAL ANALYSES

The main factors considered in the structural design of the vehicle were high reliability, minimum weight, and the use of state-of-the-art materials and construction techniques. The ablator was considered nonstructural; the substructure carried the main loads. Design criteria for the study were established early in the program and were submitted to NASA for approval. These criteria included such considerations as limit-load factors of safety, cabin-pressure limits, and ground-handling limitations.

4.1 LOADS ANALYSIS

Loads analyses were performed for the design configuration under conditions of reentry, abort, and landing. It was shown that reentry loads were not excessive, and that abort represents the most critical condition. Ground impact can also be critical if adverse landing conditions are assumed. Maximum limit stagnation pressure loads on the forecone during reentry and worst abort conditions are 22 psi and 40 psi respectively. For the maximum dynamic pressure abort condition, the maximum limit stagnation pressure load on the aft cone is 22 psi. A factor of safety of 1.4 was applied to these values in the design considerations. The maximum deceleration on the structure occurs during extra-atmospheric abort and amounts to 20 g.

The reentry design environment includes the effects of aerodynamic loading and heating. In Fig. 4.1, the forecone shell structure temperature and aerodynamic pressure are plotted as functions of time from reentry. It can be seen that the maximum pressure occurs when the substructure temperature is low; this means that the critical shell design is not affected by the heating environment. However, at second reentry, a second design condition exists, since the internal pressure and temperature are relatively high. Therefore, the design condition for internal pressure is based on the maximum shell temperature.

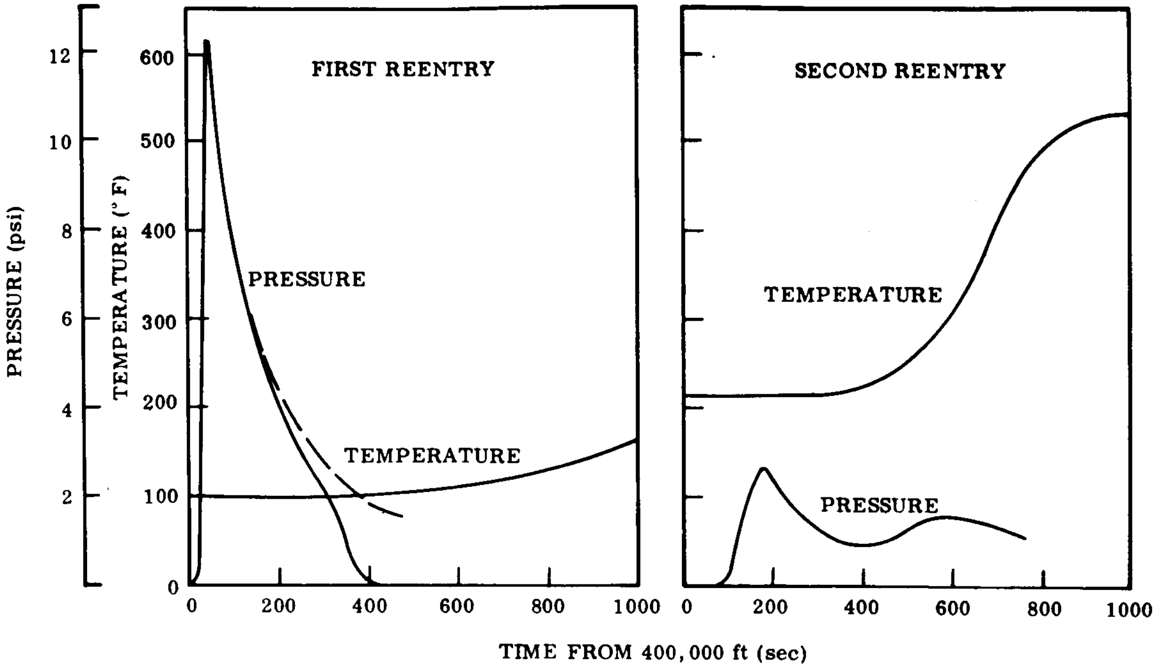


Fig. 4-1 Temperature-Load Interaction During Atmospheric Reentry

4.2 SHELL-STRUCTURE OPTIMIZATION

A choice between a number of available shell constructions must be made. The efficiency of various constructions was compared by means of an "equivalent cylinder" analysis, as shown in Fig. 4-2. The weight parameter (t/R, effective thickness/radius) is plotted against the loading parameter (P_{CR}/E, critical pressure/modulus). A honeycomb sandwich construction was chosen from the results of the comparison.

Of the state-of-the-art materials available, stainless steel was shown to be the most promising for the MMERM application. Aluminum shows a weight advantage below 450 °F, but the superior strength of steel at high temperature is advantageous and adds a margin of safety in a high-temperature environment.

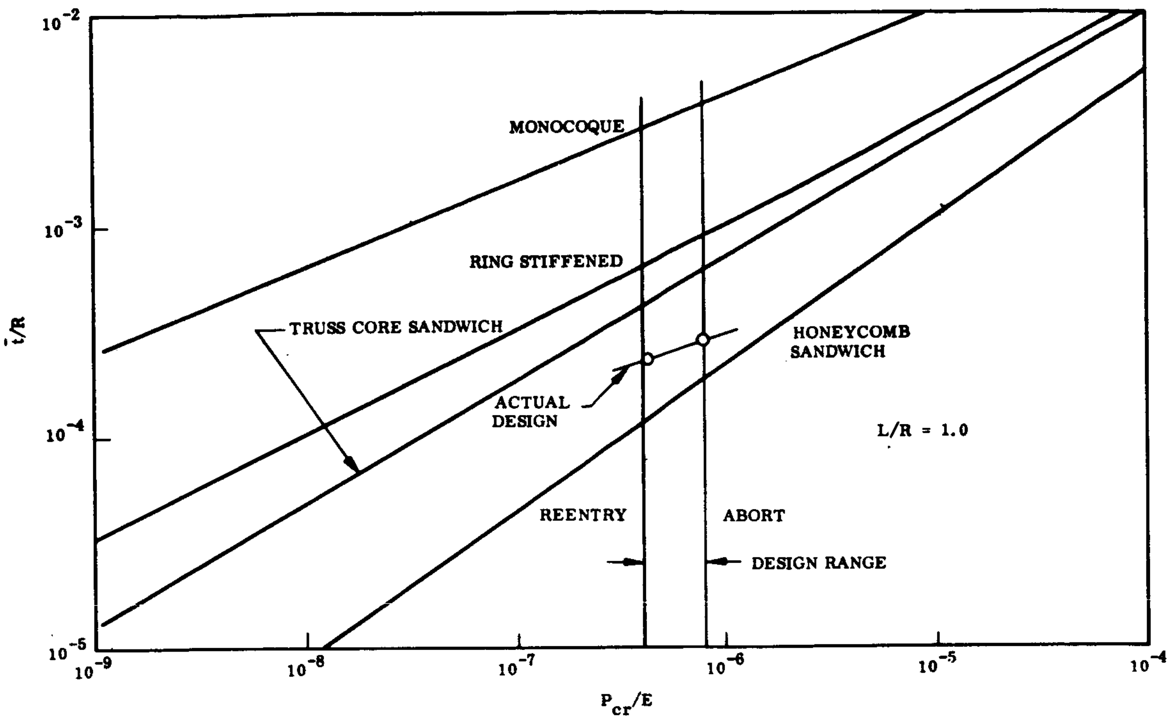


Fig. 4-2 Relative Efficiency of Cylindrical Shells Subjected to External Pressure

The possibility of an optimum maximum operating temperature for minimum structure (plus cooling system) weight was investigated. The results are shown in Fig. 4-3. It can be seen that the ablator weight is a controlling factor and that the minimum structure weight would be obtained with a substructure temperature in excess of 700 °F. However, adequate bond materials (ablator to substructure) are not available for temperatures in excess of about 550 °F. This temperature was, therefore, selected as the maximum operating value.

Consideration of the foregoing results and the two possible failure modes of the sandwich shell — general instability buckling and localized buckling of the facings and/or core — led to the shell design summarized in Fig. 4-4. Abort requirements impose a weight penalty on the shell structure. The increase in weight of the shell is about 25 percent or approximately 250 lb, but this is quite small when considered in relation to the total vehicle weight.

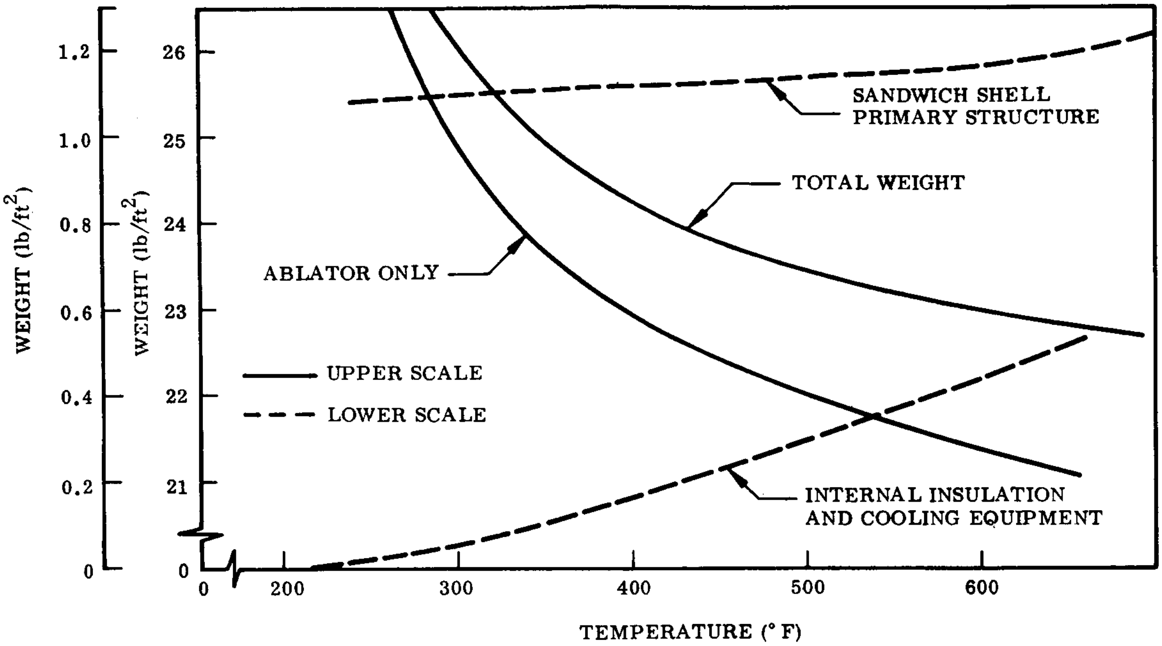
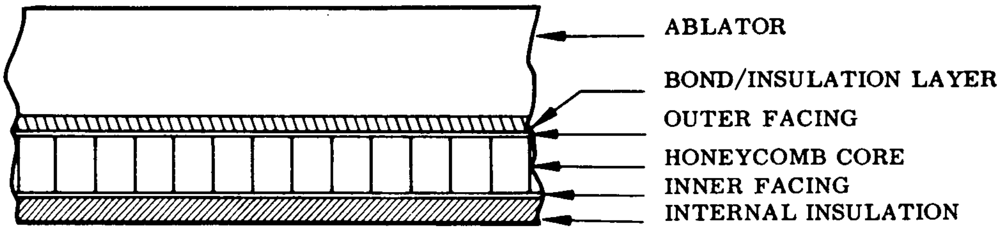


Fig. 4-3 Substructure Temperature Optimization (Forward Shell)



SUMMARY SHELL DIMENSIONS

	Forward Shell		Aft Shell	
	Reentry	Abort	Reentry	Abort
Outer-Facing Thickness (in.)	0.008	0.010	0.010	0.013
Inner-Facing Thickness (in.)	0.010	0.010	0.015	0.015
Core Thickness (in.)	1.60	2.00	1.00	2.00
Weight Sandwich (lb/ft^2)	1.18	1.45	1.32	1.66
Weight Ablator (lb/ft^2)	21.7	21.7	6.20	6.20
Weight Insulation (lb/ft^2)	0.25	0.25	0.25	0.25
Total Weight (lb/ft^2)	23.13	23.40	7.77	8.11

Fig. 4-4 Typical Cross Section of the Shell Structure

The honeycomb core sandwich shell was designed to support the maximum external loads. However, the ablator on the forecone has considerable strength because of its massive proportions. A conservative approximation of the composite shell strength may be made by simple addition of the strengths of individual layers. This procedure showed that the critical buckling pressure for the ablator was high during peak load and that the actual strength of the composite structure provides a large margin of safety for the reentry and abort conditions (a factor of about 3).

The attachment of the ablator to the substructure probably represents the most difficult structural problem related to the heat-shield design. A large differential expansion between the ablator and the steel substructure is possible during heating or cooling. A number of attachment methods were investigated such as flexible supports, segmented ablator, mechanical fasteners, and structural composites. A thick-bond technique is suggested, since a bond layer of finite thickness deforms enough to reduce the internal sheer stresses.

4.3 THERMAL STRESSES IN ABLATOR

The nylon-phenolic material is a very efficient ablator; consequently thermal gradients in the substructure remain relatively low. In the outer layer of the ablator, however, thermal gradients are quite high, but since the ablator tends to flow plastically at high temperature, the virgin material is virtually unaffected. The maximum tensile strength σ developed in the ablator (near the substructure) is shown in Fig. 4-5. It can be seen that the stress is low compared to the ultimate tensile strength (F_{tu}). The predicted stress is conservative, since the restraint of the substructure was neglected. Thermal stresses at the edges of the shell may be higher than those shown, but may be reduced to reasonable levels by a suitable edge-attachment design.

4.4 LANDING-SYSTEM ANALYSIS

A nose-landing mode using crushable structure for energy dissipation is considered the most efficient landing system for the MMERM vehicle. Stroke length, response

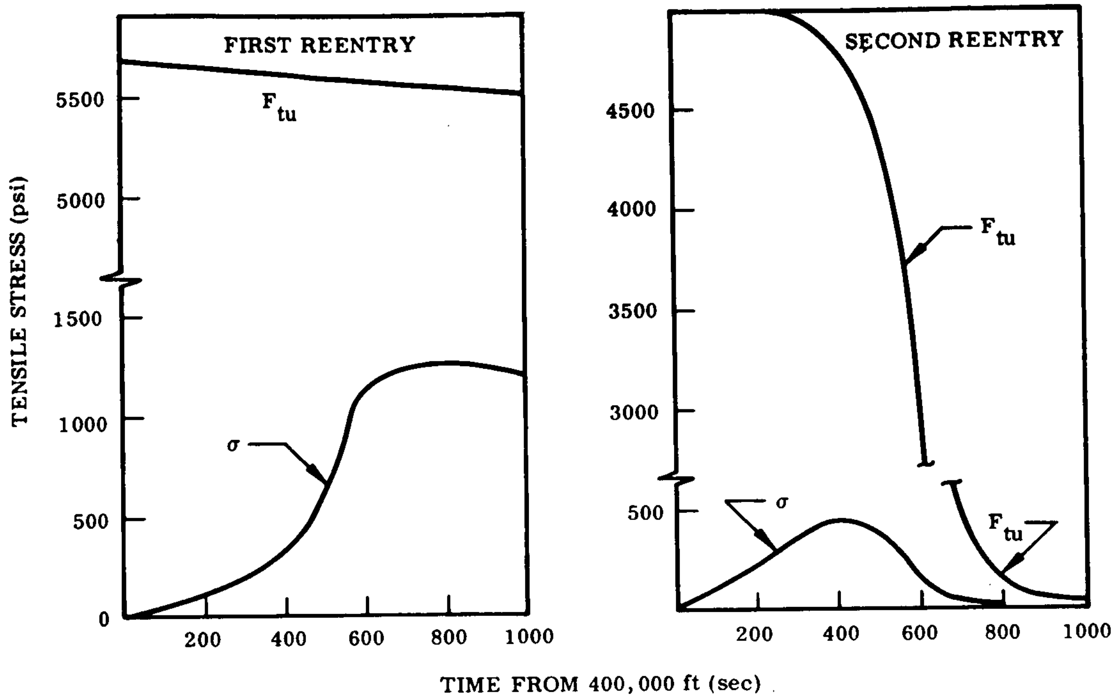


Fig. 4-5 Maximum Tensile Stress in Ablator Due to Thermal Gradient Through the Thickness

characteristics of the crushable material, and terrain conditions are the major vari ables. Figure 4-6 illustrates the maximum normal decelerations (on the structure) that can result from various extreme conditions. The effect of ground slope is also shown in the figure. Curves 1 through 4 represent decreasing efficiency of the crushable material (deceleration-stroke responses that have the following forms: (1) step function (2) concave to stroke-axis (3) linear of slope unity and (4) convex to stroke-axis). A stroke of 18 in. is available in the design. Assuming that a response of the Type (2) is feasible, normal landing conditions would result in accelerations of less than 20 g. For more adverse conditions, the values can be considerably higher. Thus the landing mode, depending on the severity of the conditions, can be a governing design criterion. Further work is required in this area, but for the purposes of the study, it was assumed that suitable techniques will become available from the Apollo program.

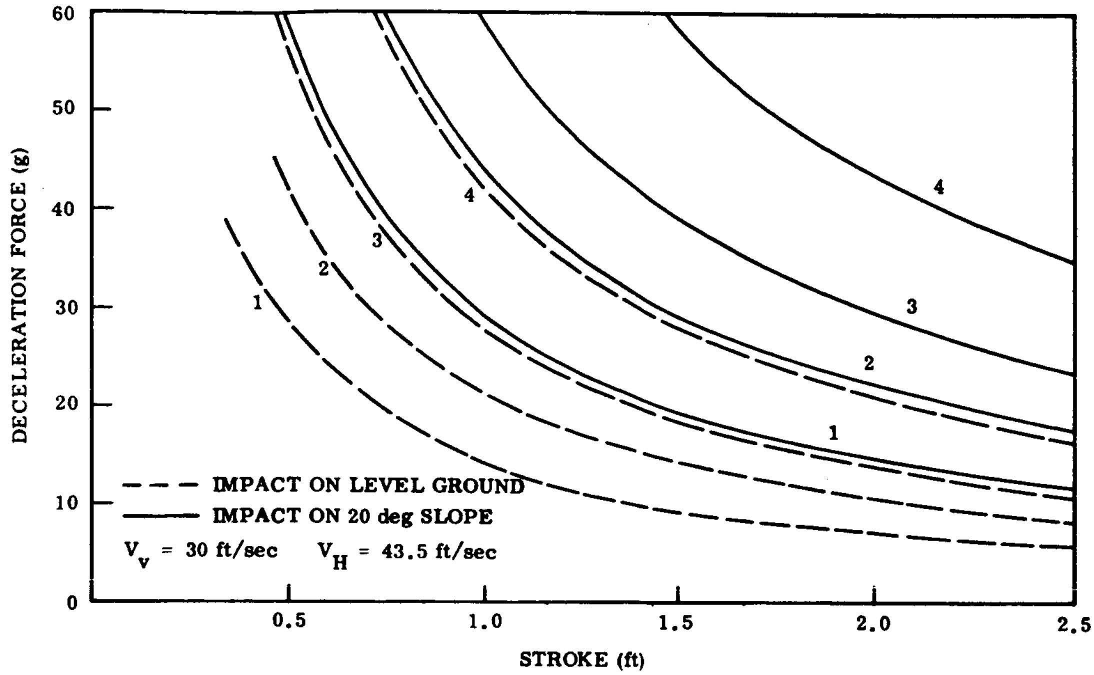


Fig. 4-6 Effect of Stroke on Maximum Deceleration for Nose Landing

4.5 SPACE-ENVIRONMENT EFFECTS ON MATERIALS

Deterioration of the nylon-phenolic ablator may occur from the combined effects of heating and cooling, vacuum, meteoroids, and penetrating radiation. Since the vehicle is stowed within the MMM, most of these effects will be negligible. However, the heat-shield temperature must be controlled within the range -50 °F to 150 °F.

Section 5
SUBSYSTEM DESIGN

The Earth Reentry Module must be selfsufficient in terms of equipment and supplies required to consumate the final phase of the Mars mission. The purpose of the subsystem design study was to interpret the main subsystem requirements (excluding structure) into realistic state-of-the-art hardware. Subsystems studied included navigation and guidance, attitude control, acceleration protection, thermal control, life support, communications, and power supplies. Operational precedures were developed, and utilization of the module during the mission was defined.

5.1 NAVIGATION AND GUIDANCE

Guidance-system concepts have been developed for the Earth-approach phase preceding separation from the MMM and the actual reentry phase.

5.1.1 Earth-Approach Phase

Navigation-measurement techniques proposed for this phase include the use of angles between points on the Earth, the Moon, and stellar references. Successive positions along the trajectory can then be determined. Range would be determined by stadiametric methods, and an accurate clock would be carried on board. A completely redundant set of data is available from Earth-based tracking stations. Navigation-system measuring accuracies are summarized in Table 5-1.

Accuracy requirements imposed by the available reentry corridor width are presented in Fig. 5-1. The model used in deriving these results is described in Ref. 7. The rate of rotation of the local vertical is determined by successive observations of the planet against the celestial sphere. Velocity corrections are then determined by comparing

Table 5-1

NAVIGATION SYSTEM MEASURING ACCURACIES

Measurement Source	Accuracy (m. radian)	(arc-sec)
Field Scanning Telescope (Scan Tube)		
R ≥ 30,000 mi	0.1	20
Horizon Scanner		
R - 10,000 mi	0.2	40
5,000 mi	0.5	100
Star Tracker, Gimbaled*	0.05	10
Sextant, Manual Operation	0.05	10
Radar Altimeter		
(Altitude < 1,000 mi)	0.2 – 0.3% of Measured Altitude	
Doppler Transponder (With DSIF)	0.2m/sec (rms)	
Ranging Transponder (With DSIF)	0.1 μsec, Round Trip (rms)	
Ground Radar Antenna		
Azimuth and Elevation (DSIF)	0.35 m. radian, 70 arc-sec (rms)	

*Similar instrument may be used with earth-based lasers.

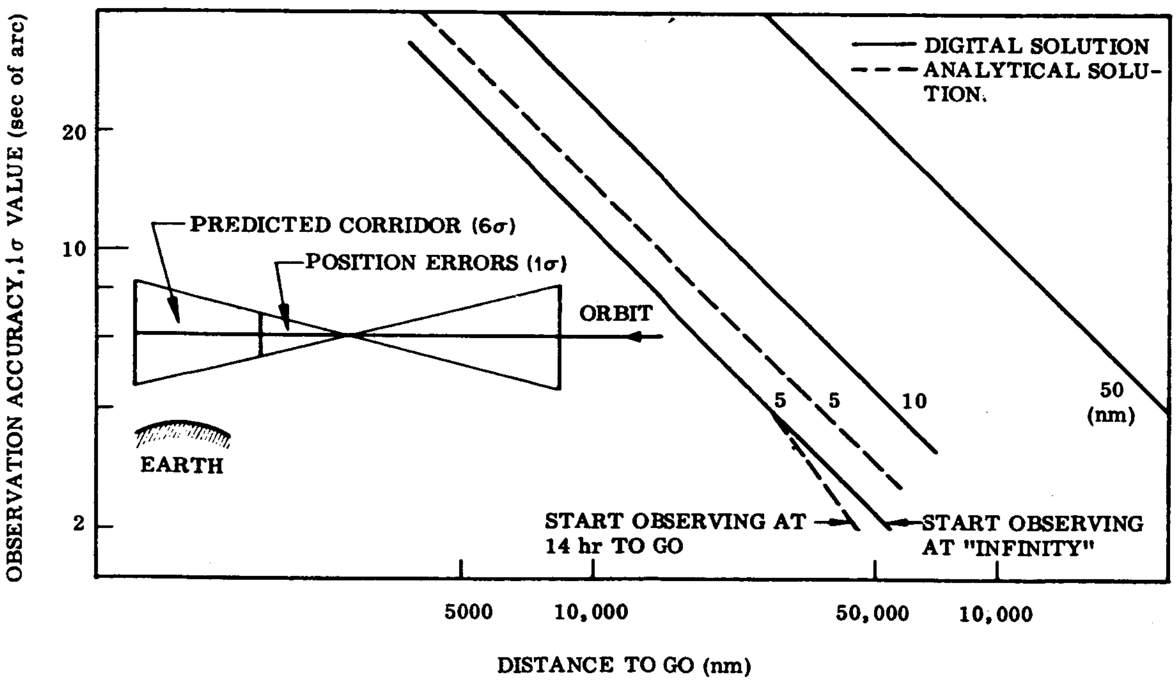


Fig. 5-1 Accuracy Requirements for Earth Reentry

the actual rate with a precomputed nominal rate. The error curves were obtained by assuming two position measurements along an approach trajectory having a perigee velocity of 65,000 ft/sec. Range error is taken to be $R \times 10^{-5}$, and the normal error as $R\sigma$ (σ is the instrument standard deviation). Perigee altitude error is determined by linearly propagating the position error along the trajectory.

An instrument uncertainty of 25 arc-sec was considered as reasonable, if optimistic; thus Fig. 5-1 indicates that a final trajectory correction 18 min before reentry (range about 10,000 nm) will give a 3σ probability of achieving a 10 nm corridor.

When perturbations of the corridor (atmospheric and aerodynamic uncertainties) and guidance-mechanism errors are taken into account, it is concluded that the approach-guidance capability for the 65,000 ft/sec reentry (corridor with 9 nm for $L/D - 0.6$) is still marginal with state-of-the-art on-board equipment. Full use of the DSIF system would improve the capability considerably.

5.1.2 Reentry Phase

During the post-separation vacuum phase, no propulsion is available for course corrections. Navigation consists of determining the local vertical for attitude reference and flight-path angle determination (using the radar altimeter) so that the required roll angle can be established. Both the on-board computer and the stellar-inertial guidance system (which are erected before separation) are used. During atmospheric reentry, navigation will be performed by the inertial instruments and computer. A closed-loop system is required, and a fast predictive technique that uses measured state variables (such as acceleration history) is necessary to maintain corridor, particularly at the higher speeds. The essential components of the module guidance system are shown in Fig. 5-2.

5.2 ATTITUDE-CONTROL SYSTEM

The attitude-control system has two functions. During the vacuum phases, the reentry vehicle must be positioned so that the various sensors can acquire their targets; also

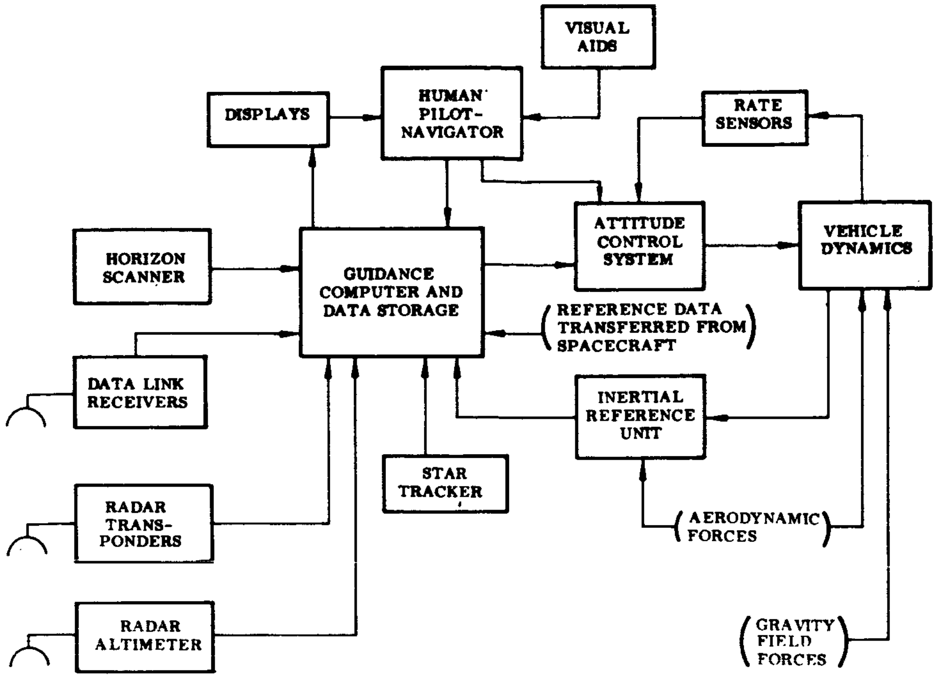


Fig. 5-2 Module Guidance System

just before reentry, the correct vehicle attitude must be attained. For atmospheric flight, the direction of the lift vector is controlled by rolling the vehicle about the velocity vector. A movable mass will be used to correct excursions in pitch and yaw, with the reaction control system providing refinement and damping.

The reaction jets are powered by a bi-propellant system consisting of a N_2O_4-NO/MMH fuel combination. These fuels are storable and give minimum system weight. The jets are located on the aft cone as far forward as possible without causing flow interference on the forecone. The attitude-control system (shown in Fig. 5-3) employs pulse modulation; it consists of six combustion chambers and ablative-cooled nozzles fed by a single propellant tank.

Elements required for the control system synthesis are shown in Fig. 5-4. The total propulsion system weight is approximately 1,000 lb; 500 lb of fluid are required for C.G. control.

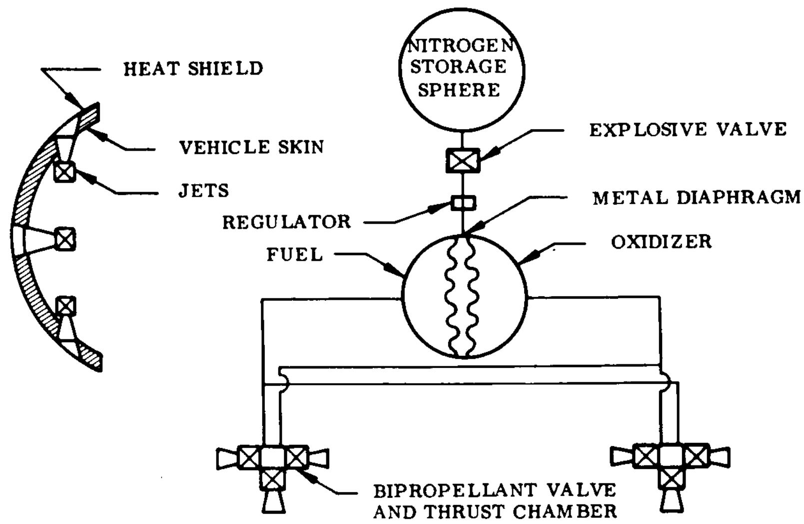


Fig. 5-3 Attitude-Control System

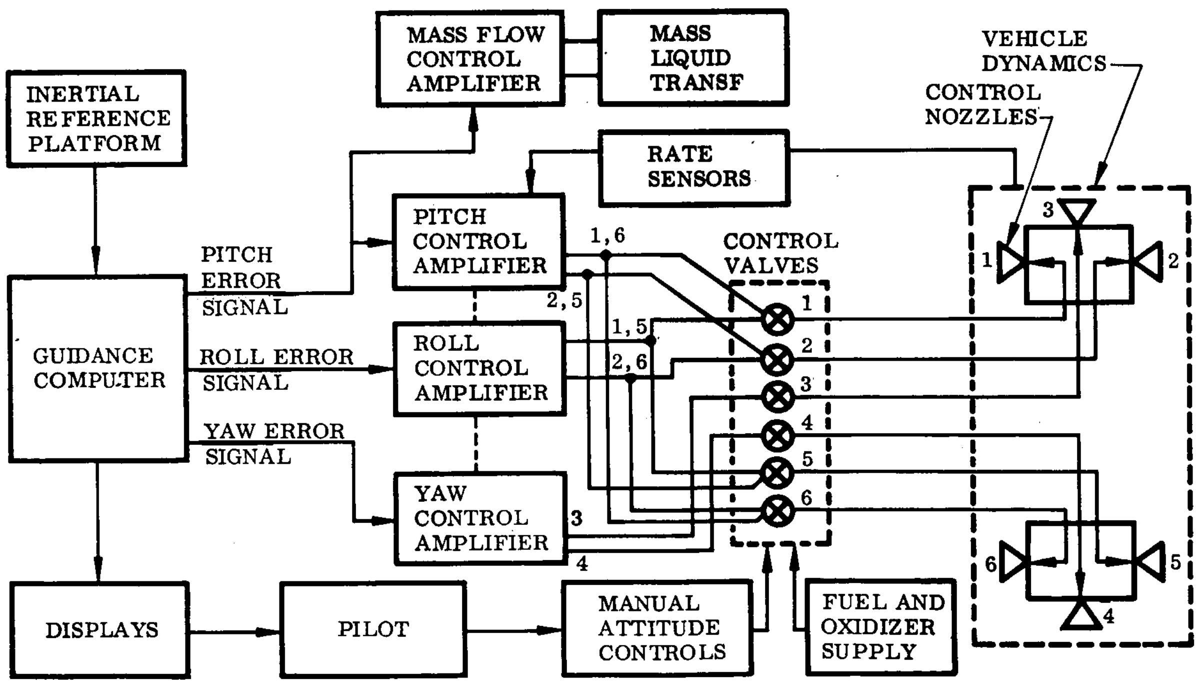


Fig. 5-4 Attitude-Control System Elements

5.3 ACCELERATION PROTECTION

The acceleration-protection system must be effective under reentry and impact conditions.

A typical acceleration profile for direct reentry is illustrated in Fig. 5-5. When a skip trajectory is followed, a further, small, G peak will be encountered at second reentry. The profile is considered tolerable with use of the contoured couch and restraint system proposed in the study; however, experimental verification is required. The couches contain built-in shock absorbers which will be used in conjunction with the crushable nose structure to alleviate impact decelerations.

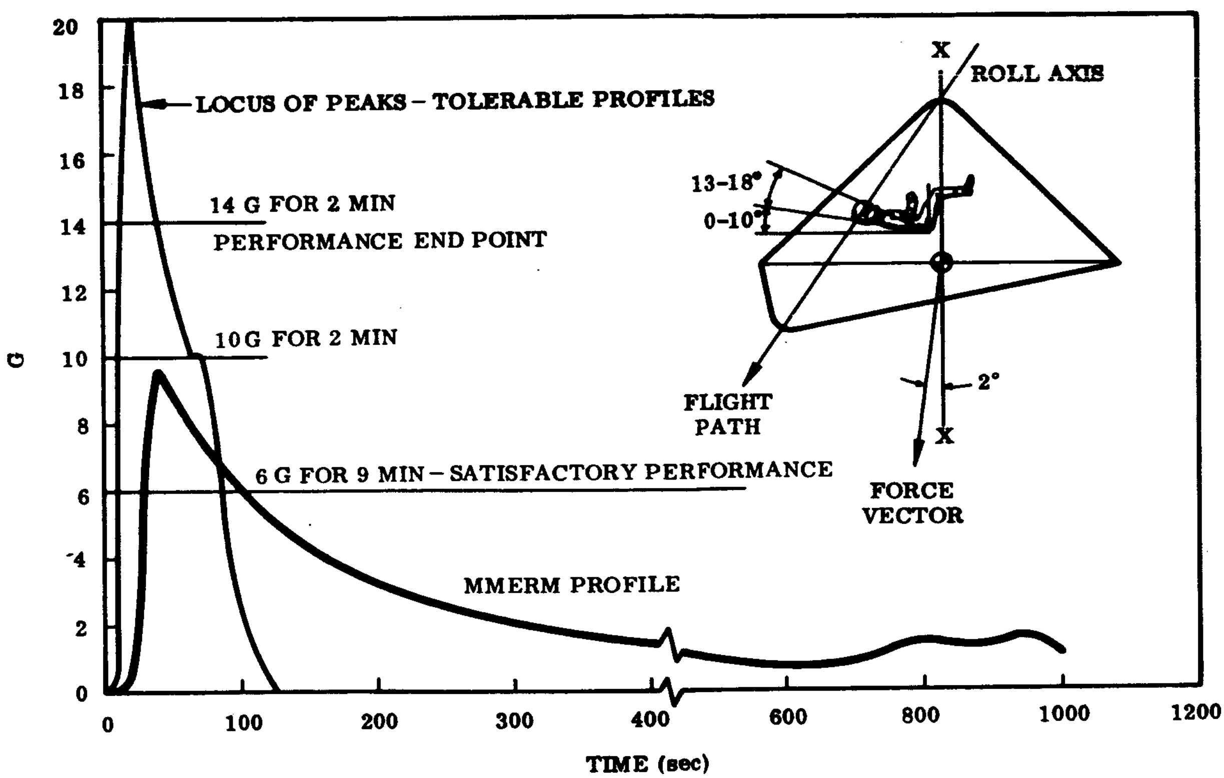


Fig. 5-5 Acceleration Profile (Direct Reentry)

5.4 THERMAL CONTROL

A detailed thermal analysis of the internal equipment and crew during reentry and recovery was conducted by means of the model shown in Fig. 5-6. The results showed that the proposed cooling technique was adequate for the thermal environment. To 90,000-ft altitude, this technique used oxygen flow through the suit in conjunction with a water evaporator. Oxygen without a heat exchanger was then used until at about 25,000 ft, atmospheric air could be passed through the suit. After impact, the capsule doors can be opened. Limiting the deep-body temperature rise to 1°F enabled the amount of internal insulation required to be determined. For the operational conditions, the amount was about 0.75 in. The results were also used in the structural optimisation studies and in defining the thermal-control system design.

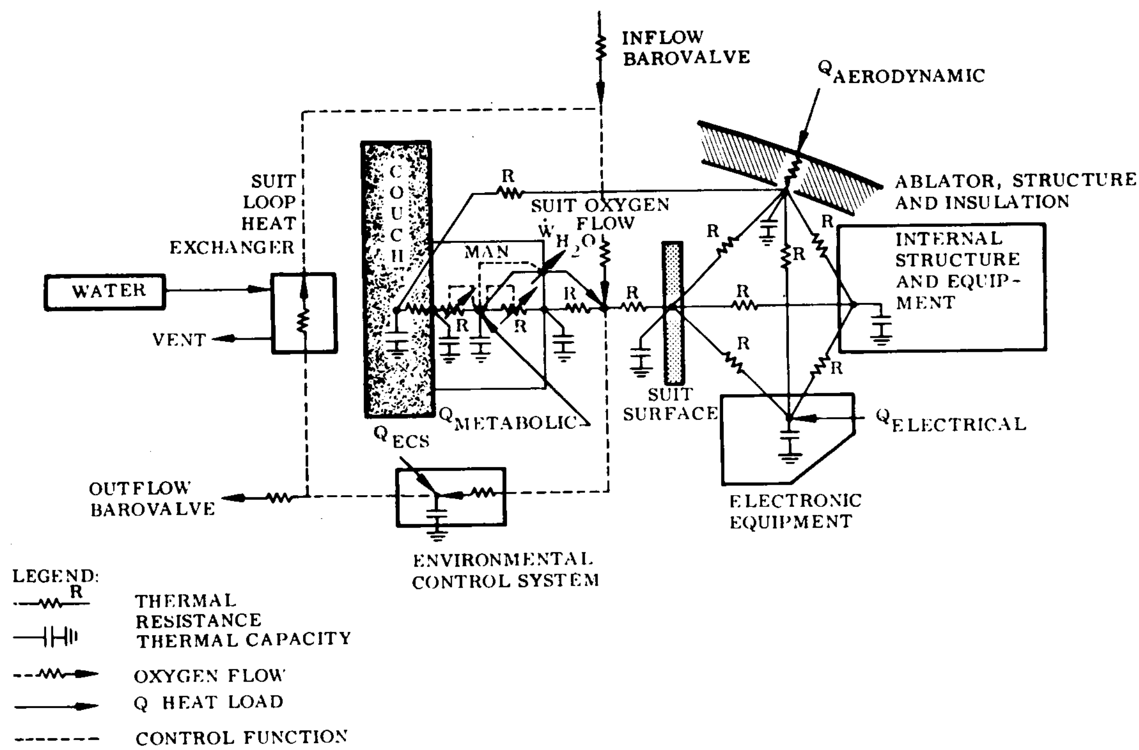


Fig. 5-6 Thermal Analysis Model

5.5 LIFE-SUPPORT SYSTEM

Because the reentry module is occupied only a short time, (check out and reentry times amount to about 8 hours), well-tried, existing technology can be used as a basis to estimate life support and environmental-control requirements. These are summarized in Table 5-2.

Table 5-2

LIFE-SUPPORT SYSTEM SUMMARY

Item	Weight (lb)	Volume (ft^3)	Power (watts)
Oxygen Supply	189 (44)*	3.3	0
CO_2 Removal	32 (11)*	1.4	0
Air Circulation and Thermal Control	147	8.0	720
Waste Collection and Storage	13	0.5	0
Seat and Restraint System	150	-	0
Pressure Suits	180	-	0
Food Supply and Storage	48	1.6	0
Water Supply and Storage	277 (226)*	5.6	0
Personal Hygiene	8	0.3	0
Survival Kit	150	7.5	0
Total	1,194 (977)*	28.2	720

*Weight at reentry if expended materials are left in mission module.

Note: Estimates are based on "Life-Support-System Usage During Mission."

5.6 COMMUNICATIONS

The communication system shown in Fig. 5-7 includes VHF and S-Band transceivers, a H.F. voice link, and a recovery beacon. The S-Band system is used in conjunction with the DSIF network. Whip antennas are required for all these systems; the antennas must be withdrawn during the reentry heating period. The radar altimeter employs a horn-type antenna deployed on the inside surface of the cabin.

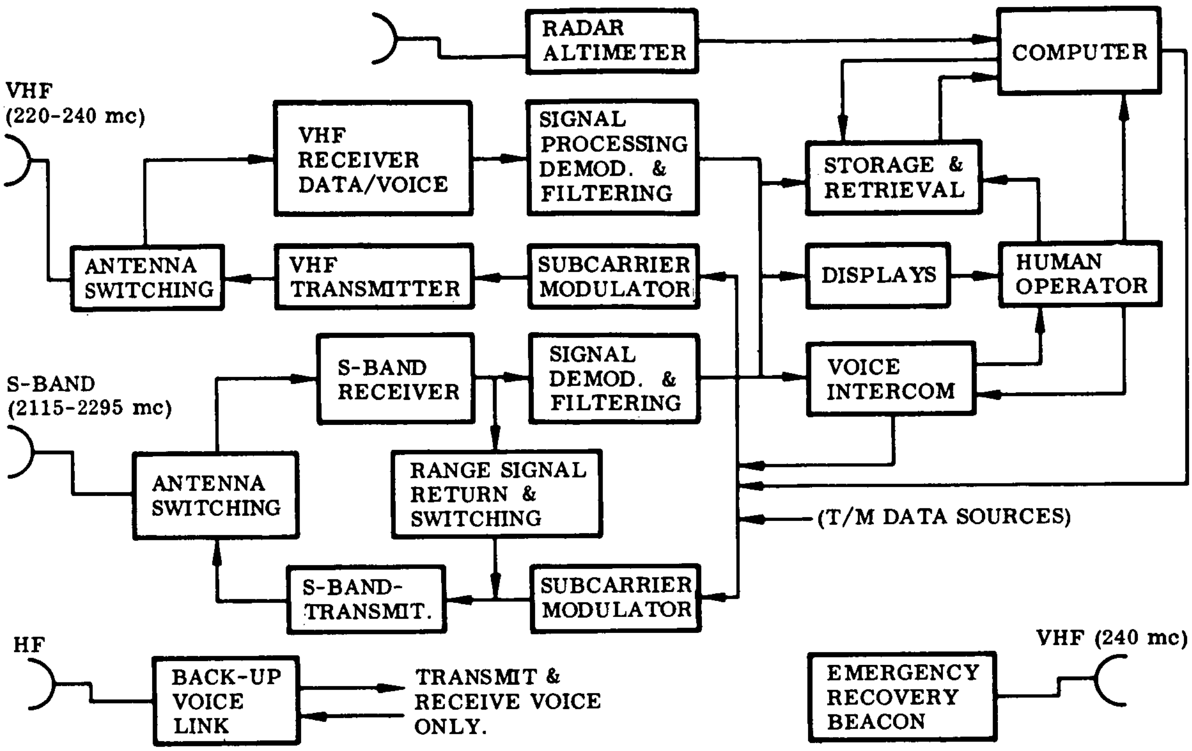


Fig. 5-7 Communications System

5.7 ELECTRICAL POWER SYSTEM

Fuel cells have been selected as the primary power source to satisfy a peak-power requirement of 1.8 kw and an average power demand of 1.4 kw during the MMERM operating phase of 10 to 12 hours. Two complete fuel-cell systems will be used to improve reliability. The cells are operated simultaneously with one cell "idling" in a standby condition. Fuel cells were chosen for restart capability and low weight penalty. Since the cells run hot, a cooling system will be required.

Section 6
MODULE-DESIGN INTEGRATION

Engineering design of the reentry module appears to present no particular problems. Apart from the inconvenience of special tooling which arises from the unusual shape, fabrication should be straightforward.

Internal-arrangement studies resulted in the layout given in Fig. 6-1 for the L/D = 0.6 vehicle. Because of the large off-center mass of the forecone heat shield, it is necessary to locate most of the heavy equipment at the lower end of the module. This main tains the required C.G. position. Access to the module is through the main hatch just above the display panel. The crew is positioned facing aft so that the acceleration forces act transverse forward (eyeballs in). The two pilots are placed in front of the display panel with controls easily accessible. Two crew members are arranged behind the pilots, and the two emergency crew members would be accommodated on either side

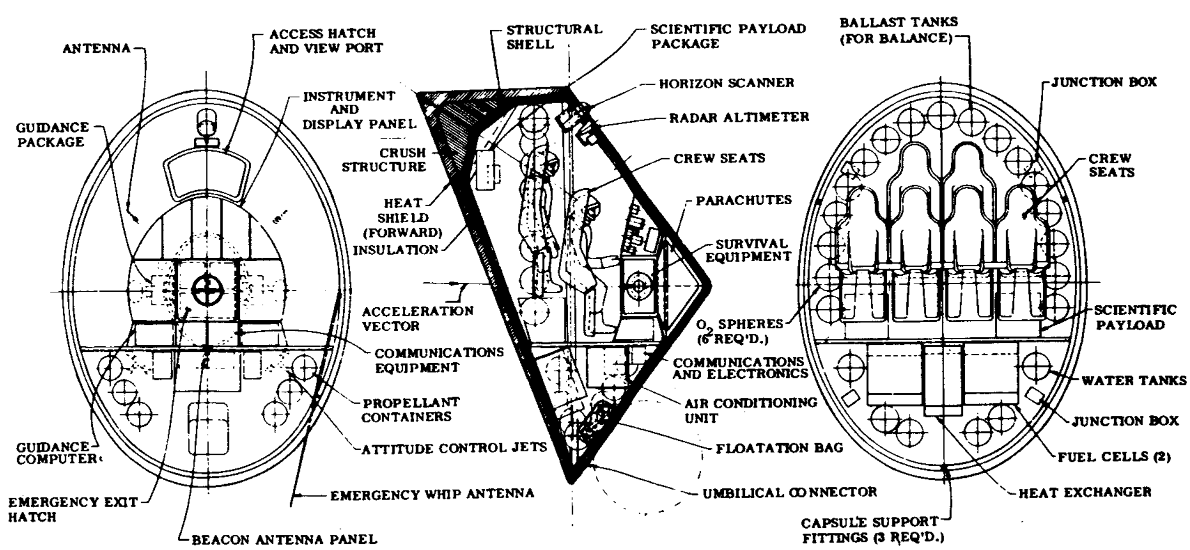


Fig. 6-1 General Assembly (L/D = 0.6)

of the pilots. View ports are provided for crew and instruments; protection during reentry is necessary.

Recovery parachutes are stored in the aft conical tip, which is ejected at chute deployment. Flotation bags are included to ensure good stability in the event of a water landing.

The total volume of the vehicle is 500 ft^3. This was found to be adequate for structure, equipment packaging, and crew arrangement. The dimensions of the elliptical section shown in the figure are: major axis 181 in., minor axis 132 in. The forecone half-angle is the design value of 35 deg.

A weight summary for the design vehicle is given in Table 6-1. The total reentry weight (without contingency) is well within the original target weight. As explained in Section 3.7, a weight reduction of some 500 lb can be expected at the optimum cone angle of 40 deg.

Table 6-1

WEIGHT SUMMARY

(L/D = 0.6, Half Cone Angle = 35 deg)		Weight (lb)
Heat Shield		5,540
Structure		1,723
Recovery System		590
Crew and Effects		1,140
Life Support		977
Controls and Displays		280
Electrical		629
Guidance and Navigation		300
Attitude Control		1,548
Communications, Telemetry and Data Acquisition		185
Scientific Payload		800
Contingency		1,288
	Total	15,000

Section 7
PROGRAM PLAN

The entire program for development of the MMERM vehicle from inception to delivery of a flight ready vehicle would span about eight and one half years. Three major phases are involved: study, program definition, and development. The last phase constitutes the major effort and includes design fabrication and testing as shown in Fig. 7-1.

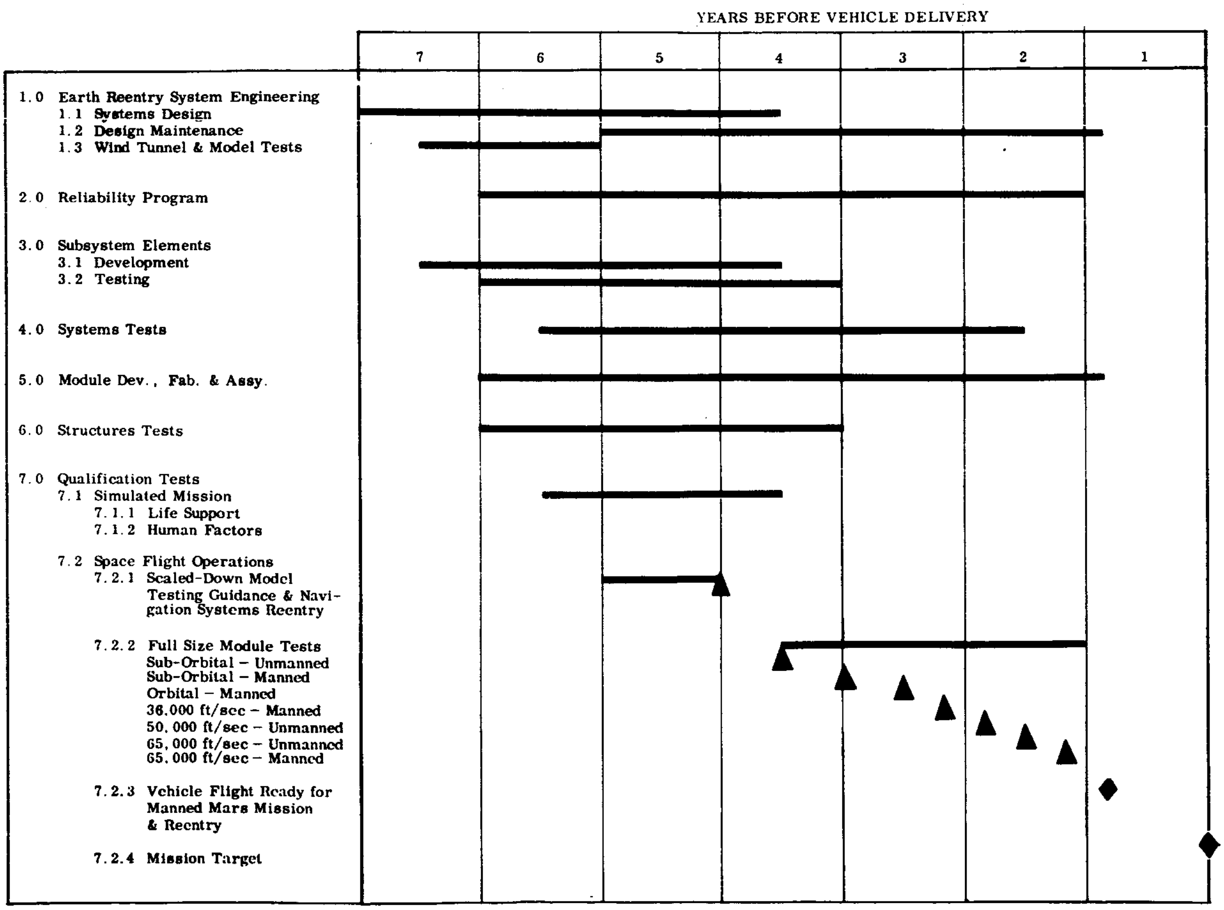


Fig. 7-1 Reentry Module Development

Flight testing of the module will play a vital role. Only a cursory look could be taken at the problems involved, but it was concluded that they are of major importance. The test program stipulated should be regarded as a starting point only, and considerable optimization is necessary. Full-scale testing at 65,000 ft/sec would involve the use of Saturn-class boosters staged in Earth orbit. The cost and complexity of such a task could seriously affect the choice of the mission and its associated reentry speed. Cost estimates based on the assumed flight-test program are given in Table 7-1. The cost includes stand-bys for scheduled flights and should be considered an upper estimate. Booster costs were predicated on numbers of vehicles necessary to achieve a reasonable chance of successfully accomplishing the full-scale, high-velocity tests. Major areas that should be studied in an attempt to reduce costs and optimize the flight-test program are listed in Table 7-2.

Table 7-1

PRELIMINARY COST ANALYSIS

			Dollars $\times 10^{-6}$
First Unit Cost (Airframe)			8
Module R&D			
Flight-Test Hardware			112
Engineering, Test, and System Integration			173
Facilities and Vehicle Launch Operations			101
Boosters and Launch Cost			587
		Total	973
Escalation	3%/yr to 1973		1300
Probable Error	100%		2600
Guidance-Development Program			100
Heat-Shield Development Program			20
		Probable Total Cost $\leq$	2720

Table 7-2

FACTORS AFFECTING DEVELOPMENT COST

MISSION SELECTION	AVAILABILITY OF LAUNCH VEHICLES
Operational Requirements vs. Development Requirements	All-Purpose Engines
	Advanced Propulsion Units
	Modification Complications
USE OF SCALE MODELS	INPUTS FROM EXISTING PROJECTS (APOLLO, ETC.)
Applicability of Results	Experiment Design
Prediction Reliability	Modification to Mission
Model Size	
Ground-Test Facilities	
LOWER-SPEED TESTING	PLANNING IN DEPTH
Critical Speeds for Certain Phenomena	Choice of Time Period and Development Time
Prediction Reliability	Relationship to Overall Space Program
Equivalent Thermal Environment	

Section 8
STUDY CONCLUSIONS AND RECOMMENDATIONS

This study has shown that reentry from a Mars mission at speeds up to 65,000 ft/sec is feasible and that the main limitation is imposed by approach-guidance requirements. The design vehicle (LMSC conical configuration) chosen for the detailed study had the following basic characteristics: $L/D = 0.6$, $W/C_DA = 204$ lb/ft^2, half-cone angle of 35 deg.

8.1 CONCLUSIONS

The principal conclusions relative to this particular vehicle and more general aspects of the study are as follows.

Range Control

This is adequate with moderate lift capability ($L/D = 0.5$ to 1.0).
A skip trajectory is used to increase down-range capabilities by 10,000 nm.

Orbit inclination changes and reentry roll program provide adequate cross-range control.

Corridor Width

For a reentry speed of 65,000 ft/sec the corridor widths for L/D ratios of 0.6 and 1.0 are 9 and 11 nm respectively.

Roll rates should not be less than 40 deg/sec for maintenance of corridor.

Guidance

Approach guidance is marginal at 65,000 ft/sec with existing hardware.

A closed-loop system is required for reentry.

Heat Shield

Ablation is adequate up to 65,000 ft/sec with an efficient ablator.

Heat-shield weight is approximately 1/3 the vehicle weight.

Radiation processes and mass injection effects are important design considerations.

Heat-shield weight is strongly dependent on cone angle.

Structures

State-of-the-art materials are adequate (stainless-steel honeycomb sandwich construction).

Abort and ground impact loads govern design.

Crushable material in nose is required for landing.

Heat-shield storage temperature limits range from -50°F to 150°F.

Attitude Control

A bipropellant reaction system is required (N_2O_4 - NO/MMH).

C.G. control (for trim) is necessary.

Life Support And Thermal Control

Existing techniques are adequate.

Design Integration

Recommended Volume: 500 ft^3.

Minimum weight vehicle (L/D = 0.6): 13,300 lb at half-cone angle of 40 deg.

Development Planning

The flight-test program greatly influences overall program cost.

Tests at 65,000 ft/sec present technical difficulties.

8.2 RECOMMENDATIONS

The need for further work is stressed, and the following main recommendations are made:

Aerodynamics And Performance

Conduct wind-tunnel tests to verify aerodynamic characteristics.

Analyze in detail the vehicle dynamic motion and perform comprehensive stability studies.

Study trajectory variables and measurement techniques for use in navigation procedures and attitude control.

Analyze the afterbody flow field.

Thermal Effects

Investigate all areas of uncertainty, particularly (1) emissivity of air, (2) effect of entropy gradient on boundary-layer development, and (3) transition under ablation conditions.

Study emission and absorption properties of ablation products.

Examine shield design at stagnation point.

Structural Design

Study the influence of unfavorable landing conditions on load-carrying structure.

Investigate the heat-shield attachment problem.

Study the internal support-structure problem.

Experimentally investigate transonic buffeting (including noise levels).

Navigation And Guidance

Analyze in greater detail the approach-guidance requirements.

Improve accuracy and reliability of N and G hardware; back-up modes should be fully investigated.

Evaluate effects of separation (MMM-MMERM) on approach trajectory.

Perform reentry guidance simulation studies.

Study the use of propulsion system on MMERM to improve approach-guidance capability.

Acceleration Stress

Study experimentally complex G-histories.

Evaluate human tolerance to reentry profiles with peaks greater than 10-G (possible increased corridor width).

Development Planning

Study development program in detail.

Evaluate flight-test requirements and develop realistic program.

Include implications of development plan and costs in selection of optimum missions.

Section 9
REFERENCES

1. M. Nardone, R. G. Breene, S. Zeldin, and T. R. Riethof, Radiance of Species in High-Temperature Air, Space Sciences Laboratory, General Electric Co., R63SD3, 1963

2. B. Armstrong, D. Buttrey, L. Sartori, A. Siegert, and J. Weisner, Radiative Properties of High-Temperature Gases, AFSWC-TR-61-72, LMSC 2-34-61-1, Lockheed Missiles & Space Co., Sunnyvale, Calif., 1961

3. B. Kivel and K. Bailey, Tables of Radiation from High Temperature Air, Research Report 21, AVCO, 1957

4. R. E. Meyerott, J. Sokoloff, and R. W. Nicholls, Absorption Coefficients of Air, LMSD 288052, Geophysical Research Paper 58, GRD-TN-60-277, Lockheed Missiles & Cpace Co., 1960

5. J. L. Stewart and K. D. Pyatt, Theoretical Study of Optical Properties – Photon Absorption Coefficients, Opacities and Equations of State of Light Elements Including Effects of Lines, General Atomic, APSWC-TR-61-71, San Diego, Calif., 1961

6. H. Hoshizaki, "Heat Transfer in Planetary Atmospheres at Super Satellite Speeds," ARS Journal, Vol. 32, No. 10, 1962

7. J. V. Breakwell, L. F. Helgostam, and M. A. Krop, "Guidance Phenomena for Mars Missions," AAS Symposium on Exploration of Mars, Denver, Col., 1963

26991

PART 14

A PLANETARY TRANSPORTATION MODEL AND ITS APPLICATION TO SPACE PROGRAM PLANNING

by

G. W. Morgenthaler
Martin Company
Contract No. NAS8-11057

A PLANETARY TRANSPORTATION MODEL AND ITS APPLICATION TO SPACE PROGRAM PLANNING[1]

George W. Morgenthaler[2]

ABSTRACT

Technology has now progressed to the point where the Nation has accomplished several ballistic missile development programs, several manned orbital flights, and has committed itself to Earth orbital rendezvous, to establishment of manned space station laboratories, and to manned lunar exploration. At this time a very great number of alternative space programs including manned and unmanned planetary flights, establishment of extra-terrestial bases, and a large number of modes of operation involving various kinds of interplanetary spacecraft and large Earth-to-Earth-orbit boosters are possible. In the face of this great number of possibilities, it is necessary to develop analytical tools which will enable planners at the highest level to trade-off alternatives and arrive at preferred space programs and perferred technical development plans.

Martin-Denver, under contract NAS 8-11057, entitled "Development of a Basic Planetary Transportation System Model" is engaged in development of a computerized model which accepts as inputs various complete space programs including alternative spacecraft, alternative boosters, and alternative exploration missions. The model will perform quantitative evaluations to assist planners in arriving at valid overall plans.

I. THE NEED FOR A PLANETARY TRANSPORTATION MODEL AND ITS OBJECTIVES

Earlier speakers at this Symposium have presented numerous detailed feasibility studies for particular space vehicle designs to perform a given mission. Dr. von Braun in his opening remarks has indicated the vital role which such

---

[1] The material discussed in this paper was developed under contract NAS 8-11057. Major contributors to the effort were: Dr. R. Novosad, Project Manager, M. Capehart, G. Fosdick, C. Haff and E. Shirley.

[2] Manager, Electronics Research Department, Martin Company, Denver.

studies play in enabling planners to envision future alternatives. Other symposium speakers have presented trade-off studies involving subsystems such as various planetary entry vehicles, and they have shown the influence which the time of performing various missions has on total orbital departure mass, etc.

Though of unquestioned value for providing "realistic" weights, costs, configurations, and facility needs, these studies alone do not completely fill the planner's requirement for the relationship of total space program scope to the total national space resources. In this paper we shall endeavor to indicate what is still missing and shall suggest a detailed model which is designed to synthesize the collection of such individual studies into quantitative measures of the total space program.

Suppose that the points of the plane of the paper in Figure 1 each represent a complete national space program - where we shall travel; when; by what maneuvers of staging, orbit phasing, entry; and which boosters and spacecraft we shall employ for each mission. There are an uncountably infinite number of points on the page - of course, there are an uncountably infinite number of different national space programs which we may elect to perform (we call programs "different" when they differ in missions, maneuvers, vehicles, or times of travel).

Dr. Koelle once sketched a box labeled "feasible space programs." This box contained those space programs which do not violate physical laws, do not exceed the state-of-the-art, are within the national space resources (technical manpower, money, management, etc.), and containing only those missions for which we are "ready," i.e., for which we have developed a national will to perform them. These various barriers are lines which are the sides of the box. The last line, national will, is more hazy than the others, but the significance of this line may be noted by observing the crystallizing effect which the late President Kennedy's announcement of the lunar exploration mission had on the aerospace industry. Or, contrariwise, we may note the difficulty of "selling" the civil defense program to the Nation, even though it may have much logic on its side to commend it. In this "feasible" box, then, are still an infinite number of possible alternative space programs, and from these we must somehow select those that are "preferred," "make good sense," or are "optimum."

As examples of the need for a model to include the many complicating factors, we may look at the history of aircraft development. At first, decisions were simple: Man wanted to demonstrate a flyable design. Later, early mail aircraft were compared as to speed, reliability, economy, and safety. With the appearance of jet aircraft considerations of number of passengers carried, speeds,

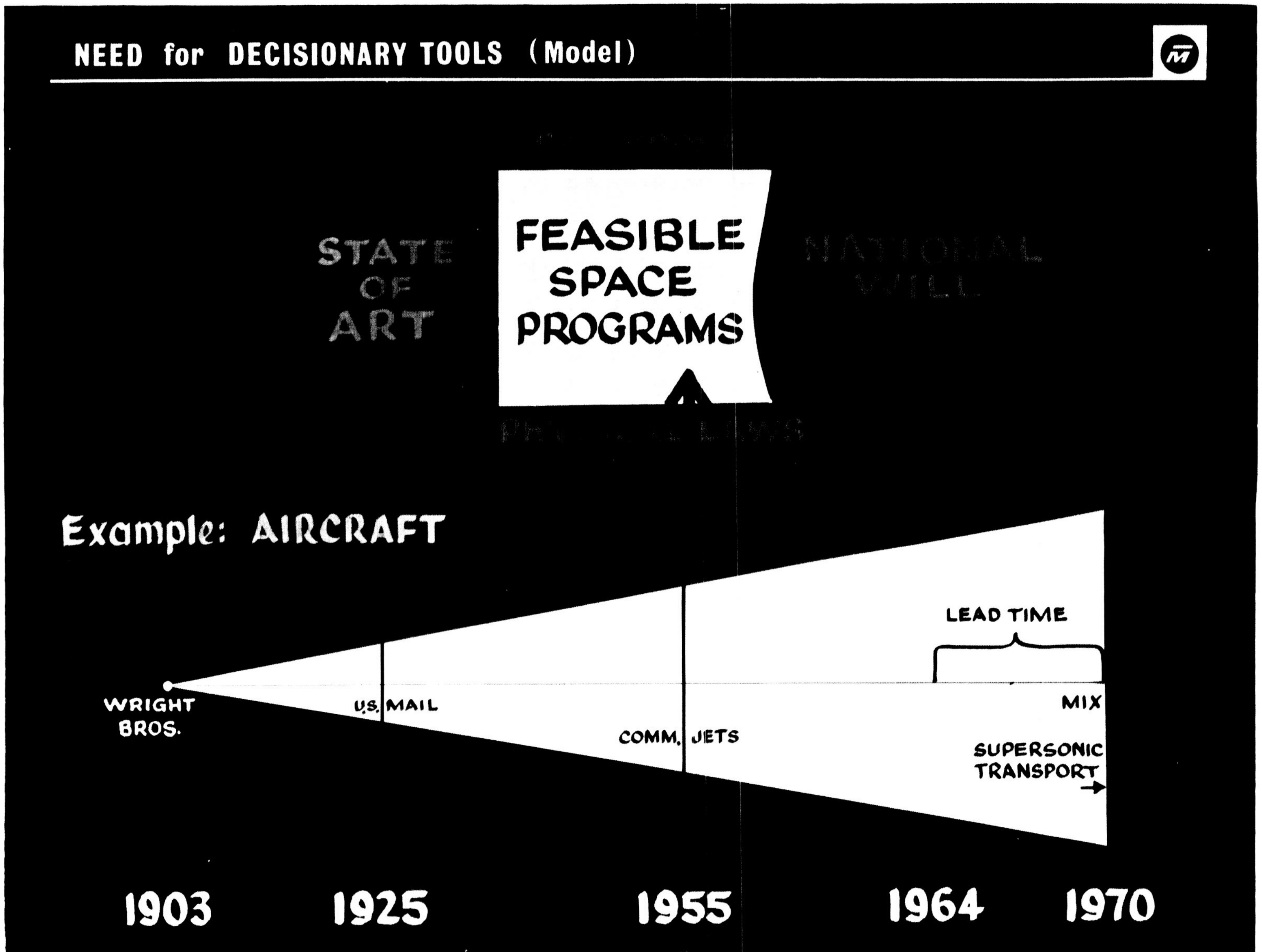


FIGURE 1. NEED FOR DECISIONARY TOOLS (MODEL)

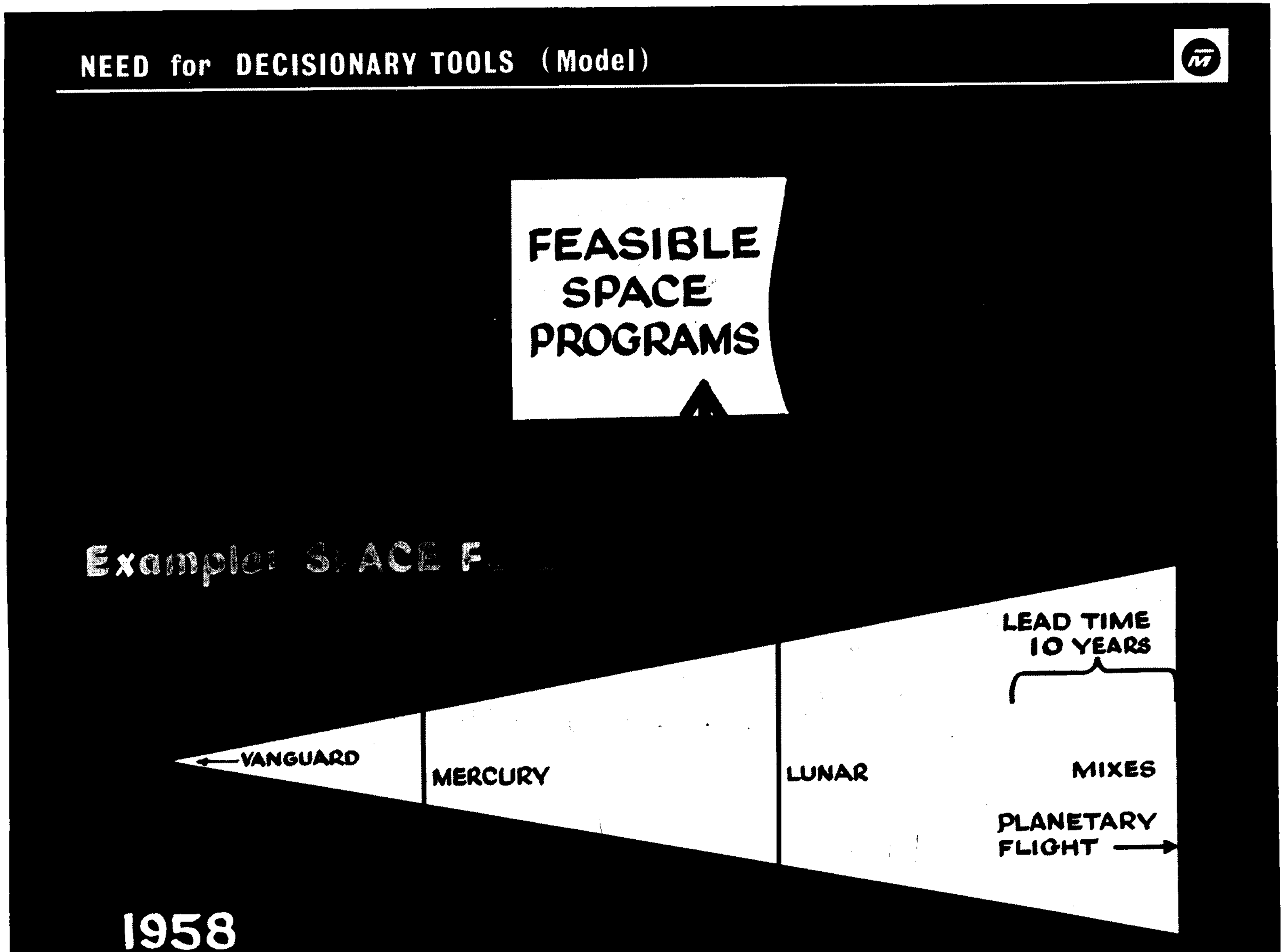


FIGURE 2. NEED FOR DECISIONARY TOOLS (MODEL)

which routes were already economically serviced by propeller craft, etc., became important. In other words, not only was it necessary to select from among jet designs on the basis of inherent design features, but the overall aspects of the transportation network had to be considered in deciding when and what kind of jet transport to build. This decision is even more apparent in the case of the supersonic transport where the European aircraft design forced the U.S. design to achieve Mach 3 if it was to offer advantages.

The longer lead times needed today from decision on go-ahead to the date of the operational vehicle has increased the need for valid planning. The evolution of spacecraft from Vanguard and Explorer I to the planetary designs discussed at this symposium are a further example of this complicated interdependence between new commitments to missions and vehicles and the already existing hardware. Moreover, if we select the optimum booster and spacecraft for each mission (as happens when only single missions are analyzed) we have no guarantee that a space program which is the totality of these optima is an optimum space program. In fact, if we evolved a space program which did not develop several standard spacecraft and boosters, but developed those which are optimum for each mission, a disproportionate amount of R and D money would be expended per ton delivered to planetary and other destinations. We would not be developing economical space transportation, but providing "economy at any price."

If we are to evaluate whole space programs quantitatively, we must define precisely what we shall mean by this term. By definition in this paper a "space program" shall mean a statement of all trips from origin point i to destination point j with vehicle k or vehicle components k1, k2, k3, etc., to be accomplished in a given year with specified fast or slow trip time. Thus, if the model is to be utilized to select one of a number of space programs, (that is, to select between ambitious programs, austere programs, fast moving or "crash" programs, or slow, deliberately developed programs) one could obtain quantitative measures of the value of such space programs in terms of yearly costs, total costs, payloads delivered, and other measures, merely by applying the model to the set of space programs under consideration. Each program is laid out in detail from the present year to the year 2000. On the other hand, if it is desired to explore a given space program to improve its overall efficiency, one can re-run the model with a given space program input and move or change the dates of various missions within the program, change the vehicles selected, change the mode (that is, fast or slow) with which a space exploration shall be accomplished, etc. Figures 3 and 4 outline a sample space program. Notice how several corridors of technological development lead to a specific mission goal. This may be seen for the lunar base, where Gemini-Apollo technology is finally superceded by the nuclear avenue of development.

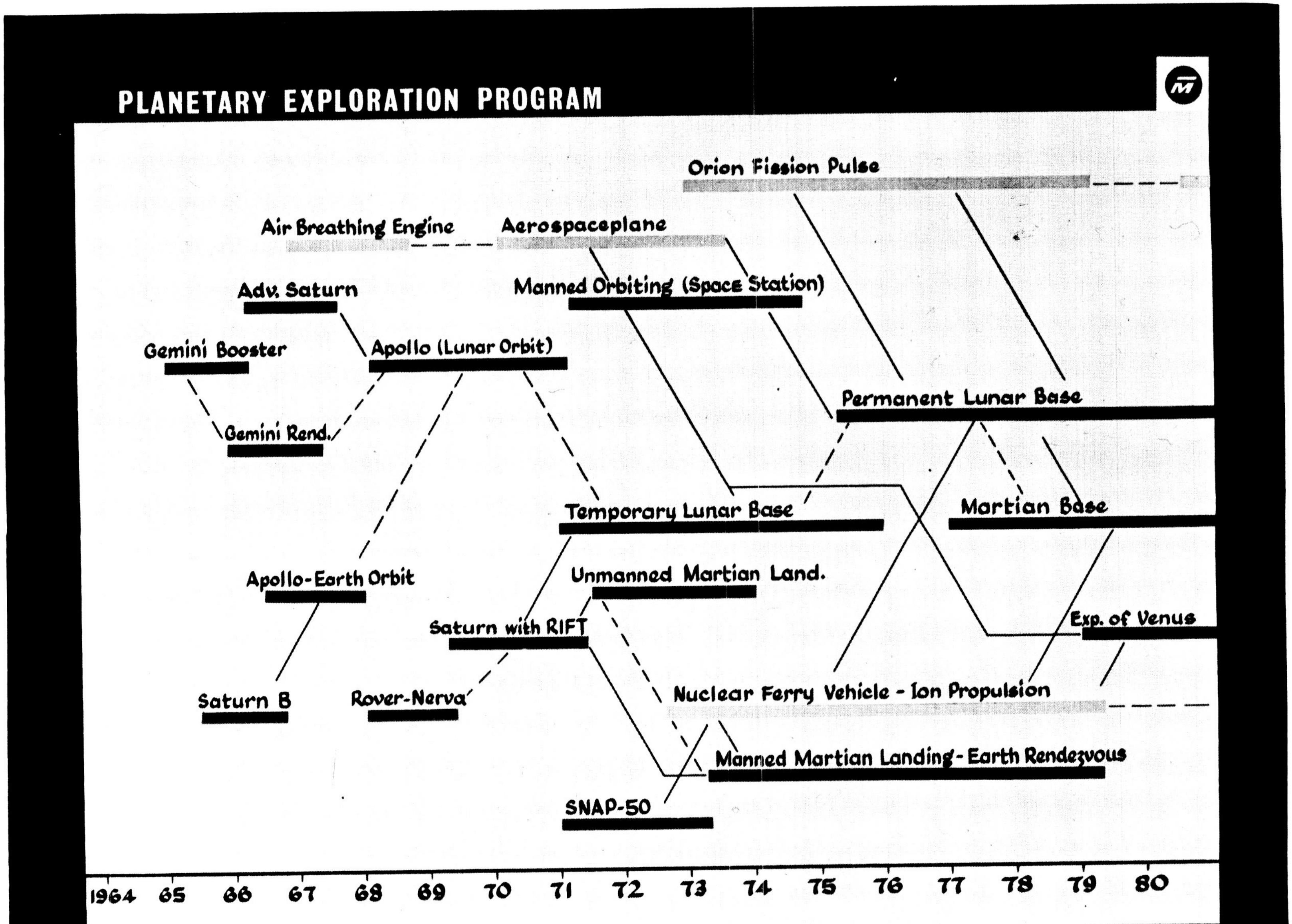


FIGURE 3. SAMPLE SPACE PROGRAM AS MODEL INPUT

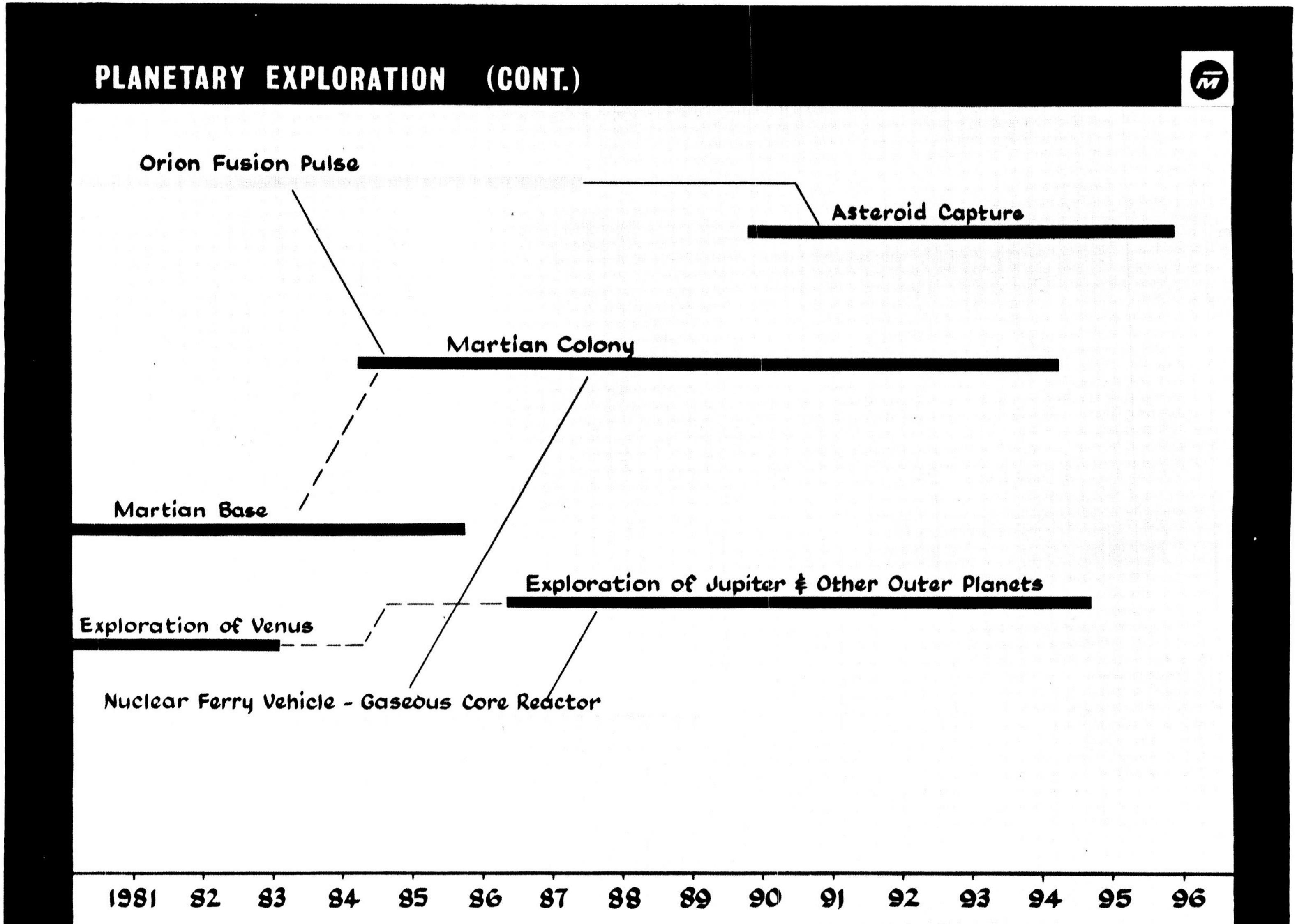


FIGURE 4. SAMPLE SPACE PROGRAM AS MODEL INPUT (CONT.)

What are the requirements for developing this planning model? The selection of a large booster, for example, the post-Saturn booster, or the decision to develop a particular family of spacecraft such as ion-propelled interplanetary craft, does not rest alone on what is optimum for a particular mission, as has been stated. Rather, the capabilities and efficiencies of the vehicle under consideration must be related to the entire space program. Hence, the analytical model must accept as an input entire space programs and must evaluate the efficiency of a concept with respect to all other activities of that program.

Secondly, the model must also, in its final form, be capable of real time employment in support of planning conferences; for example, at a planning meeting, questions as to the consequence of moving the development of a booster forward or backward in time, or to delay a program of development, should receive reaction and data from the running of the analytical model while the conference is in session. In this way, the planning tool can be of immediate value and can contribute to the improvement of decisions. In effect, the model does not make decisions, rather it provides planners with a measure of the consequence of various decisions and can be used to determine relative space program optima by repeated model runs. It will require careful model design and maximum use of modern input-output techniques to contain enough detail in the model to preserve validity, but not so much to make it tedious to use or unbelievable by overwhelming the user.

How shall the finished model be used to aid the planners decisions? As Figure 6 indicates, the inputs to the planetary model are particular space programs, the vehicles or vehicle concepts that are to be employed in implementing the space program, and the missions and modes of travel which are scheduled within the program. The complete specification of this mission-vehicle-time sequence might be viewed as a large $\overline{x}$-vector, the knowledge of the components x_i spelling out the total input. The planetary model is essentially an operator which accepts this vector input. By means of a complicated logic network, by using the basic equations of planetary motion, and from the vehicle design equations the model calculates a vector output $\overline{y}$. The output vector $\overline{y}$ is composed of raw output values such as numbers of launches of specified boosters each year, payload delivered by mission each year, etc., and also indices of merit of evaluators of the space program. Among the latter we consider indices such as annual program costs, probabilities of attaining key missions at scheduled times, dollars per pound delivered to orbit, to the Moon, and to other destinations, dollars per man-year spent at various destinations, etc.

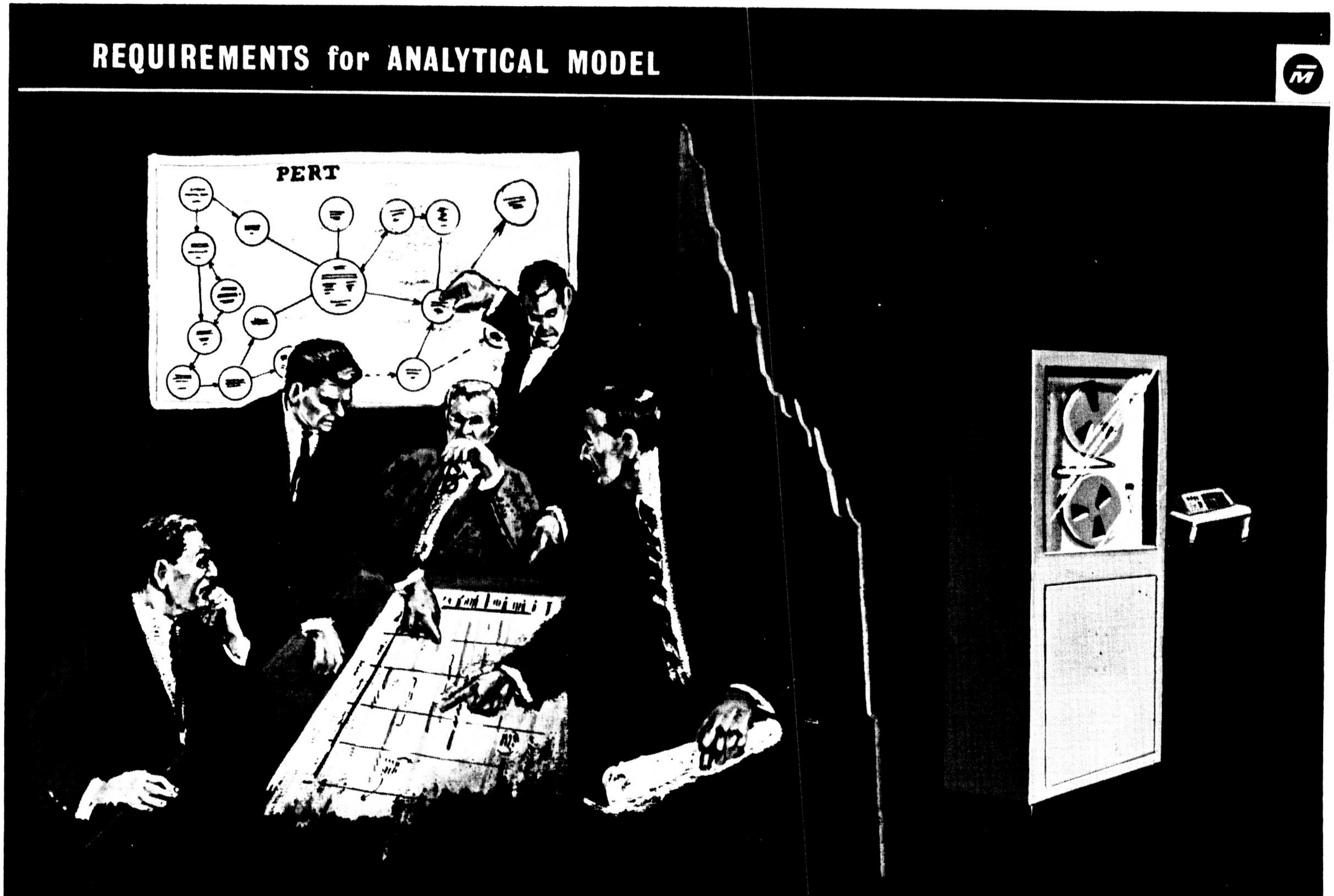


FIGURE 5. REQUIREMENTS FOR ANALYTICAL MODEL

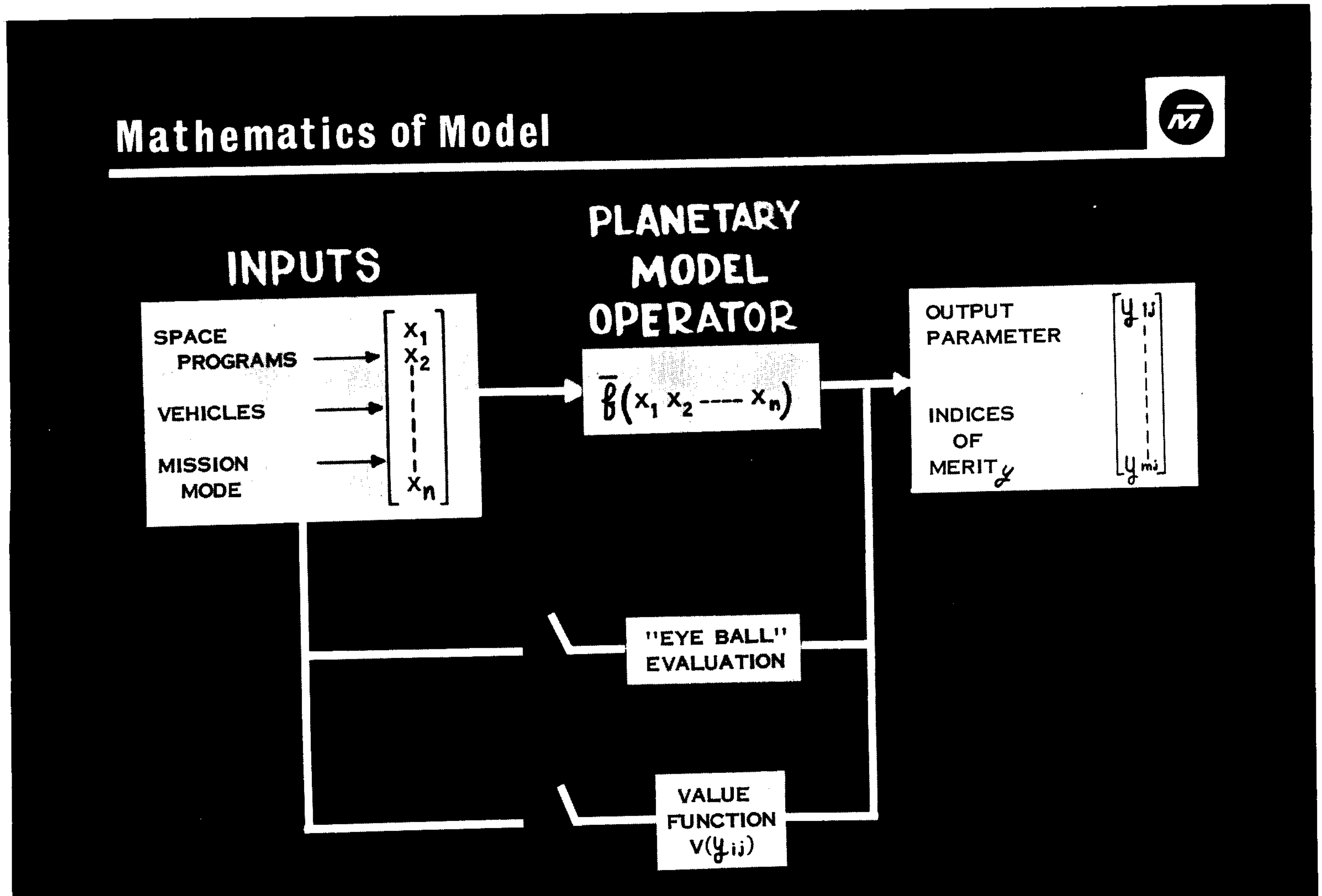


FIGURE 6. MATHEMATICAL ANALOGY OF THE MODEL

At first the model would be used in conjunction with "eyeball" evaluation feedback. That is, by scanning the output vector $\bar{y}$, one would determine what changes seem to be required in the Space Program, the vehicles, or the mission modes of travel to improve certain indices. These, in turn, would be re-inputed and the output compared with previous outputs, thus "closing the loop." At a later time, various value functions $V(y_{ij})$ might be optimized. For example, it might be desirable to optimize "dollars per pound delivered to Earth orbit" by automatically sequencing the model through a large number of calculations in an attempt to determine an optimum set of inputs from among a specified set of inputs.

Re-examination of Figures 3 and 4, a sample space program, clearly indicates how missions may be varied within a space program. But, how do we vary the vehicles which perform the various maneuvers of the journey?

As one plans a particular space program as an input into the model, it is necessary to specify the particular vehicle which shall accomplish a certain journey or a leg of a journey. In the interest of model flexibility, this can be accomplished in two ways. First, a specified vehicle design as it has been developed in ancillary design studies may be employed directly (see Vehicle Design Cards in Figure 7). For example, one may specify Saturn V and Titan III in full detail: gross weights, delivery capabilities, reliabilities, etc.. Alternatively, one may elect that a certain portion of a specified journey shall be accomplished by a given family of vehicles, but no specific design of that family may be known to the model user. In this case, the model, in consideration of the particular time of the mission, will determine the ΔV requirement and, with the aid of simplified vehicle design formulas pertinent to the vehicle class and from payload delivery demands, will generate gross payload weights, reliability estimates, and gross vehicle weights for use in the overall evaluation of the space program. At the same time, cost data is stored for later use in the costing subroutines.

II. BASIC LOGIC AND PROGRESS TO-DATE OF THE PLANETARY MODEL

Before discussing the indices of merit further, the progress on implementation of the model, and broader application of the model, it is desirable to summarize in some detail the macro-logic of the model and some of the major subroutines. This is done in the next few figures.

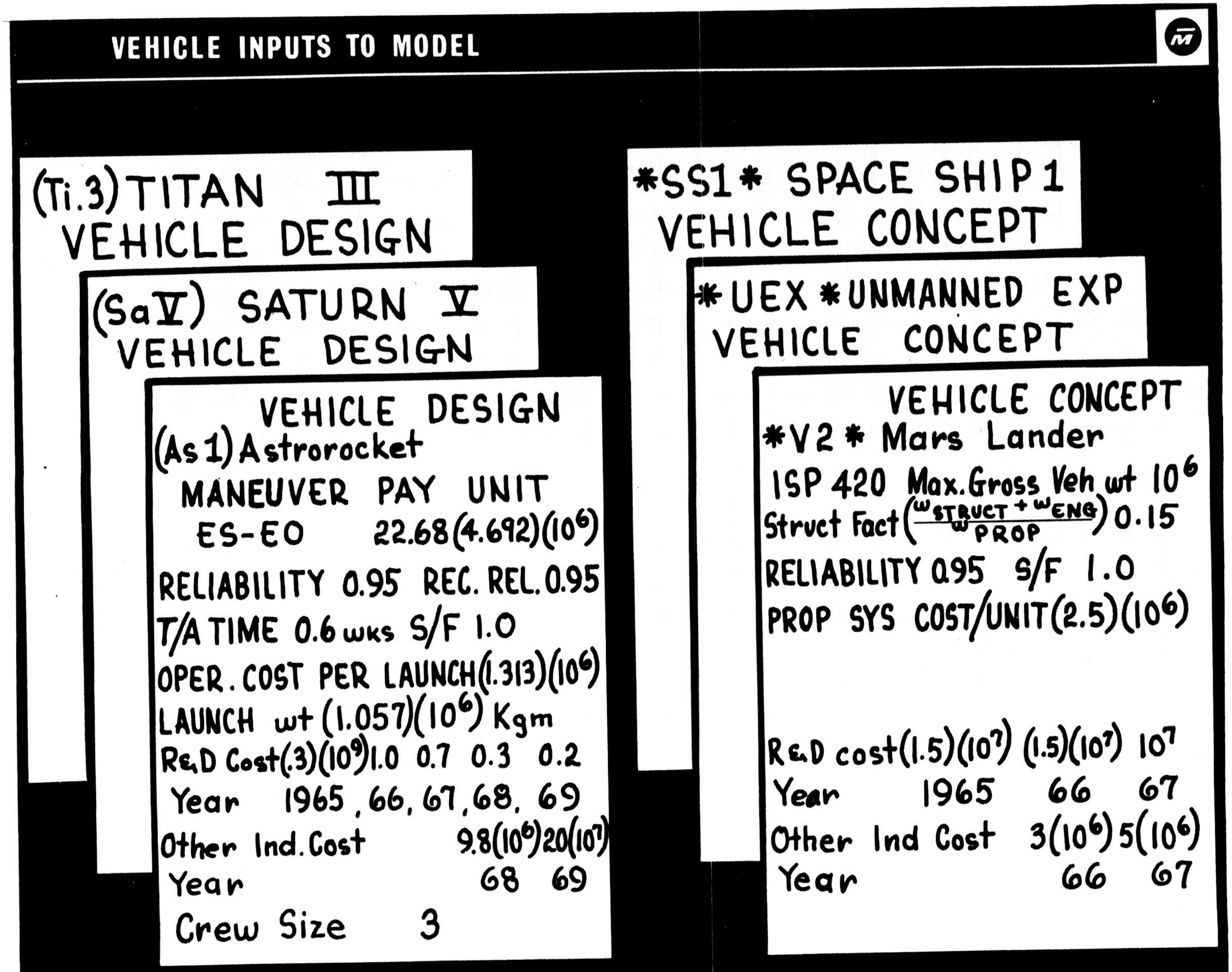


FIGURE 7. VEHICLE INPUTS TO MODEL

The flow diagram in Figure 8 shows the manner in which calculations are carried out for a given space program input. First basic planetary tables, gravitational constants, etc., are read in. The time analysis (probability distributions of attaining mission dates) can be used by itself or as part of a complete analysis of a space program. We shall speak of this time analysis subroutine later. After the time analysis, all missions and maneuvers (legs of journey within a mission), vehicle data, and the mating of a vehicle to each maneuver of a mission is read in. Then each mission is processed, maneuver by maneuver, until the space program is completed. For example, we begin with the first mission and find the vehicle specified for the first maneuver of that mission (launch to orbit by Saturn V, say). As will be explained in other figures, we determine ΔV, payload, and then determine how many Saturn's are needed. If a specific booster such as Saturn V is not specified, but instead we see "large chemical booster," with specified propulsion stages, mass fractions, etc., we size the booster. In this manner, we cycle through all maneuvers of mission one, then proceed to mission two, etc. Details calculated are ΔV's, payloads, trip times, number of launches, etc. Finally, after all missions are processed, each vehicle and each mission is costed and costs are assigned to the various years. Now the evaluation of particular vehicles (boosters or spacecraft) can be undertaken by computing the appropriate indices. Finally, the indices, which offer an evaluation of the space program itself, may be computed. Then we may proceed to the next space program, and so on.

In the next few figures we shall explore briefly some of the subroutine logic.

When the trajectory subroutine, Figure 9, is employed in conjunction with a concept vehicle, the mission date is put in by the user and is checked against available launch windows. If the trip date is unfavorable for reasons of solar activity, high energy requirements, or other constraints, which may be given, the next available date is used. Note that on the right in Figure 9 several trajectory phases are available to be selected. Each phase is a branch of computer logic. The program thus enables other phases of orbital maneuver to be easily added without destroying the available computer program. For a parking orbit or, say, a Mars surface to Mars surface maneuver, stay time only is calculated. For the other phases both ΔV and trip time are outputs. Where applicable, table look-up is used to obtain a hyperbolic excess speed from which ΔV is calculated. The user is free to select fast or slow trips, Hohmann transfers, aerodynamic or retro-thrust reentry, etc. Other phases which have not yet been programmed, but which shall be available in the final model program are:

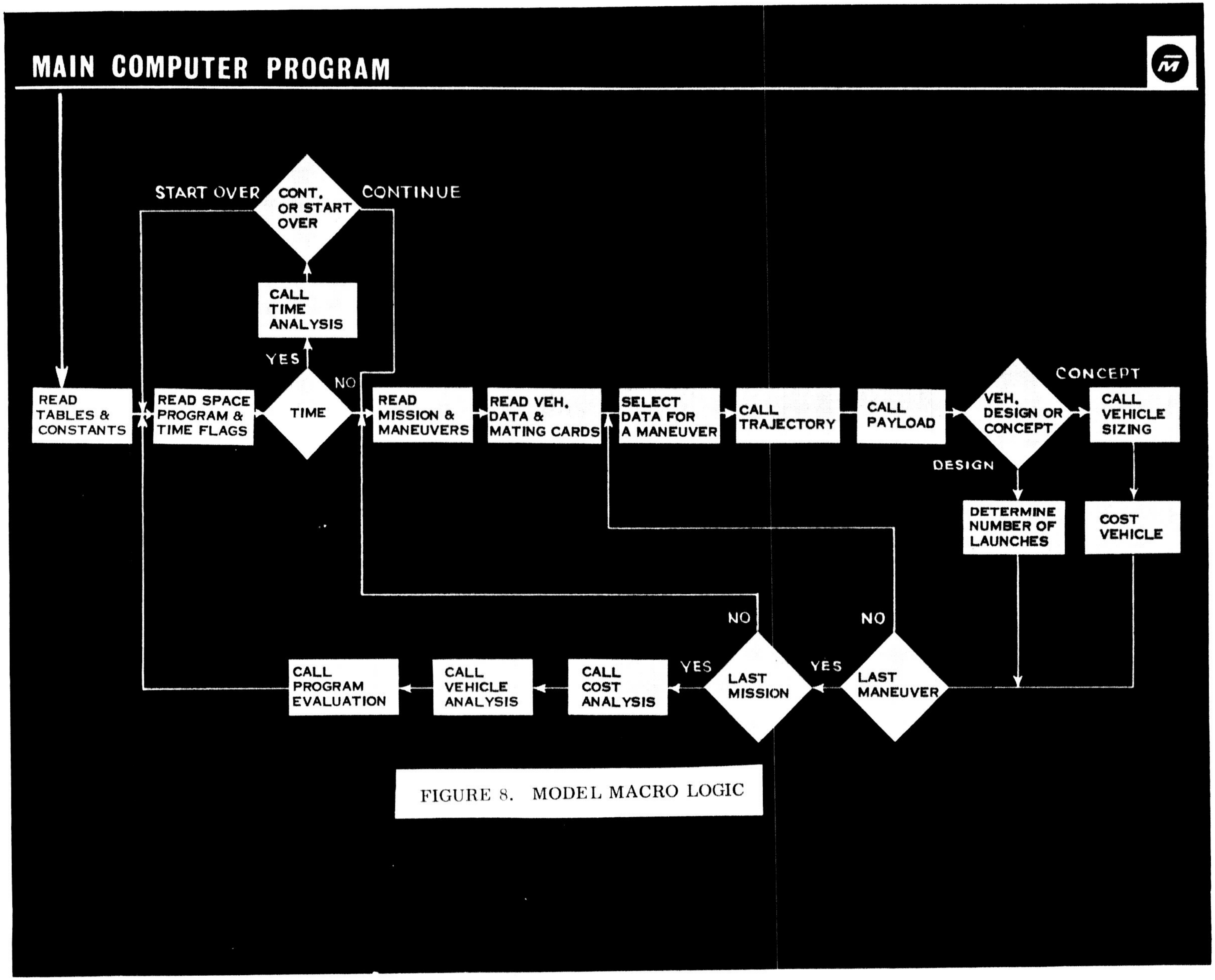


FIGURE 8. MODEL MACRO LOGIC

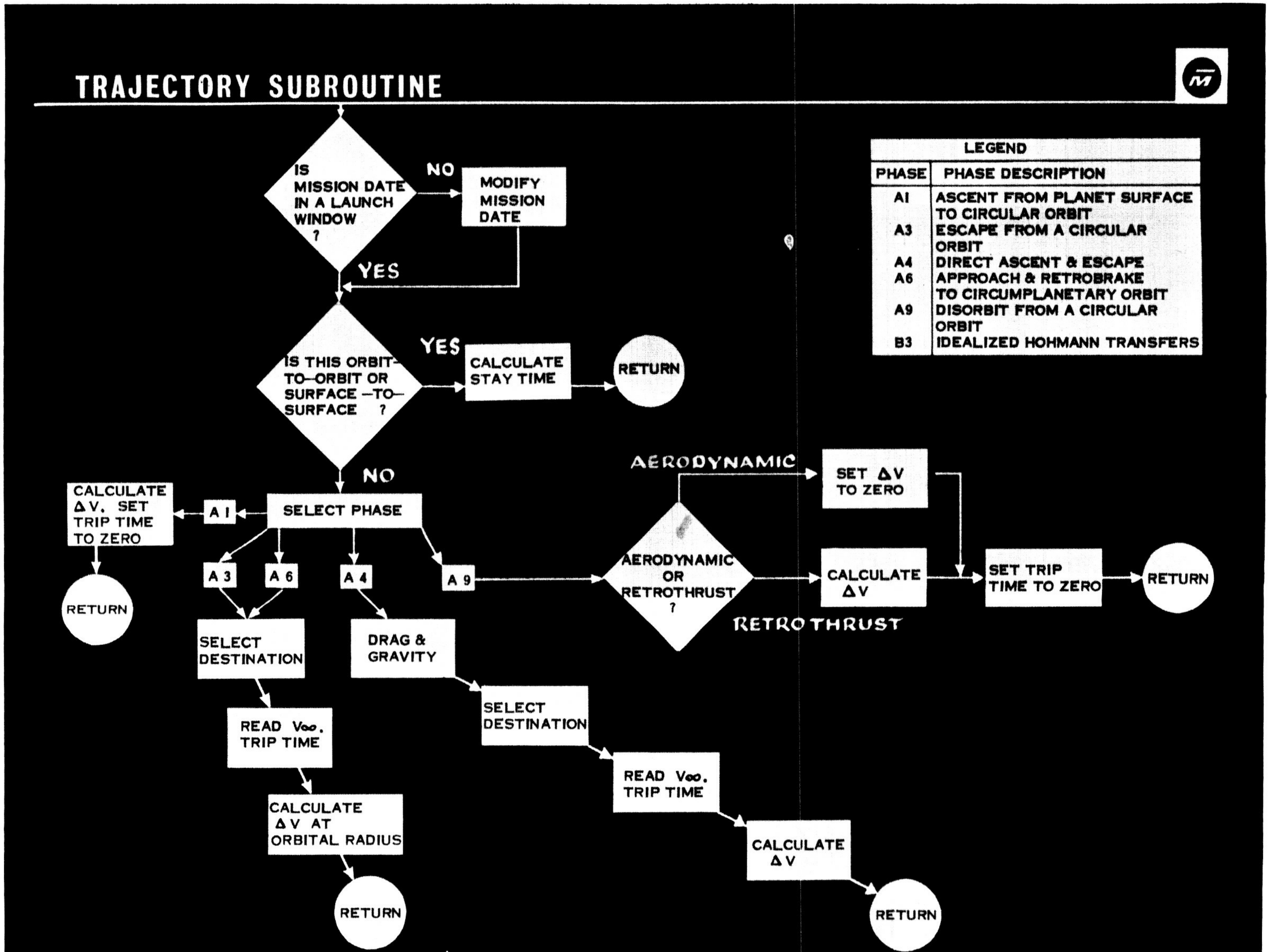


FIGURE 9. TRAJECTORY SUBROUTINE

(a) plane changes

(b) circularize elliptical orbit

(c) heliocentric phases.

In the payload subroutine of Figure 10, the gross payload items treated are: storm shelter shielding weight, weight of power supply, life support weight, net payload (a user input), and the calculated weights of other vehicles or upper stages. Under storm shelter shielding we have not yet included a model to calculate shielding for a conceptualized nuclear powered vehicle, but this is planned. Four levels of shielding (including zero shielding) are available to the user, and there are six types of life support including open system, regenerative systems, partially and completely closed systems. There is provision for the user to assign an orbital burden factor of his choice. By this factor we mean that if a convoy is to be assembled in orbit, the mass delivered to Earth orbit exceeds that to be launched from Earth orbit. The excess depends upon the total traffic flow being processed by the orbital operation and includes the space station, the men (and their logistic support) used to prepare the space launches, the checkout gear, the fuel boil-off losses, the losses in rendezvous and handling operations, etc. A subroutine is under preparation to generate an orbital burden factor as a function of traffic flow. The total plan of orbital operation certainly also changes this factor.

A similar ground operations model will be added to "size" the ground facilities for a given type of booster. The facility must meet the launch demand schedule which is necessary to permit orbital operations to launch the missions within the various launch windows.

Using the output of the trajectory submodel and the structures factor, the mass ratio, payload ratio, and gross vehicle weight are calculated. If the vehicle is part of the payload of another vehicle or stage, the gross weight is properly stored in a block of storage reserved for payload of that other vehicle (Fig. 11).

The user may specify the class of a vehicle by specifying a maximum gross vehicle weight. If the calculated gross vehicle weight exceeds this maximum, the payload is distributed over a calculated number of properly sized vehicles.

If the vehicle is involved in a rendezvous maneuver, the structures and propellant weights may, at the user's option, be burdened or penalized with extra rendezvous fuel and structural weight for rendezvous.

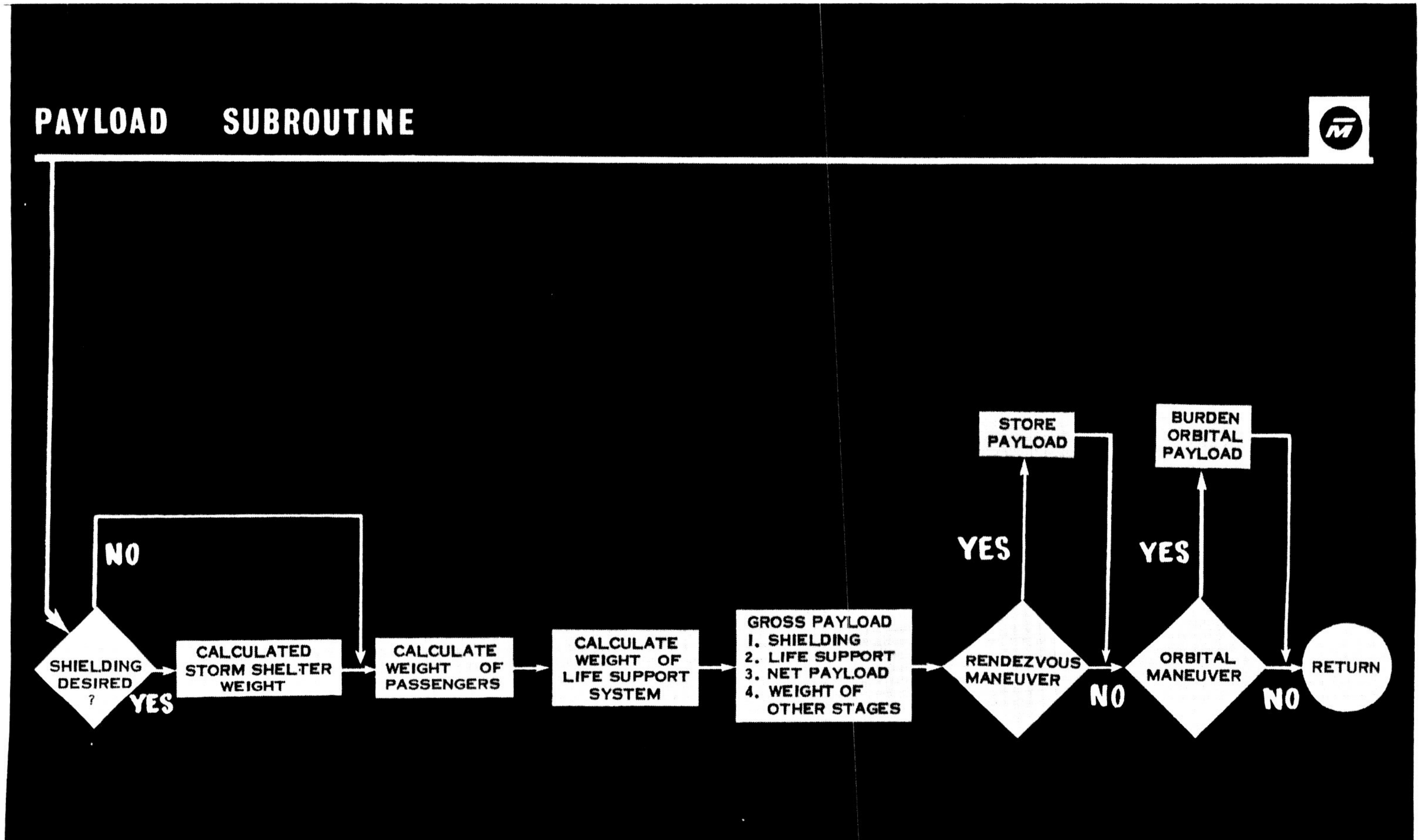


FIGURE 10. PAYLOAD DETERMINATION SUBROUTINES

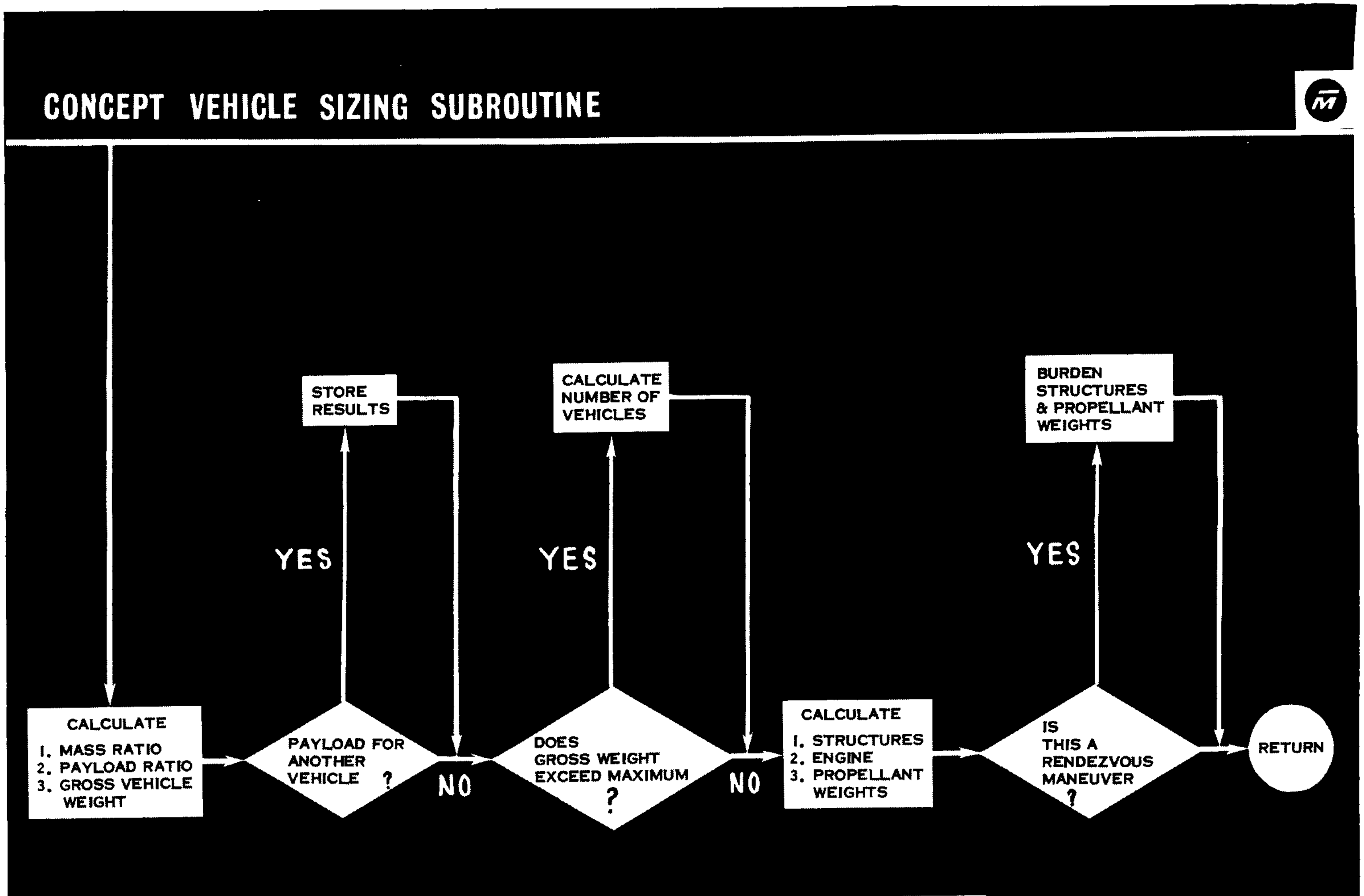


FIGURE 11. CONCEPT VEHICLE SIZING SUBROUTINE

Figure 12 is greatly oversimplified. Each stage or spacecraft module is determined to be recoverable and reusable, or not. If expendable, then the total number of successful launches required for this type of vehicle as used in an entire Space Program is calculated and is combined with launch and recovery reliabilities and turn-around-time to calculate, by learning curve theory, the number of vehicles to be procured and the operating cost of each vehicle. In determining costs, the ground operations model must have been previously employed to cost required launch facilities. For recoverable vehicles we must estimate the number of reuses possible per vehicle and must also purchase refurbishing facilities sufficient to give satisfactorily short turn-around times. These are estimated by other rather simple models. All of these costs, R&D costs, and other indirect costs are distributed over the appropriate cost assignable years. These are then part of the total annual costs and total space program costs out to the year 2000. Such costs assist in evaluating or understanding the implication of a proposed space program.

The model will calculate and tabulate certain output parameters such as number of astronauts needed per year which are of value in their raw form for planning such secondary programs as the training of personnel.

The output parameters can be combined in various ways to give indices on scales useful for making quantitative comparisons. Different indices are produced for the needs of the engineer, the budget planner, the administrator, etc.

Two indices in particular from Figure 13, the probability of meeting key mission dates, and budget fit, are explained in succeeding Figures.

We mentioned earlier the option of a mission time analysis. If a particular vehicle is included in a Space Program, a technical development program must be laid out for that vehicle. This leads to a PERT type network. For each event or milestone in the network, the user may insert (or the computer program may provide) an earliest completion data, expected completion date and latest plausible mission date. The user may also specify a triangular, rectangular or arbitrary (this is more input work) probability distribution of completion times. Where such events or milestones are in series, the completion time is the sum $(t_1 + t_2)$, and the distributions of t_1 and t_2 are to be convoluted. Where the events are in parallel, the completion time is max (t_1, t_2), and this distribution is determined by the computer program. The timetable subroutine thus calculates a probability

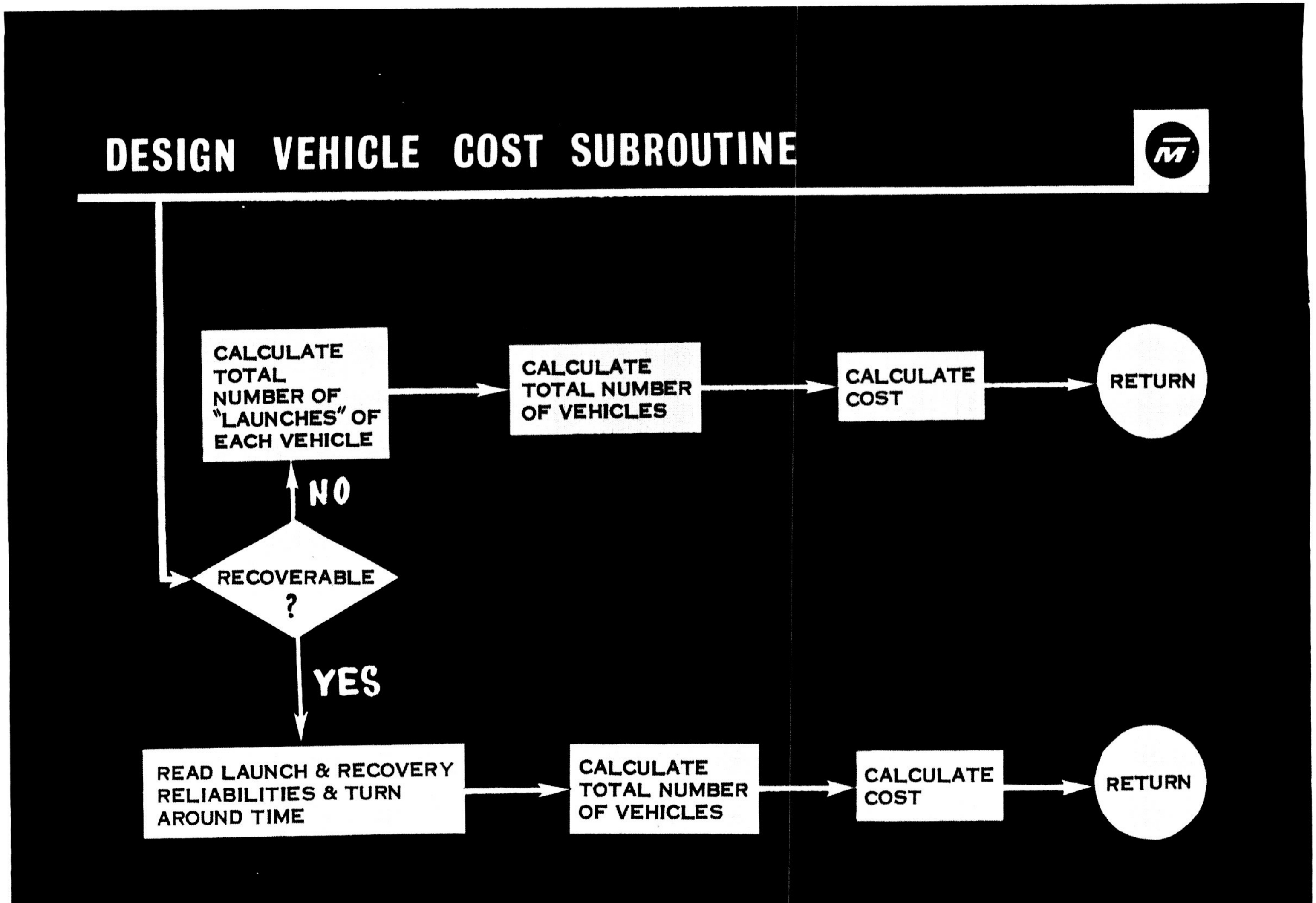


FIGURE 12. DESIGN VEHICLE COST SUBROUTINE

INDICES AND VALUE ANALYSIS

OUTPUT PARAMETERS
- $ PER YEAR – TOTAL PROGRAM
- LAUNCHES PER YEAR BY VEHICLE TYPE
- PROBABILITY OF MEETING TIMETABLE
- MANPOWER REQUIREMENTS
- NUMBER OF ASTRONAUTS
- PROBABILITY OF MISSION SUCCESS
- DIRECT OPERATING COSTS FOR VEHICLE "A", "B", ETC.

INDICES
- $ PER KILOGRAM DELIVERED IN ORBIT
- $ PER KILOGRAM DELIVERED TO TARGET PLANET
- SAFETY FACTOR (FATAL ABORTS PER NUMBER OF TRIPS)
- GENERALIZED COST
- $ PER KILOGRAM DELIVERED AT KEY DATE
- BUDGET FIT

FIGURE 13. INDICES AND VALUE ANALYSIS

distribution of the dates on which the vehicle achieves operational status and the dates on which particular missions may thus be accomplished. The methods of calculation are a refinement of some of the techniques used in PERT type analyses. The network in Figure 14 illustrates possible milestones for the Lunar Landing Program.

In addition to providing the probability of meeting scheduled mission dates, a measure of technological risk can be injected into decisions between vehicles of the same type - say launch vehicles. Figure 15 shows the results of applying the PERT type analysis to the readiness dates of two types of Post Saturn boosters, NOVA I and NOVA II. Suppose that on the basis of the indice, dollars per pound to orbit, NOVA II appeared to be a better choice than NOVA I. However, as the figure indicates, one can be more certain of realizing the savings of NOVA I at an early date whereas the greater savings ratio of NOVA II may not be realized until later. The total program cost up to 1980 may be more with NOVA II! Such information could be quite pertinent to the decision.

Although not enough effort has been placed upon careful projection of the resources for space exploration, some estimates are available. Figure 16 shows a projection of Gross National Product obtained by assuming an annual percentage increase. By assuming a fixed percent of such resources (or other relationship) available for the space program, optimistic and pessimistic budget lines may be drawn as in Figure 17. Similar charts may be plotted for the various manpower resource types: scientists, engineers, technicians, etc.

Not only must the total 1964 - 2000 space program cost (in these various terms) be within budget, but in no year should a large overrun occur. This condition is shown in Figure 17. Using the model, the planner can move missions, change vehicles, change maneuvers, etc., until the funding hump is smoothed, thus averting future embarrasment.

III. IMPLEMENTATION OF THE PLANETARY MODEL

As of the date of this symposium, the basic approach and philosophy of the model has been established. As indicated in Figure 18, programming of the macro-logic and most of the major subroutines is complete. The core program has been operated once, and estimates of operating times to change and to run with a new space program input, or to perturb a space program input, are given.

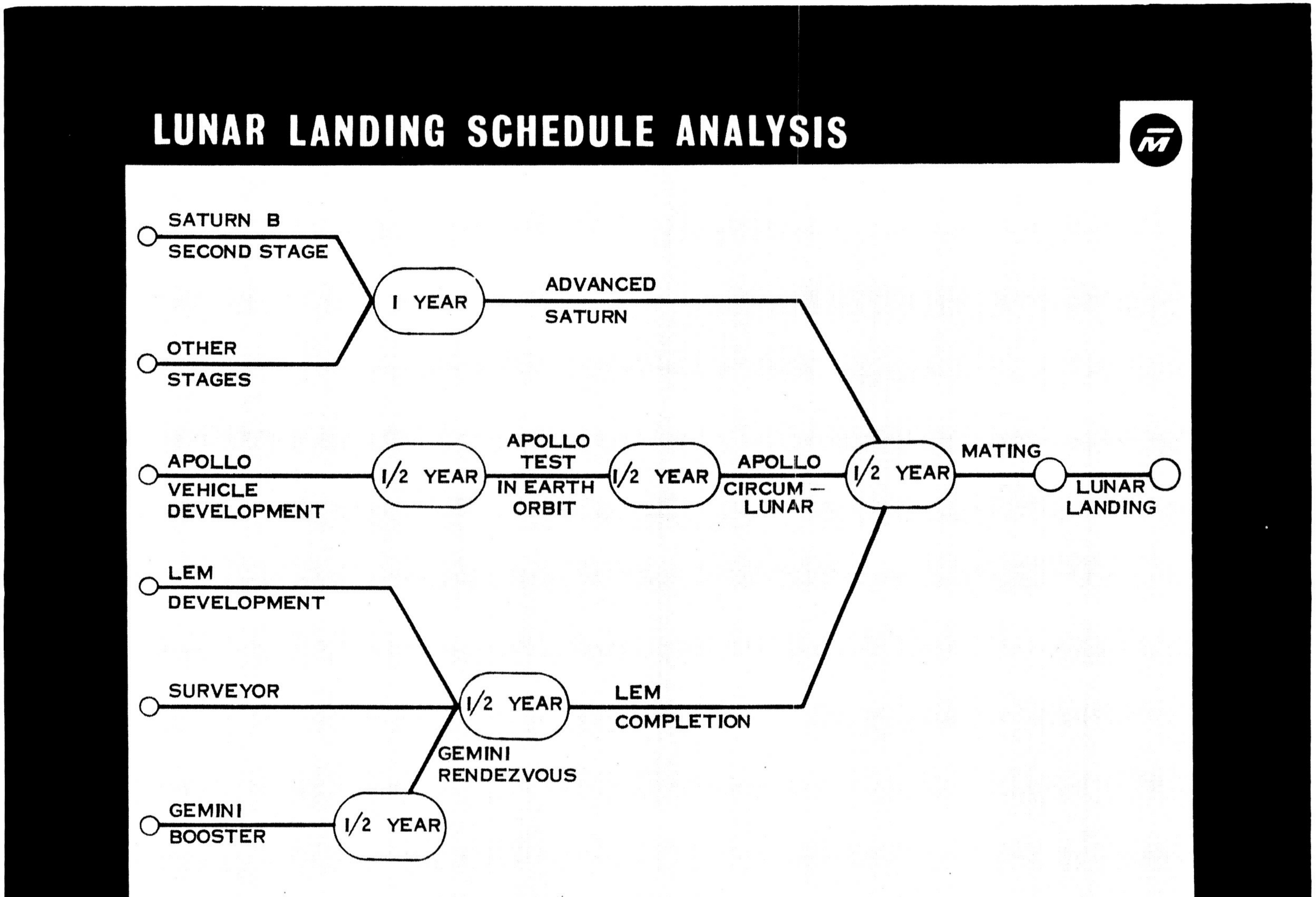


FIGURE 14. LUNAR LANDING SCHEDULE ANALYSIS

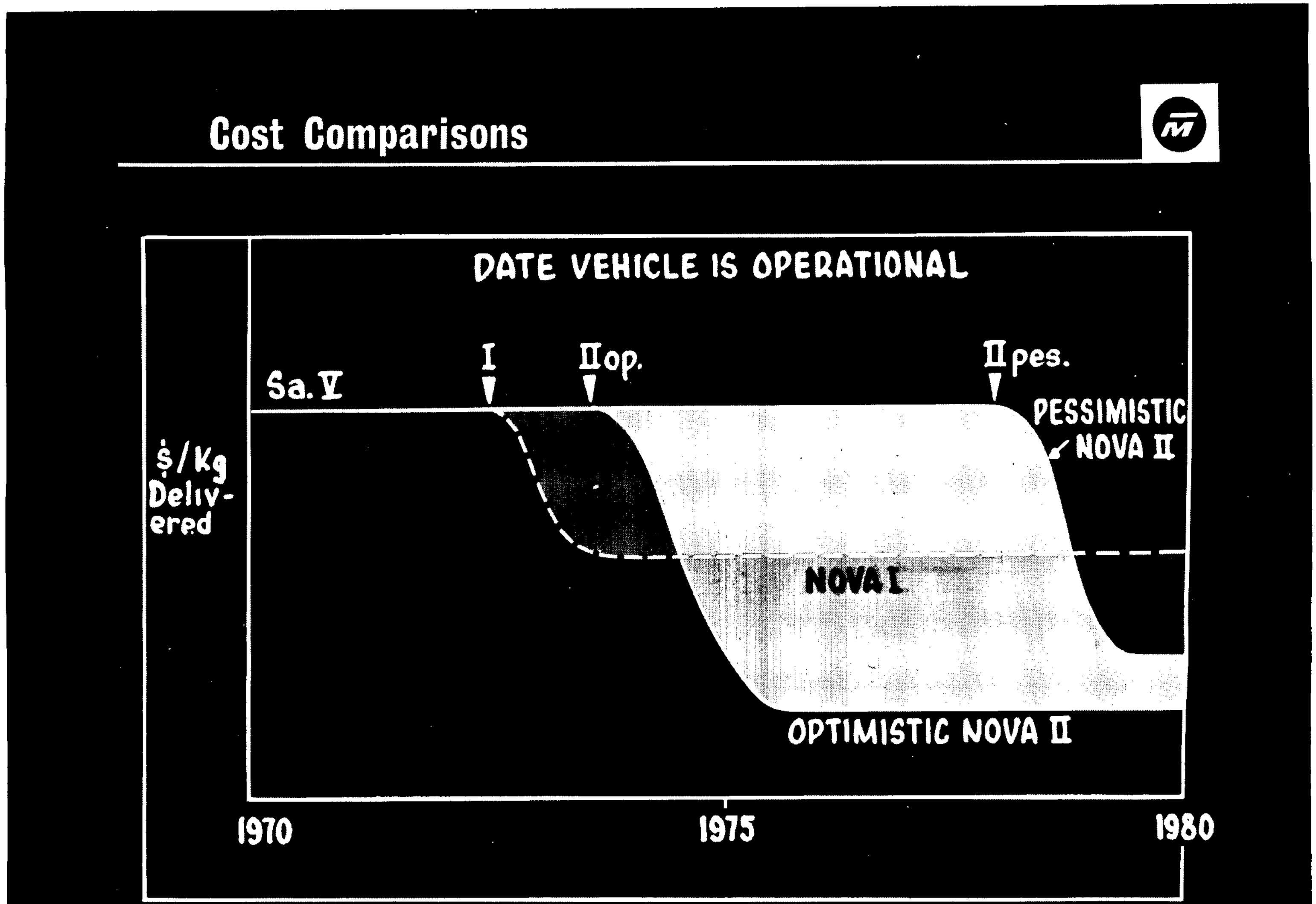


FIGURE 15. COST COMPARISONS

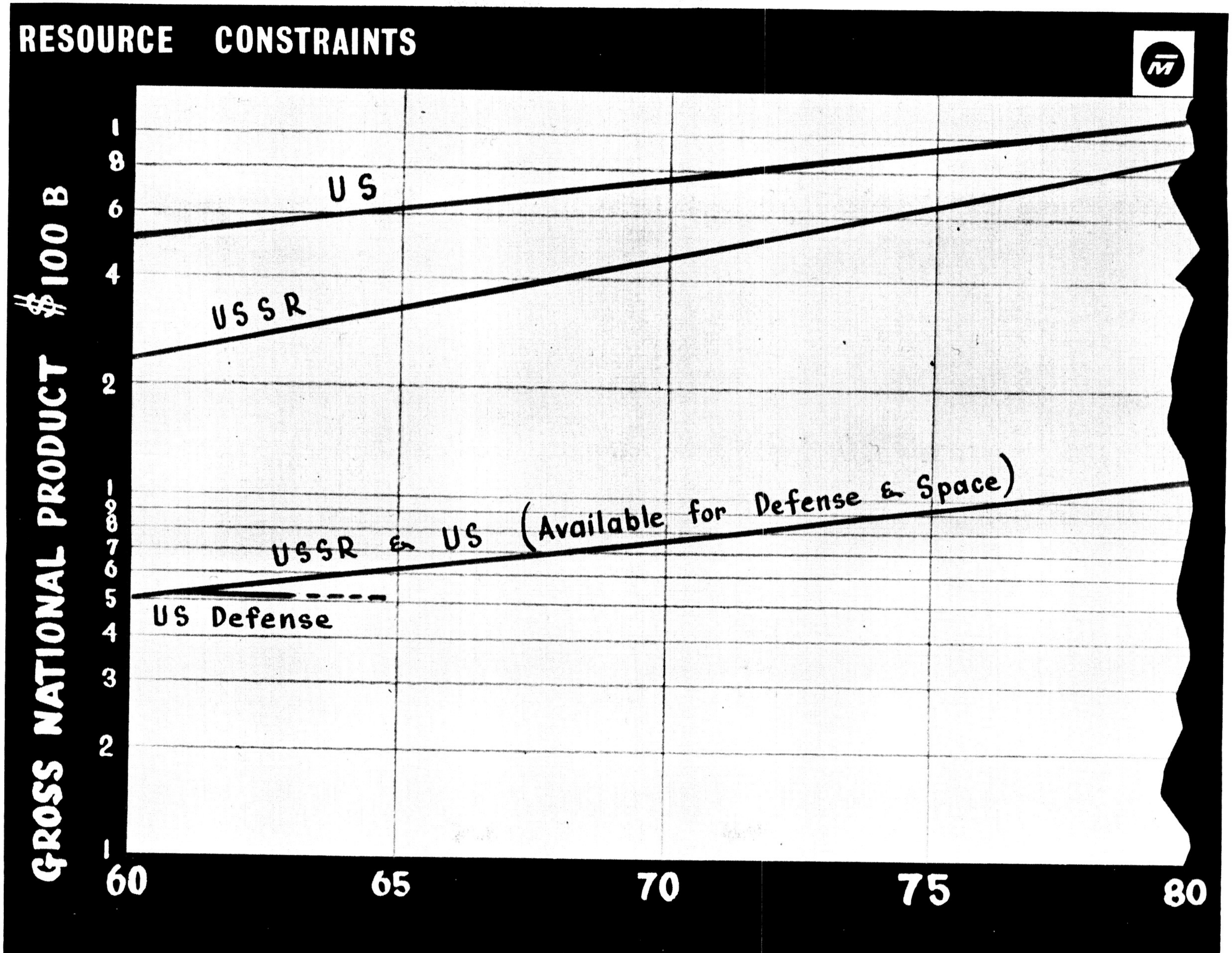


FIGURE 16. RESOURCE CONSTRAINTS

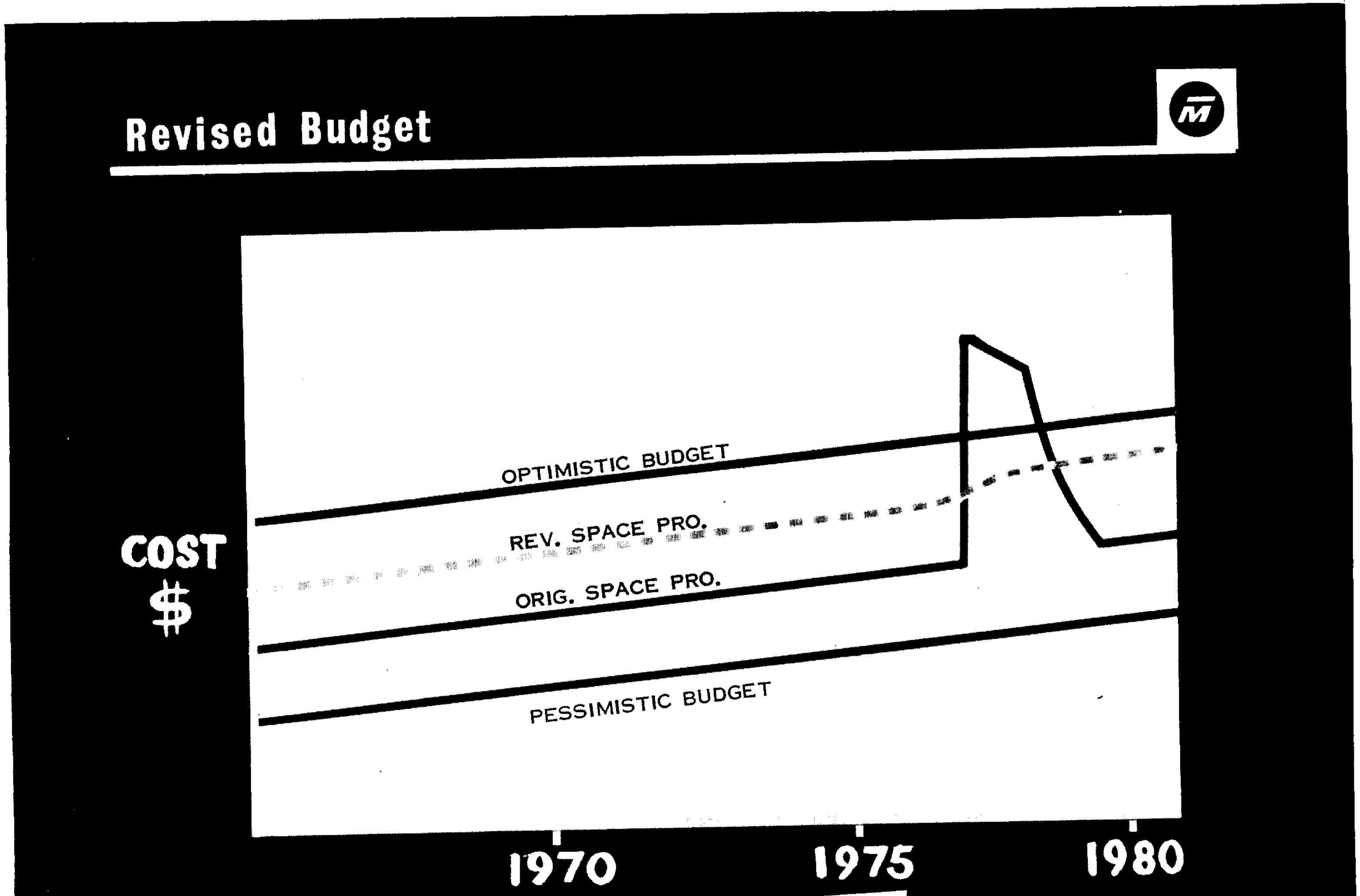


FIGURE 17. REVISED BUDGET

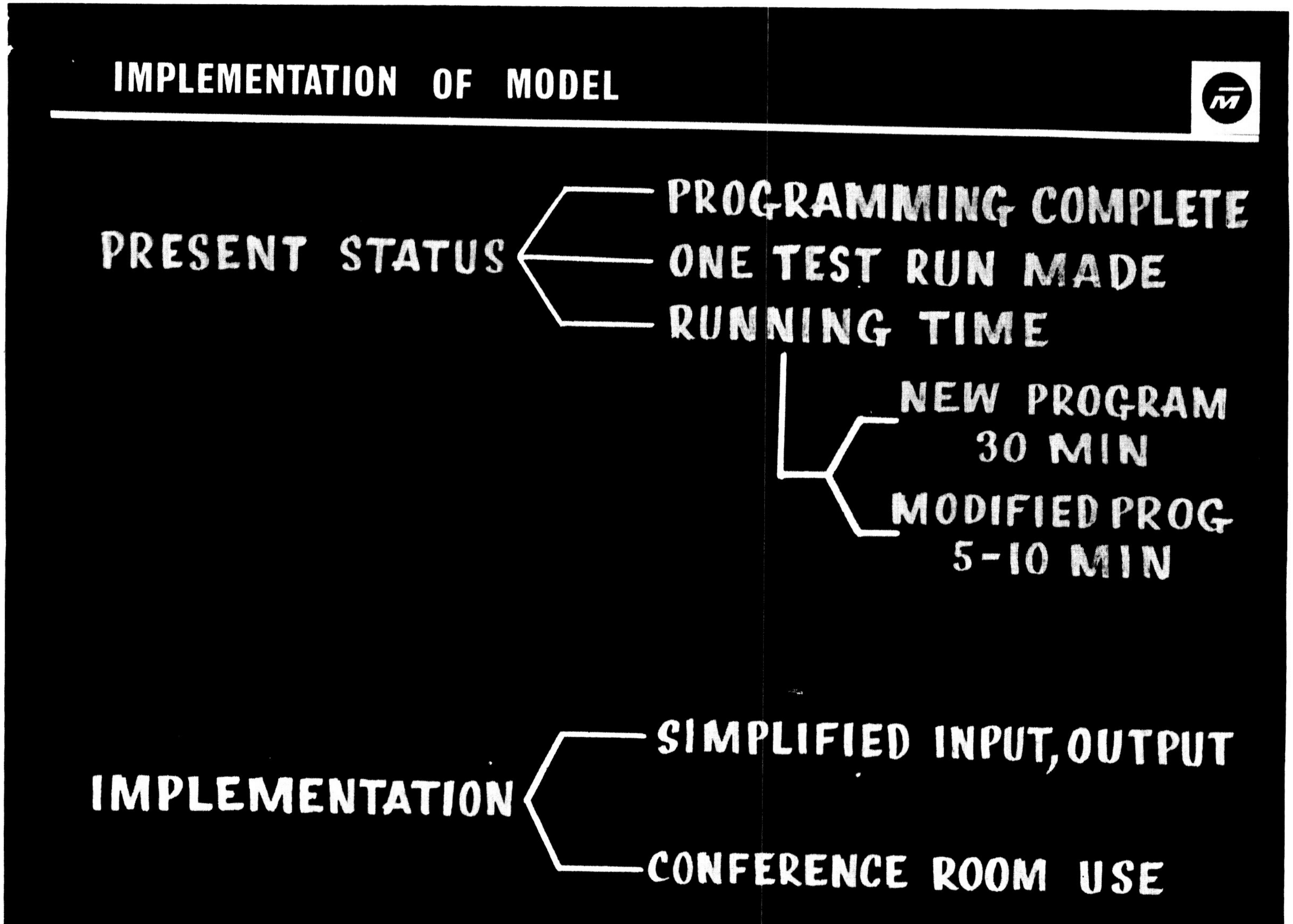


FIGURE 18. IMPLEMENTATION OF MODEL

There will now follow a series of runs in order to gain practice in using the tool. These runs will give insight into the kinds of problems which can best be answered by the present model and will indicate the necessary steps to produce an improved model.

The program has already attained a high level of flexibility by:

(a) being written so that space and launch vehicles that the user has designed himself or has read about in a report may be used to implement the analysis of a Space Program. Just use the vehicle design card format of Figure 7.

(b) being written so that additional tables of hyperbolic excess speed may be added.

(c) being written in a subroutine language so that the model may be expanded with a minimum of effort.

(d) having data input cards for already designed vehicles and missions available to the user.

(At the symposium presentation an example of the model as it was applied to a specific space program was displayed.)

IV. VALUE ANALYSIS AND FUTURE MODEL RESEARCH AREAS

One of the difficulties in building a model is to make it reflect value scales of different individuals. Dr. Koelle has suggested four basic value factors; payload delivery capability, scientific knowledge gained, prestige, and technological fall-out. The difficulty in using such factors is still twofold: how do we measure these factors and how do we weigh them? In Figure 19 several indices from the model output vector $\overline{y} = (y_{ij})$ are mentioned as possible contributory measures. Secondly, we may allow the user to weigh the factors in different ways. Thus a staff military planner and a committee of astrophysicists would rank the worth of "Payload Delivery Capability," "Scientific Knowledge Gained," "Prestige," and "Technological Fall Out" quite differently. This may be accomplished by assigning different numerical values to α, β, γ, and δ when calculating the total value of a program. Other non-linear value functions are under investigation. Additional feedback on value functions from high level planners is needed so that better value parameters may be added to the program output.

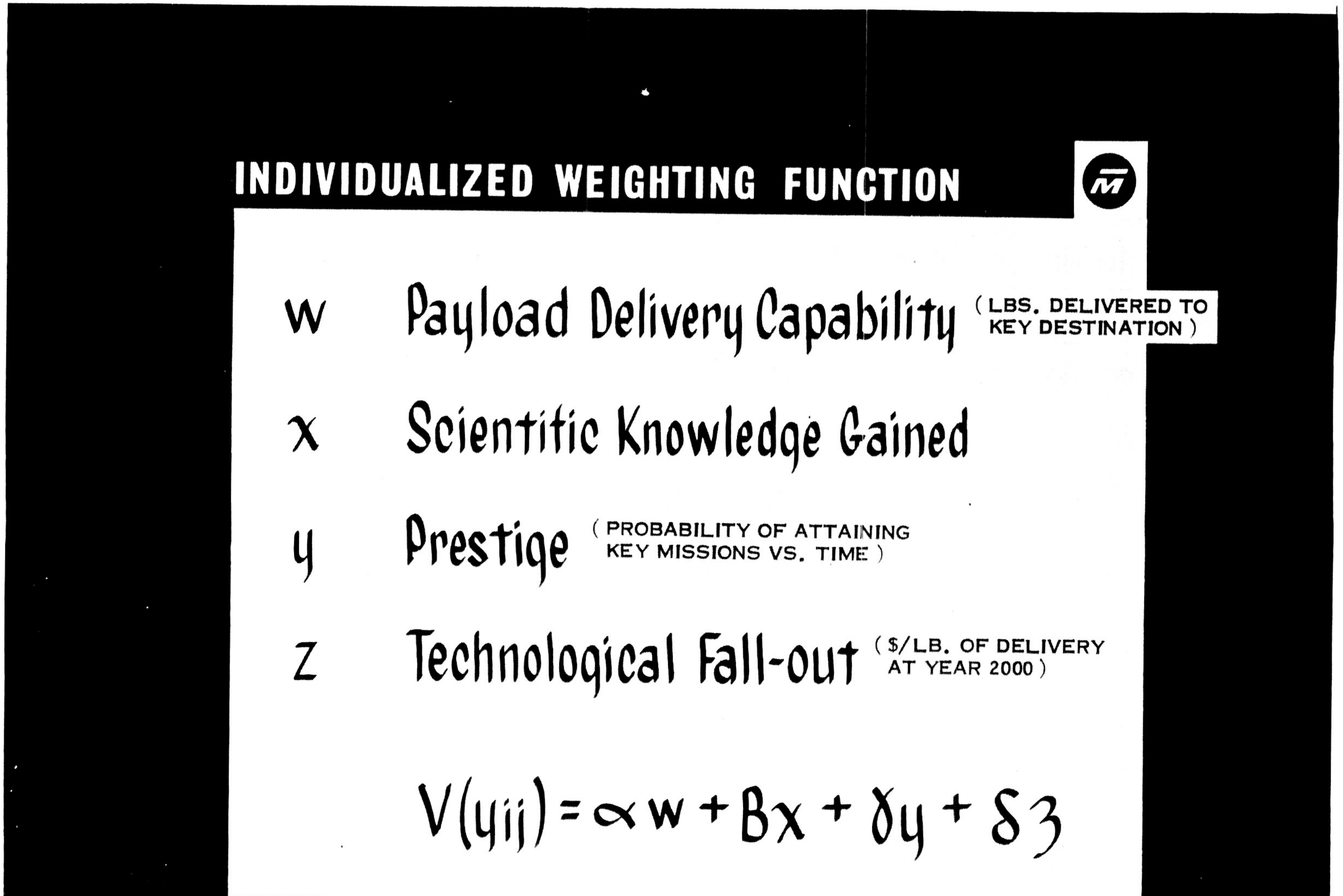


FIGURE 19. INDIVIDUALIZED WEIGHTING FUNCTION

Remaining problem areas are discussed in Figure 20. Further experience in the use of the model is required to find methods of speeding up input and output. The procedures for conference room use will be studied so that results will be more quickly available to high level planners.

Earlier, we discussed resource limitations. Resource constraints must receive further study so that they can more validly be added to the computer program.

Additional research must be done to determine how to more properly distribute R&D and other indirect costs by year. The present model also requires the user to assign his own orbital burden rates. Further study of the results of NASA Study NAS 8-11123, "NOVA Vehicles Systems Study," will produce data for a submodel to generate burden rates within the computer program itself.

FUTURE MODEL PROBLEMS AND RESEARCH AREAS

Experience Using Model

Conference Room Procedures

Resource Constraints

Value Analysis - Planning Requirements

Costing Methods

Orbital Burden Rates

FIGURE 20. FUTURE MODEL PROBLEMS AND RESEARCH AREAS

26992

PART 15

A STUDY OF THE DEVELOPMENT OF A BASIC PLANETARY TRANSPORTATION SYSTEMS MODEL

by

Dr. K. A. Ehricke
General Dynamics/Astronautics
Contract No. NAS8-11084

1. <u>Study Objectives</u>

The development of a Basic Planetary Transportation System Model (BPTSM) has the purpose of assisting NASA in the formulation and evaluation of plans for the manned exploration of space. These plans require long lead times and involve a variety of complex projects and programs in supporting roles, such as the development of appropriate Earth launch vehicles, adequate propulsion systems, unmanned deep space probe programs, planetary landing and launch vehicles, orbital laboratories and others.

The prime objectives of the BPTSM study are:

- Develop a basic methodology of model formulation.
- Develop a computer program which will represent a reasonably comprehensive planetary transportation system model for the 1975/95 period.

Secondary objectives are:

- Formulation of analytic results in terms of quantitative figures of merit.
- Demonstration of validity and utility of study results by testing the preliminary computer program on appropriate sample cases.
- Preliminary identification of new methods of computation indicated in the course of the study whose incorporation into the model would increase significantly the accuracy of the model in subsequent development phases.

On the basis of the introductory remarks above, the following guidelines have been adopted as axiomatic in developing the model:

- High degree of versatility to assure broad responsiveness to NASA's programs and evaluation requirements.
- High degree of flexibility to permit high-accuracy problem handling with many inputs; or to permit quick-survey problem handling with only a few important inputs.
- Capability of high accuracy.
- Broad usability, leaving the customer free to readily vary inputs and individual input data, so as to apply the latest data in a given field to the computer analysis.
- Growth potential.

2. Concept

To be of maximum use, the BPTSM should furnish two types of outputs: engineering-type outputs and management-type outputs. Fig. 2-1 shows the normal sequence of events. A problem, recognized by engineering and management, requires first concepts for its solution. Initially, a variety of concepts has to be considered. To permit objective and consistent evaluation, these concepts must be broken down into characteristic parameters and measured in terms of specific criteria. The evaluation results permit selection of one or several preferred alternatives from the engineering point of view. If there were only one possible engineering solution to the problem, further evaluation would be pointless and management's decision would have to be in binary form: yes or no. In practically all cases, however, there are alternatives among the preferred selections. A decision among these, by definition, transcends the power of engineering resolution. It must be presented to management in the proper formulation for further evaluation and final decision.

Fig. 2-2 outlines this process in greater detail. Common, fundamental engineering inputs/outputs can be grouped into mission analysis, analysis of mission objectives and associated payloads, propulsion analysis and vehicle analysis. The first two groups deal with the requirements; the latter two groups with ways to meet these requirements. After having dealt with these fundamental engineering input/output groups, the treatment varies primarily according to two principal types of investigation: mission-oriented or program-oriented. On the mission-oriented case, the "objective" (on top of Fig. 2-2) is a particular mission type or possibly a closely related group of missions; e. g. an early Mars mission (it could be fly-by (and there are variations to this one) or capture (again there are variations) or even a modest surface excursion attempt). In the program-oriented case a sequence of missions is considered; where the missions are generally not similar. In other words, emphasis is on a space program which represents a section of the national space program; or even on the national space program as a whole. Examples are the treatment of the Earth orbital space station program; or the development of bases on Mars through a series of missions; or the exploration of the solar system either by instrumented probes or by manned missions or by both through a series of increasingly ambitious missions.

In the mission-oriented case one has to integrate the mission requirements with the vehicle and the vehicle with the payload requirements.

In the program-oriented case one has to go one step further. In the mission-based case one synthesizes the individual missions into a sub-program and sub-programs into a program; e. g. lunar missions into a lunar sub-program; planetary missions into a planetary sub-program; lunar and planetary sub-programs into a major portion of the national space program. Such a case is illustrated in

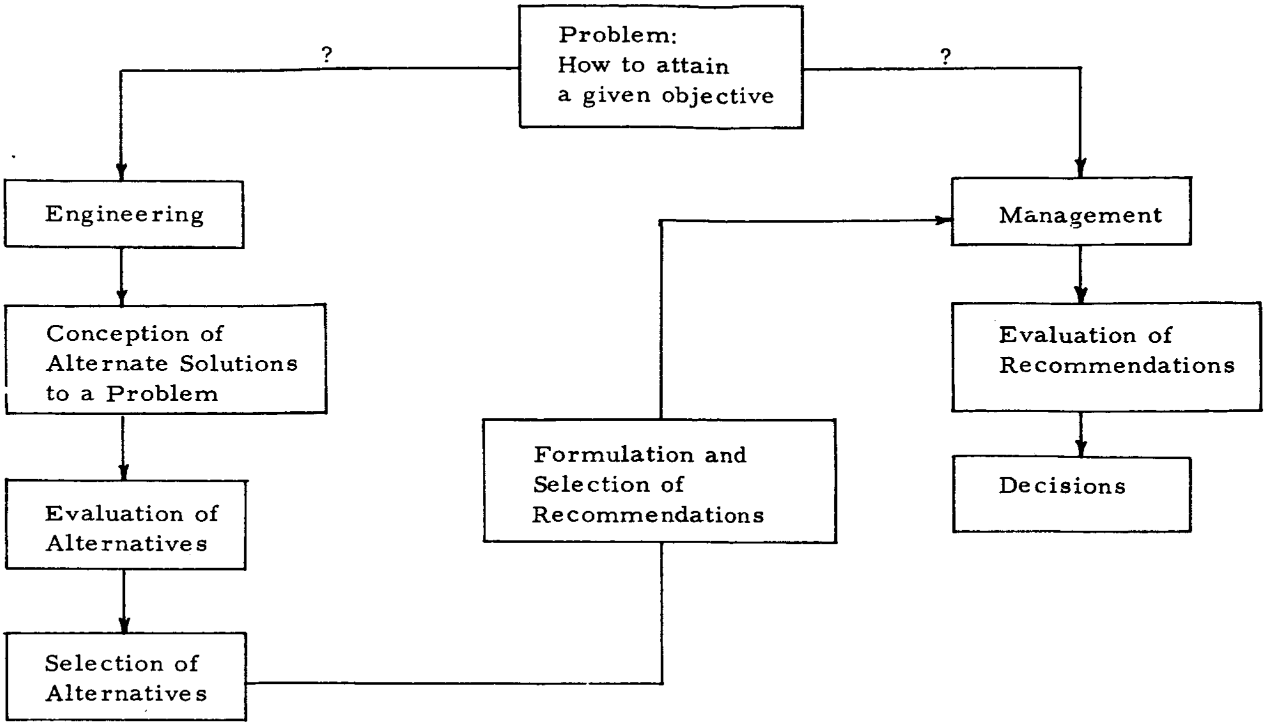


Fig. 2-1 SYSTEMATIC PROBLEM HANDLING: CONCEPT TO DECISION

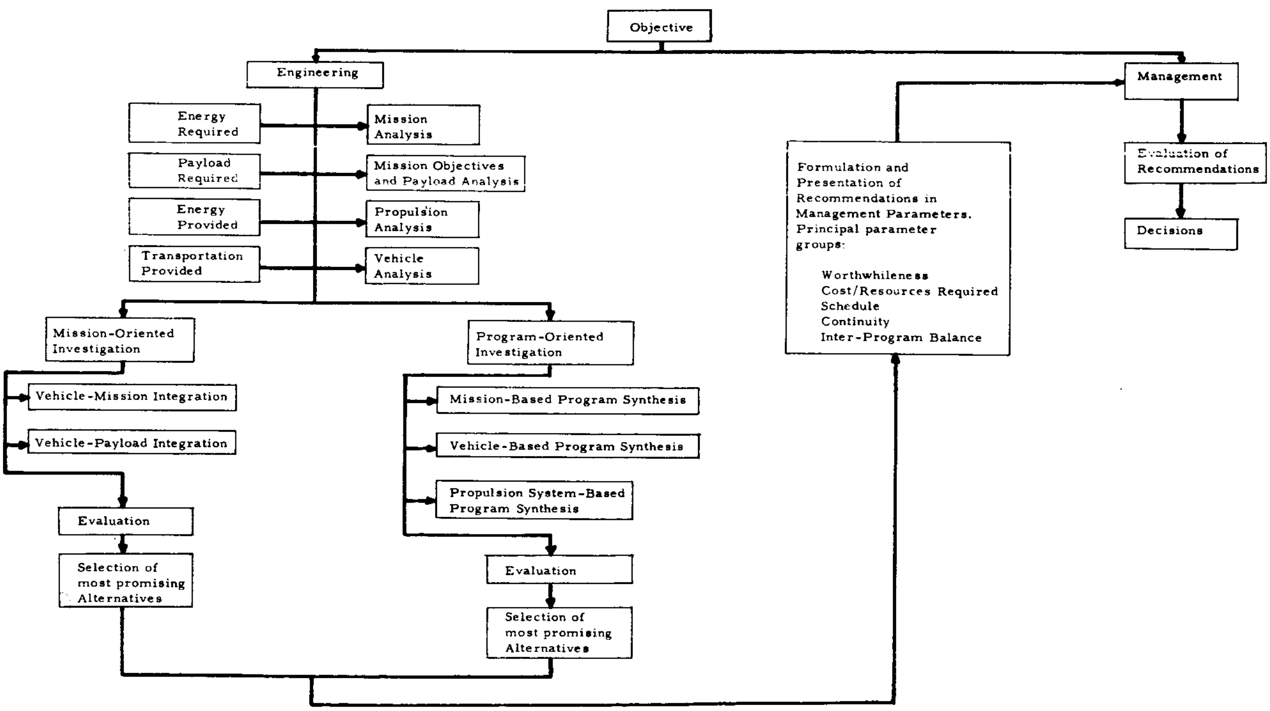


Fig. 2-2 ANATOMY OF EVALUATION MODEL FOR SPACE TECHNOLOGICAL OBJECTIVES

Fig. 2-3. It is apparent, however, that the process can be reversed. Beginning with a specified sub-program or beginning with a specified program and sub-program arrangement, one investigates particular propulsion systems or vehicle systems and evaluates their potential in the framework of the selected programs. These are the vehicle-based or propulsion system-based program syntheses. The results lead to a specification of individual missions, vehicles and payloads as dependent variables.

Whichever path is followed, the results must be evaluated in terms of meaningful parameters (evaluation criteria). This process is illustrated in Fig. 2-4. Evaluation criteria are comparable to units of measurement and as such are neutral. They make it possible to compare the attributes of the individual alternatives objectively and consistently. Once made comparable in terms of evaluation criteria, they can be processed through a rating procedure which permits evaluation of the most promising alternatives. This rating process consists essentially of two parts: a value factor is assigned to each of the above evaluation criteria, to reflect the relative value of each criterion with respect to the overall objective; a weighting factor is assigned to each attribute, relevant to the respective criterion, to indicate the relative importance of the particular attribute.

Finally, the formulation and presentation of the results is of considerable importance and much emphasis has been devoted to it in this study, as is shown below in this summary.

In conclusion of this section, the observation is made that the frequently formed emphasis on the model construction is accompanied by a tendency to underrate the importance of input data. But, any model is only as useful a management and decision-supporting tool as its input data are accurate and reliable. Careful study and evaluation of data inputs is therefore of critical importance. Wherever possible, supporting studies have been made to approach this goal with respect to application of the computer model to planetary transportation.

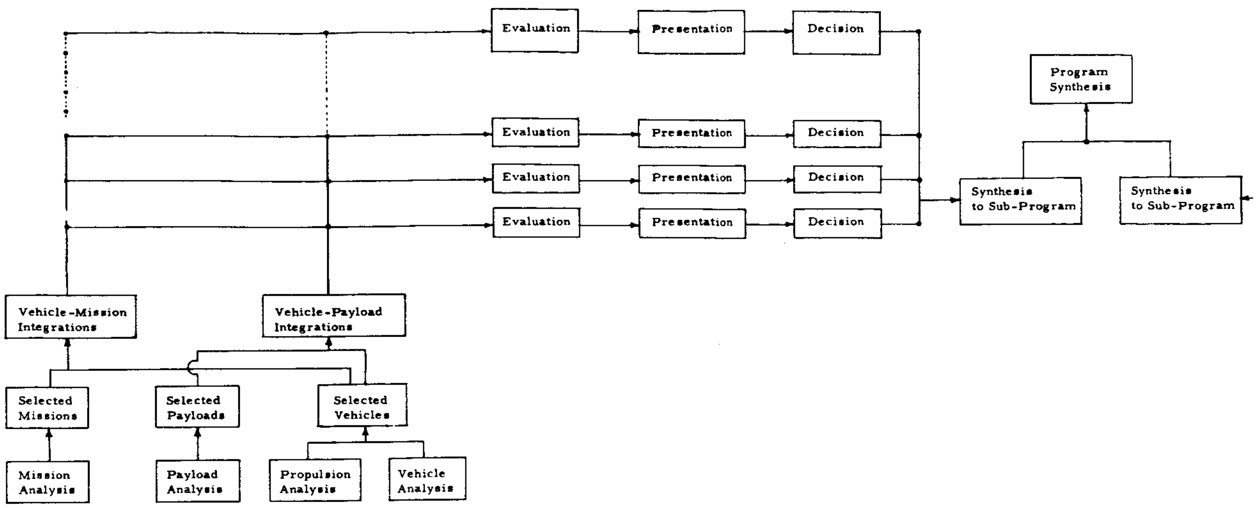


Fig. 2-3 PROGRAM SYNTHESIS FROM SUB-PROGRAMS. SUB-PROGRAM SYNTHESIS FROM INDIVIDUAL MISSION INTEGRATIONS (MISSION-BASED PROGRAM SYNTHESIS)

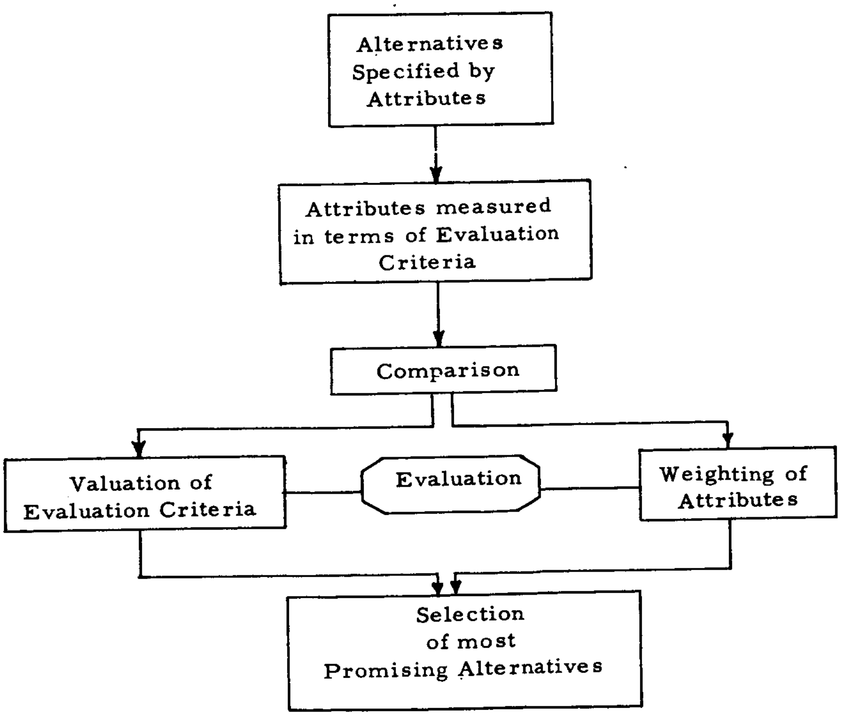


Fig. 2-4 EVALUATION AND SELECTION

3. <u>Engineering and Management Models</u>

Because of the complexity of the overall problem, versatility and ease of handling of the BPTSM requires extensive modularization, permitting a coupling of modules in any arrangement needed for the particular investigation.

As a first step, the overall model has been divided into two classes: the <u>engineering model</u> and the <u>management model</u>. The engineering model illustrated in Fig. 3-1 emphasizes the engineering aspects of the transportation system. Its objective is to define promising avenues of approach and to eliminate from further consideration engineering alternatives which are not attractive or even feasible; or alternatively, its objective is to rate various engineering approaches relative to their engineering and technological attributes before pursuing them further. The management model illustrated in Fig. 3-2 emphasizes those aspects which are primarily of management concern. This, incidentally is not to mean that they are not of concern to the engineer also. The distinction between engineering and management made here, relates to a distinction between disciplines, rather than personalities. In fact, the management model too will be handled largely by engineers in their attempt to express engineering recommendations in terms of parameter which are of greater interest and meaning to management personnel than celestial mechanics, propulsion physics etc. In addition, if looked at as a tool by management personnel, the BPTSM should be free of modules and sub-routines of little or no interest to management evaluations and decision-supporting inputs. From this viewpoint, too, modular separation of the engineering portion from the predominantly management portion of this model is of advantage.

The two parts, however, can be coupled and treated as one model. On the other hand, each model part is modularized further to guarantee maximum versatility, flexibility and growth potential in the engineering as well as the management model.

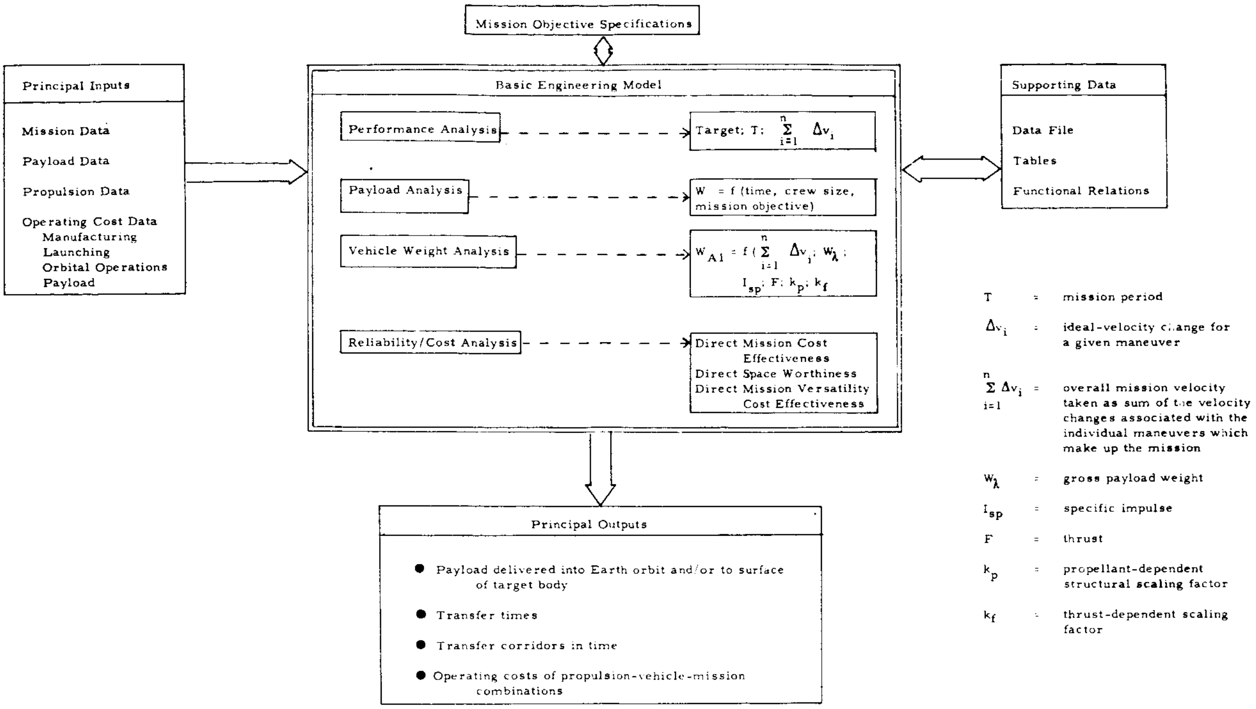


Fig. 3-1 ENGINEERING PART OF THE BASIC PLANETARY TRANSPORTATION SYSTEM MODEL

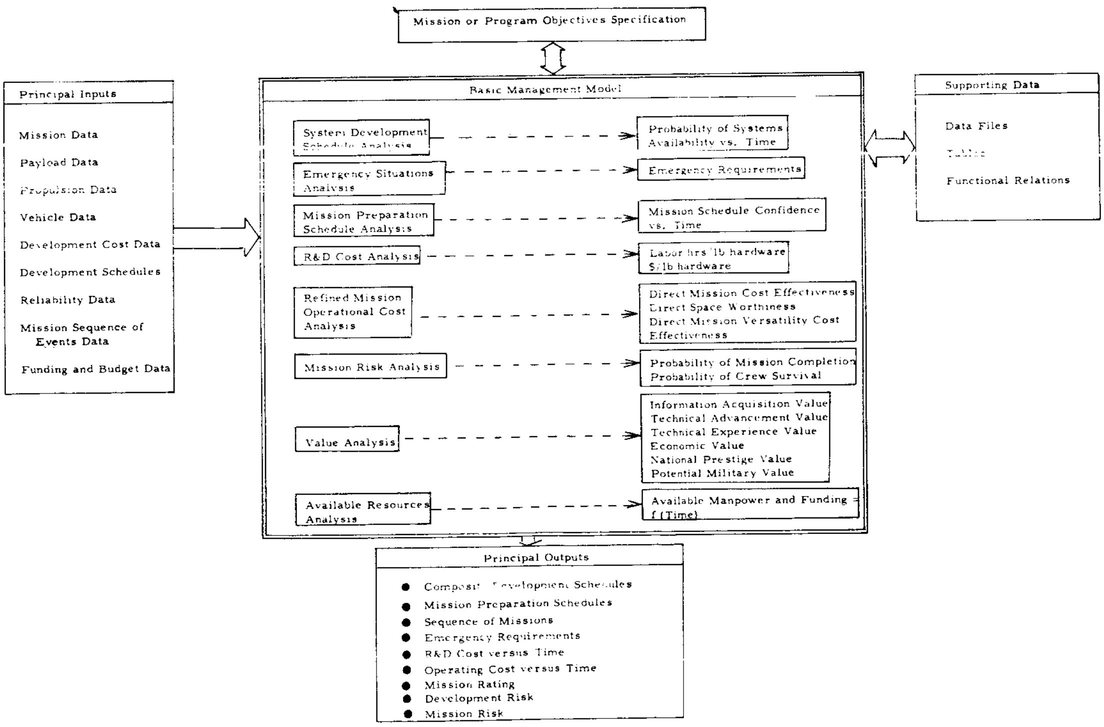


Fig. 3-2 MANAGEMENT PART OF THE BASIC PLANETARY TRANSPORTATION SYSTEM MODEL

4. Principal Input-Output Areas (Sub-Models)

Regardless of the orientation of the model toward predominantly engineering or management aspects, the principal input/output areas have been grouped together and provided with code numbers for ready identification and modularization (Fig. 4-1). The model complex is broken down into nine major input/output areas, ranging from mission analysis to resources-based program synthesis. To ascertain ample growth potential, each of the major subject areas is provided with a 2-digit number space (e.g.; for mission analysis 100 - 199, etc.) which in turn is divided into 1-digit number spaces (e.g. 110 - 119, etc.).

Number space 0 - 99 has been reserved for extraterrestrial activity specifications and all 1-digit spaces but the last one (090 - 099) have been defined. The extraterrestial activity specifications provides the frame of of reference for the mission analysis (100) and the mission objectives and payloads analysis (200) and at least sets the stage for the selection of propulsion systems (300) and vehicle systems (400). The last of the singular input-output areas is the available resources analysis (500). The previously (Sect. 2) discussed mission-based integrations of vehicle and mission and of vehicle and payload occupy code number spaces 600 and 700. For program-oriented syntheses the primarily technologically oriented cases are treated under code number 800 and the primarily resource oriented cases under 900.

It is apparent that there are various ways of approaching subject 100 through 500. Taking 100, for instance, mission analysis can be carried out on the basis of simply determining performance plateaus; or by using simplified planar transfer analysis between two planets in circular caplunar orbits; or by 2-body analysis between the actual, elliptic and mutually inclined planet orbits; or by using multi-body transfer analysis, cutting in all significant perturbations by other planets, notably Jupiter and perturbations of escape or approach hyperbolas due to oblateness and/or mass inhomogeneity of the primary. Beyond these differences, a variety of approaches is dictated by the nature of propulsion systems, such as continuous thrust versus pulse thrust, high thrust versus low thrust, and so forth. The first nine numbers (e.g. 101 - 109, etc.) are therefore reserved for braking the mission analysis down into basic approaches, some simpler, some more complex, some very complex. The next nine numbers (110 - 119, etc.) are reserved for definitions and programs. For example computer programs will be described and detailed, functional relationships listed or tables catalogued in the number space 110 - 119 and accordingly for 210 - 219 and others. The series 120 - 129, 220 - 229 etc. is set aside for the establishment of facts; that is flight path computations, survey of relevant payloads (sub-divided into establishment and maintenance phases of a given installation, state-of-the-art surveys of propulsion systems, determination of design characteristics of vehicles and survey of present and, based thereupon, estimated future resources. The 30-series (130 - 139, etc.) is oriented toward interpretation of the facts gathered or computed in the 20-series, namely, interpreting the results of flight path calculations in terms of performance, requirements and, analogously

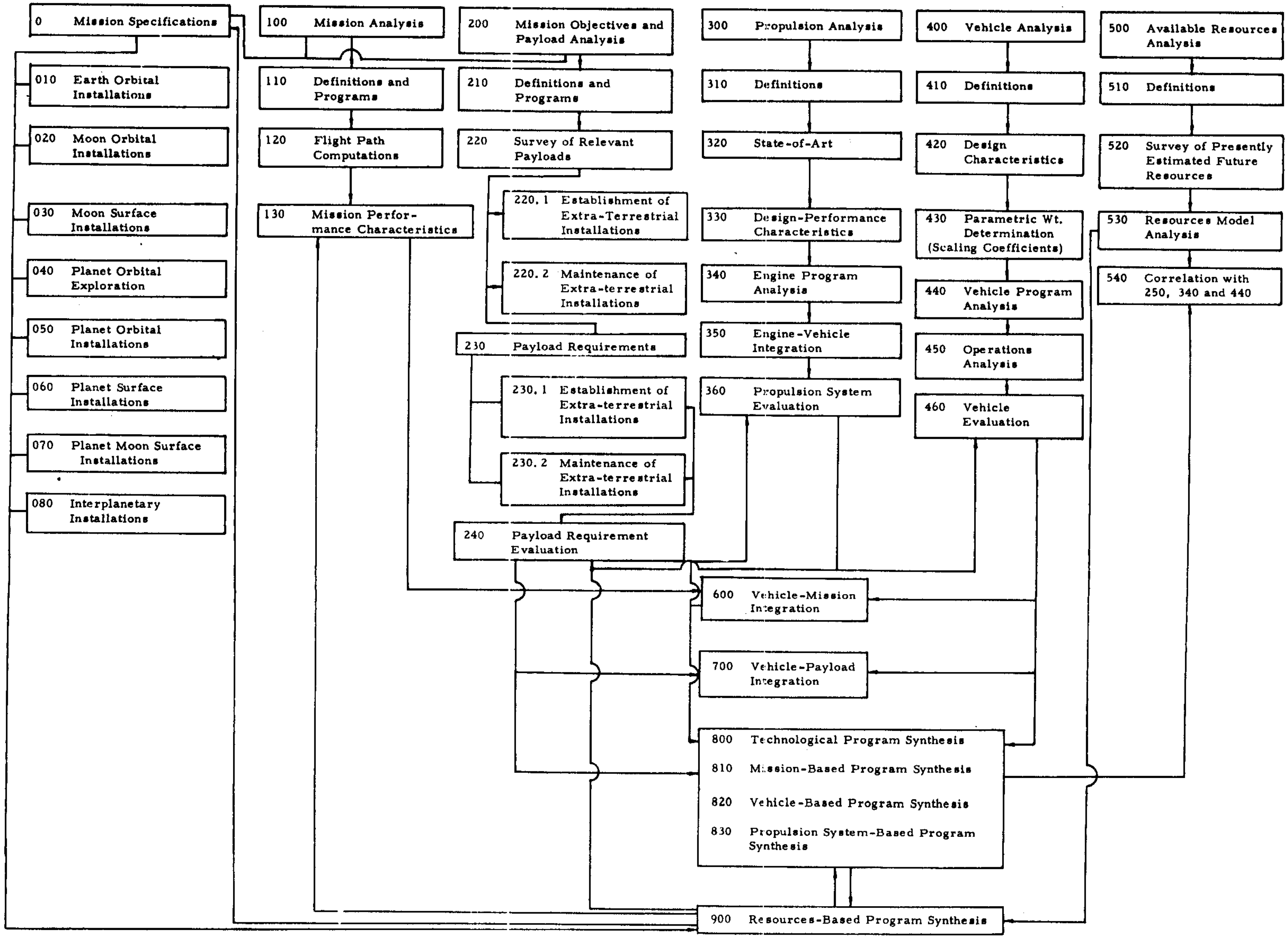


Fig. 4-1 PRINCIPAL INPUT/OUTPUT AREAS OF THE BASIC PLANETARY TRANSPORTATION SYSTEM MODEL (Sub-Models 0 to 900)

develop design and performance characteristics for engines, parametric weight data for vehicles and development of a resources model. The 30-series is significant, inasmuch as it represents inputs into the computer model. Proper presentation is, therefore, of importance. The 30-series definitely represents the terminal point for 100 and 500. For 200, 300 and 400 this is not necessarily so; or at least there the 30-series does not necessarily represent the only input. The potential necessity for additional information in the area of development requirements and operational requirements, and for evaluations on the "local level" must be recognized. The 40-series, therefore, deals with aspects of the program analysis of the respective systems or subsystems. The 50-series is set aside for aspects of systems integration or operations analysis. The 60-series is oriented toward system evaluation on the "local level". These six phases assure in most cases as complete coverage of all relevant aspects as appears necessary for BPTSM. In many cases, it will be perfectly acceptable to omit several phases.

The results of the individual analyses are then fed into the composite phases, namely, mission-based integration and program synthesis.

The terms operations analysis and program analysis used above and in Fig. 4-1 are defined in more detail in Tab. 4-1. From the items which each subject contains it is apparent that not all apply in every case and that some apply only to the areas 700, 800, 900, others only to the areas 200, 300, 400; and, again, some apply to both, on different levels of comprehensiveness.

OPERATIONS ANALYSIS	PROGRAM ANALYSIS
• Ground Operations	• Critical (Pace-Setting) Developments Analysis
• Orbit Delivery Operations	• Ground & Flight Test Development Program Analysis
• Orbital Operations	• Development Schedule Analysis
• Mission Operations	• Cost Analysis
• Emergency Operations	• Mission Value Analysis
• Reliability Analysis	• Sequence of Missions Analysis
• Orbit Delivery Success Analysis	• Program Integration Evaluation
• Orbit Operations Success Analysis	• National Space Program
• Orbit Departure Success Analysis	
• Mission Success Analysis	

Tab. 4-1 DEFINITION OF OPERATIONS ANALYSIS AND PROGRAM ANALYSIS

4.1 Extraterrestrial Activity Specifications (EAS)

The extraterrestrial activity specifications are outlined in Tab. 4-2 in somewhat greater detail.

The purpose of the EAS in BPTSM-program is to serve as pre-selector for the computer in the areas 100 through 400, inasmuch as certain mission profiles, payloads, propulsion systems and vehicle systems may be eliminated in advance. For example, selection of 010-E (Earth orbital installations establishment)will automatically eliminate low-thrust or pulse-type flight mechanics and associated propulsion systems and vehicles; and it will also eliminate certain payloads; whereas EaOLF-M does not necessarily exclude low-thrust considerations, since low-thrust vehicles may be used as taxis or as coast guard vehicles for cislunar or even planetary rescue missions. Eventually, the value of each relevant vehicle-propulsion system approach relative to each specified mission can be assessed, the best one or two approaches (from the engineering and management standpoint) selected and the mission-dependent computer input data reduced considerably as a result. Thus, as the BPTSM-program structure does not necessarily become more complex, as it is refined progressively; there are also trends which, if utilized, will cause it to become simpler. After all, this program can only have the ultimate objective of serving man's decision-making capability, not replacing it. Good servants are non-complex and oriented toward what is important to their masters. So, the computer program development must strive toward ultimate simplicity without sacrifice of comprehensiveness and reliability, rather than complexity. Intuition assures us that the ultimate solutions and choices will be simple - as they always are - once the complex period of digestion of masses of data is accomplished to everybody's satisfaction; to wit: liquid propellant selection had the following (more recent) history: progression from Lox-diluted alcohol in German V-2 to Lox-kerosene in American missiles in 1950-1956 period; massive propellant evaluation studies with enormous quantities of data, covering almost every conceivable combination under the Sun in 1956-1958 period; settling on O_2/H_2 for upper stages of Earth launch vehicles and one non-cryogenic combination for liquid fueled, second-generation missiles. Shear economic compulsion will enforce similar ultimate simplicity as far as propulsion systems and vehicles, as well as some mission profiles are concerned.

The EAS are concerned with the transport objective, not with specific mission profiles or payloads. They are, however, indicative of the energy levels required and of the transport volume in terms of payload quantity and frequency of delivery (mission frequency).

All Earth orbital installations imply the commitment for maintenance and supply. Somewhat arbitrarily, the terms used in 011 through 015 have been assigned the following implications:

0 Extraterrestrial Activity Specifications (EAS)

- - E = establishment of installation (e.g. 051-E; or MoB-E)
- - M = maintenance of installation (e.g. 063-M; or MoC-M)
- - P = special personnel traffic to and from installation (e.g. 012-P)

010 Earth Orbital Installations

- 011 Laboratory (EaOL)
- 012 Space Station (EaSS)
- 013 Launch Facility (EaOLF)
- 014 Base (EaOB)
- 015 Cislunar Rescue Station (EaCRS)

020 Moon Orbital Installations

- 021 Reconnaissance Station (MoRS)
- 022 Launch Facility (MoLF)
- 023 Base (MoOB)

030 Moon Surface Installations

- 031 Surface Excursion (MoSE)
- 032 Base (MoB)
- 033 Colony (MoC)

040 Planetary Orbital Exploration

- 041 Fly-By
- 042 Capture
- 043 Planet Moon

050 Planet Orbital Installations

- 051 Reconnaissance Station
- 052 Launch Facility
- 053 Base

060 Planet Surface Installations

- 061 Surface Excursion System
- 062 Synodic Base
- 063 Long Term Base
- 064 Colony

070 Planet Moon Surface Installations

- 071 Surface Excursion System
- 072 Synodic Base
- 073 Long Term Base
- 074 Colony

080 Interplanetary Installations

- 081 Asteroid Exploration
- 082 Asteroid Long-Term Base
- 083 Comet Exploration
- 084 Heliocentric Orbit Installation

Tab. 4-2 EXTRATERRESTRIAL SPECIFICATIONS

B. A. C.

011 Laboratory (EaOL): Small installation ($\lesssim$ 100 t)
012 Space Station (EaSS): Intermediate installation (100-450 t)
013 Launch Facility (EaOLF): Intermediate/large installation (100-2000 t)
014 Base (EaOB): Large installation (> 450 t)
015 Cislunar Rescue Station (EaCRS): Small installation ($\lesssim$ 100 t)

Weights, rather number of personnel are given, because for this group of xtra-terrestrial installations establishment in orbit is a more critical or pace-etting factor than the supply of almost any number of personnel, including values p to 1000.

For lunar, planetary and interplanetary installations, the number of persons o be maintained represents the critical parameter, because personnel supply tends o exceed the weight requirement for new parts and for replacements. For operations t the surface of other celestial bodies, surface transportation requirements and heir propellant consumption represent another major weight item which is also n the first approximation a function of the crew size. Typical crew sizes under onsideration are shown in Tab. 4-3.

Fig. 4-2 shows a simple time correlation of the various mission specifi-ations. The link between Earth and Earth space is the Earth launch vehicle (ELV). t paces the development of orbital and lunar installations.

Mission capabilities 010, 021 and 031 are therefore, paced by the ELV de-elopment. As soon as lunar installations are considered which involve commitments or supply operations (023, 032, 033) an interorbital space vehicle, the lunar supply ehicle (LSV), becomes a pace setter for the first time. Lunar supply-dependent perations, therefore, are paced by ELV and LSV, the degree of criticality being constant for both vehicles together, i.e. if the ELV state-of-the art progresses apidly, the LSV becomes less of a critical pace setter and vice versa.

Early planetary activities, 041 (Ve, Ma), 042 (Ve, Ma) are controlled y the ELV technology and, in this case from the very first mission on, by an dvanced interorbital space vehicle, namely, the interplanetary vehicle (I/V). Later missions, implying commitment to supply operations, are dominantly controlled by the development of adequate interplanetary supply vehicles, that s, by the development of adequate gaseous core reactor engines, nuclear pulse drives and/or nuclear-electric propulsion systems. Thus, a few years after perational availability of each of these drives and their associated vehicle systems new, wide field of extraterrestrial activities opens up.

Because of these interrelations, one of the major efforts in the Apollo Program consists of the development of Saturn V. Other programs will be imilarly paced by launch vehicle development or by the development of suitable ropulsion systems and vehicles for interorbital transfer.

020	Moon Orbital Installations	
	021 Reconnaissance Station	3 persons
	022 Launch Facility	10 - 50 persons
	023 Base	10 - 100 persons
030	Moon Surface Installations	
	031 Surface Excursion	2 - 10 persons
	032 Base, small	3 - 10 persons
	medium	11 - 100 persons
	large	101 - 200 persons
	033 Colony	> 200 persons
		(permanent or long-term residence; bi-sexual population)
040	Planetary Orbital Exploration	≥ 6 persons depending on planet, capture period and mission period
050	Planet Orbital Installations	
	051 Reconnaissance Station	6 - 30 persons
	052 Launch Facility	10 - 50 persons
	053 Base	10 - 100 persons
060	Planet Surface Installations	
	061 Surface Excursion System (Mars, 30 - 60 days)	2 - 20 persons
	062 Synodic Base (Mars; 200 - 300 d)	10 - 40 persons
	063 Long Term Base (Mars), small	40 - 100 persons
	medium	100 - 200 persons
	large, or network of small & medium bases	200 - 1000 persons
	064 Colony	> 1000 persons
070	Planet Moon Surface Installations	Approximately as for 060
080	Interplanetary Installations	
	081 Asteroid Exploration	10 - 20 persons
	082 Asteroid Long-Term Base	20 - 100 persons
	083 Comet Exploration	6 - 12 persons
	084 Heliocentric Orbit Installation	10 - 100 persons

Tab. 4-3 NUMBER OF PERSONNEL FOR VARIOUS EXTRATERRESTRIAL ACTIVITIES

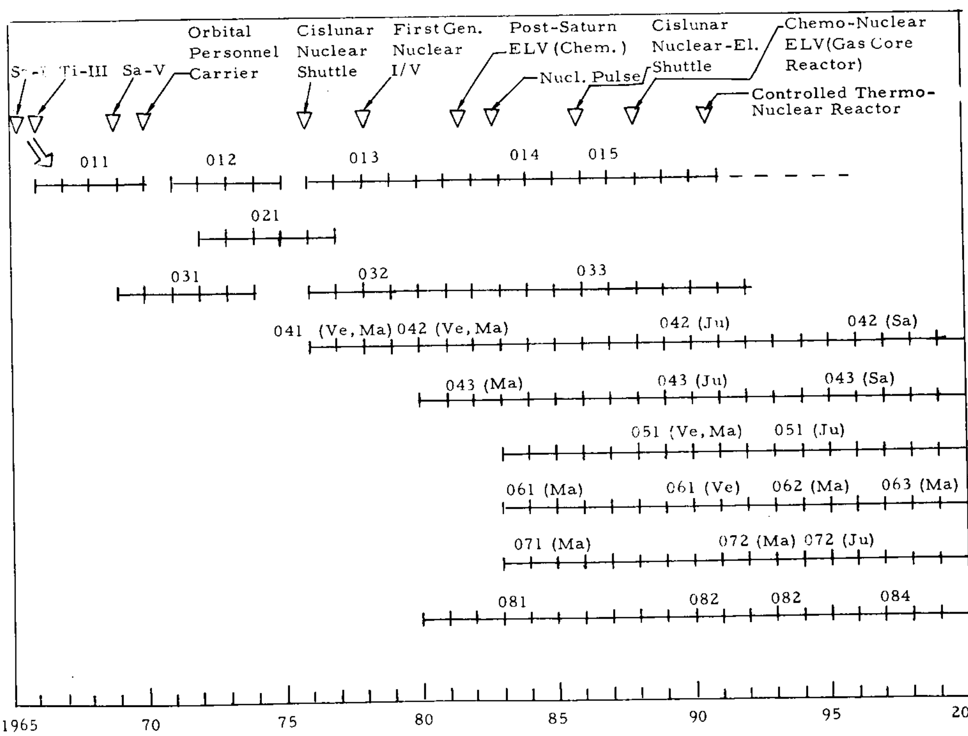


Fig. 4-2 EXAMPLE FOR RELEVANT TIME PERIODS FOR VARIOUS EXTRATERRESTRIAL ACTIVITIES

Fig. 4-3 illustrates schematically the interaction between and correspondance of vehicle development and extra-terrestrial activities. The figure is concerned with manned operations and, therefore, does not show the effect of vehicle development on the instrumented probe program. The upper line in the chart shows the overall NASA activity level which can be measured in funding dollars, man-hours, fraction of the gross national product or fraction of the national budget. Inside the envelope, the upper portion indicates schematically the "dents" made by NASA's various vehicle and engine developments. These dents are usually preceded for many years by low-level efforts in the realm of feasibility studies, laboratory experiments, state-of-the-art developments, i.e. advanced research and technology efforts. When the strips flare into large areas (high levels of effort) an engine or vehicle development is indicated. The possibility of fluctuating engine and vehicle development funding is indicated. The correlation with years is merely an illustration and should not be interpreted as proposal or result of detailed analysis. It illustrates what one must expect to find as result of model analyses. There is a large variety of potentially promising propulsion systems. On the other hand, there are only three principal areas of extra-terrestrial activity which can only develop in response to what happens in the propulsion and vehicle field. Marginal capabilities have no lasting significance, since they are incompatible with compelling economic constraints. At best, they have a transient significance as means of early exploration, as needed trail blazers for triggering more adequate developments and/or as national prestige boosters in which respects they may be quite useful and important. Mercury, Gemini, Apollo and possible early planetary mission projects belong into this category. Orbital operations will for the first time have a chance of being placed on a permanent basis following the development of Saturn I and IB and of Titan III (not shown here, since not a NASA development). The potential possibilities offered to the space station development may not be fully exploitable until the development of a recoverable orbital personnel carrier is completed. Finally, the availability of an advanced recoverable orbital personnel carrier and of a larger chemical post-Saturn ELV can raise still further the level of permanent Earth orbital activities. Apollo becomes possible as a consequence of Saturn V. As steady lunar activity, however, must await the development of an adequate cislunar shuttle vehicle and from thereon can be boosted further only by such vehicles as the chemical post-Saturn ELV, a first-generation nuclear pulse and, finally nuclear-electric cislunar shuttle vehicles, gaseous core reactor engine powered cislunar shuttles and/or the chemonuclear post-Saturn, consisting, in this example, of a chemical lofter and a gaseous core engine powered upper stage. Even after only some of these developments have been accomplished, the possibilities of economic lunar operations are virtually limitless. In the final decade of this century, Moon should virtually have become Earth's front porch.

Early planetary flights appear to require for all, but fly-by missions a nuclear propulsion module to amend the, for these purposes, marginal capability of Saturn V. Clearly, no economic sustained planetary operations are possible before the advent of at least a chemical post-Saturn and of either nuclear pulse or the gaseous core reactor engine or a suitable nuclear-electric engine. It appears that, for planetary applications, nuclear pulse and gaseous core reactor may be ready earlier than nuclear-electric systems.

Finally, with the advent of an advanced nuclear pulse engine on the one hand and the merging of the principles of the gaseous core reactor, the nuclear-electric drive and controlled thermodynamic reaction ($He^3 + D \rightarrow He^4$ fusion), the possibilities of economic sustained planetary operations in the entire solar system become virtually limitless. This condition may be reached by the end of this century.

Fig. 4-3 presents, in a nutshell, the basic problems and aspirations of the basic planetary transportation system model. The problem of "dovetailing" the growth of our propulsion and vehicular capability with the expansion of our extraterrestrial activities; and the aspiration of showing the most efficient and economic way of solving this problem, including the evaluation of whether the great multitude of propulsion and vehicle developments, shown in Fig. 4-3, is really necessary and, if not, where the cuts could be made most judiciously.

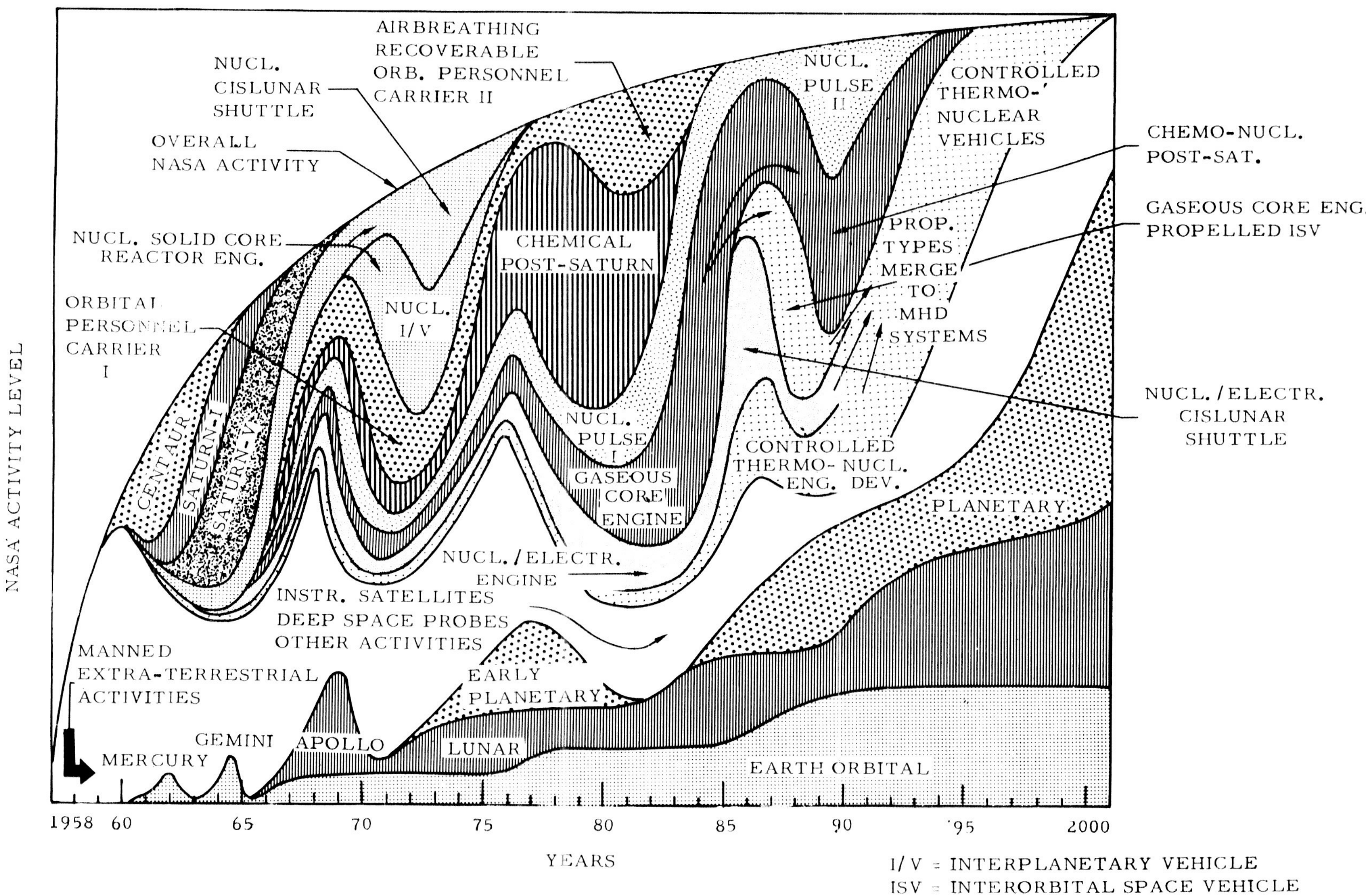


Fig. 4-3 INTERACTION BETWEEN AND CORRESPONDENCE OF VEHICLE DEVELOPMENT AND EXTRA-TERRESTRIAL ACTIVITIES

4.2 Mission Analysis

The objective of the mission analysis is to define the computer sub-modules needed for performance determinations and to provide the functional relations and tabular data where needed. The expected outputs consist of mission profile determinations (flight times, heliocentric distances, solar heat flux rates, solar and cosmic radiation levels, mission velocities), specification of mission windows, establishment of basic operational characteristics, such as burning time, number of starts required; and to present results in suitable analytical, tabular, graphic or nomographic form.

The mission analysis area is broken down in some more detail in Tab. 4-4. While series 120 and 130 refer to the computation and outputs proper, series 110 contains the definition of the tools and of important parameters. Without being necessarily complete or final, the 110 series is presented in Tab. 4-5 in greater detail for the interested reader. For the casual reader it is an added illustration of the reasons for dividing the overall model into an engineering and a management part. In the final reporting, series 110 will contain description of the programs and most of the analytic and tabular background.

A mission is characterized by the destination (either bodies or space regions), mission class, principal mission operations and by the mission maneuvers. Appendix A of this report contains the code designations used in connection with the mission analysis.

Based on these codes and on the information contained in Tab. 4-5, the example of a mission analysis sub-module is shown in Fig. 4-4. Shown is the schematic for computing the mission profile and performance data of a Mars round-trip mission. Mission class C-2 indicates that only one target planet is involved and that one of the transfers is bi-elliptic. Accurate central force field computation (103 in Tab. 4-5) is used as approach. LAM indicates that finite propulsion periods (non-impulsive maneuvers) are to be used, at least in planetocentric maneuvers. The main stations in the mission are Earth, Mars and Earth. Mission class specifications show circular capture at 1.3 Mars radii, perihelion brake at return flight and entry into the Earth atmosphere at maximum hyperbolic speed of 0.45 EMOS. Thus, programs 1114-I and 1114-II, explained in Tab. 4-5, are needed for the heliocentric transfer orbit computations. But first an Earth departure window (EDW) must be specified or selected by the computer on the basis of a long-period calendar, or a detail calendar. After the

100 Mission Analysis

110 Definitions and Programs

111 High Acceleration Missions (HAM)

1111 Powered Ascent
1112 Planetocentric and Selenocentric
1113 Cislunar Flight
1114 Heliocentric Flight

112 Low Acceleration Missions (LAM)

1121 Central Force Field

113 Powered All-the-way Missions (PAM)

1131 Central Force Field
1132 Trans-Pluto and Interstellar (non-relativistic)

120 Flight Path Computations

121 HAM

1211 Powered Ascent

12111 Mercury
12112 Venus
12113 Earth
12114 Moon
12115 Mars
12116 Ju-IV
12117 Sa-VI

1212 Geocentric Flight

1213 Selenocentric Flight

1214 Planetocentric Flight

12141 General
12142 Mercury
12143 Venus
12144 Mars
12145 Jupiter
12146 Saturn
12147 Uranus
12148 Neptune
12149 Pluto

1215 Cislunar Flight

1216 Heliocentric Flight

12161 Mono-Elliptic
12162 Bi-Elliptic
12163 Tri-Elliptic

122 LAM

1221 Mercury
1222 Venus
1223 Earth
1224 Mars
1225 Jupiter
1226 Saturn
1227 Uranus
1228 Neptune
1229 Pluto

123 PAM

130 Mission Performance Characteristics

(Velocity changes, mass ratios, flight times, departure windows, etc.)

131 HAM
132 LAM
133 PAM

Tab. 4-4 MISSION ANALYSIS

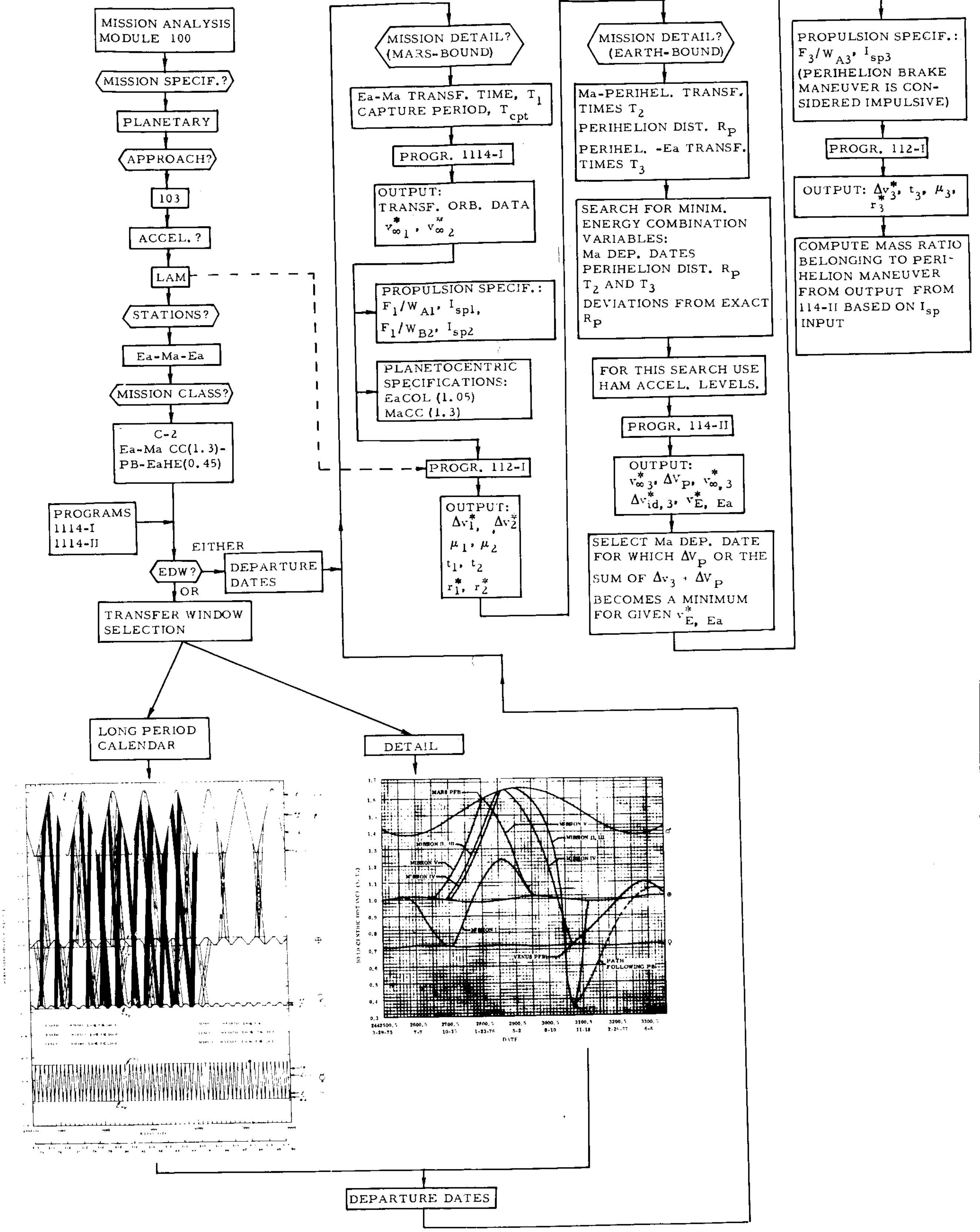


Fig. 4-4 EXAMPLE OF A MISSION ANALYSIS SUB-MODULE: MISSION CLASS C-2 TO MARS WITH LAM DEPARTURE EARTH AND MARS AND PERIHELIC BRAKE ON RETURN FLIGHT

departure window dates are set, the Mars-bound transfer time and the capture period are defined; the latter specifying the date or dates available for Mars departure. With the aid of Program 1114-I the Mars-bound transfer orbit elements and the hyperbolic excess velocities relative to Earth and Mars are computed.

Next certain propulsion systems specifications are needed for the LAM computations. These are, for the Earth departure mode; initial thrust/weight ratio and specific impulse; and for the Mars arrival mode, terminal thrust/weight ratio (W_B = vehicle weight at termination of respective powered maneuver) and specific impulse. Additional planetocentric specifications are needed, defining in this example Earth circular orbit launch at 1.05 radii and, already stated before Mars circular capture at 1.3 radii. Now the LAM program 112-I computes the actual velocity changes associated with the maneuvers (dep. Earth and ar. Mars), and the associated mass ratios, burning times and distances at beginning and end of powered flight.

Determination of the return orbit requires as inputs the time from Mars to perihelion, T_2, the perihelion distance R_p and the flight time from perihelion to Earth. Within the range of Mars departure dates, perihelion distances, flight periods T_2, the computer determines the velocity vector change Δv_p at the perihelion and at off-perihelion, but near-perihelion, points associated with a systematic variation of T_3, yielding a range of Earth entry velocities, $v_{E,Ea}$. While searching for the optimum combination for the specified Earth entry velocity, the computer switches to impulsive velocity changes, to reduce computation volume. Selecting the Mars departure window around the optimum and using the propulsion specifications of the Mars departure propulsion module, the computer changes back to finite propulsion period and determines the Mars departure maneuver parameters. Finally, the mass ratio required for the perihelion maneuver is computed, based on the specific impulse of the propulsion maneuver.

Introducing the finite propulsion period integration into the mission analysis, increases computational labor and is justified only (a) if the thrust/weight ratios are sufficiently well known at this point of the analysis, (b) if it is the intent of the particular mission analysis, to determine mass ratios and (based on these and specific impulses) the actual ideal velocities (i.e. velocities which include the effect of gravitational losses) for given transfer windows in an attempt to determine realistic "performance plateaus" for vehicles powered by low-thrust engines (e.g. low-thrust high-I_{sp} gaseous core reactor engines with radiation coolers).

In all other cases, the less laborious approach is to use HAM, i.e. impulse velocity changes and use finite propulsion later in the weight determination program where warranted on the basis of more accurate vehicle data.

A third alternative is to compute independently the actual ideal velocities for a wide range of initial or final thrust/weight ratios (for departure and arrival) and I_{sp}'s for each planet. The results, which are of permanent validity, form part of the data file available to the mission analysis sub-model and subsequently eliminate the need for further use of this program, thereby simplifying the mission analysis sub-model. Use has been made of this alternative to the extent of covering Venus, Earth and Mars with a wide range of thrust/weight ratios and specific impulses and, to a more limited extent, covering Jupiter and Saturn.

In line with constant awareness to keep the individual sub-models as flexible as possible, the mission analysis sub-model is developed as a loose conglomerate of individual programs and sub-routines which can be combined in various forms. This is shown in Fig. 4-5. The four sub-routines shown on top can be coupled to any of the principal programs which themselves are organized in three principal groups: two-dimensional transfer analysis, two-body transfer analysis, based on the method of Gauss-Lambert and special perturbations transfer analysis. Each program can be used to compute short ($<180^{\circ}$ transfer angle) and long transfer orbits. The fourth group combines the planetocentric maneuver programs. Where applicable, either impulsive or low-acceleration computer programs can be applied.

The two-dimensional analysis is least complicated but also least accurate. The special perturbations transfer analysis possesses highest accuracy such as needed for navigational purposes but involves considerably more computer time. While several hundred two-body transfers require five to seven milli-hours on the IBM-7090, a single special perturbation transfer computation may require two to four minutes. The two-body transfer analysis is the most attractive alternative on the IBM-7090, considering the objectives of the BPTSM.

With so many programs, target planets and mission profile alternatives, an enormous amount of data can quickly be accumulated. These data are arranged in tables in the data file for computer look-up. But in this form it is of little use to the engineer or manager. Therefore, attention has been given to the presentatio of essential data. Shown in Fig. 4-5 are four of the most important forms of mission analysis data presentation: velocity-transfer time correlations, velocity-departure dates correlations, velocity-capture period or velocity-mission period and presentation of transfer corridors in time. While a detailed discussion of the pros and cons of each of these methods of presentation is beyond the frame of this summary, it is noted that the first method is quite suitable for transfers to and from the outer planets; the second and third approach are useful for round-trips to and from all planets. The fourth method is perhaps the most attractive method of presentation for managerial personnel, because it shows most lucidly transfer corridors, selected for attractiveness for one reason or another. For example, it is possible to show transfer corridors which are limited to a certain maximum

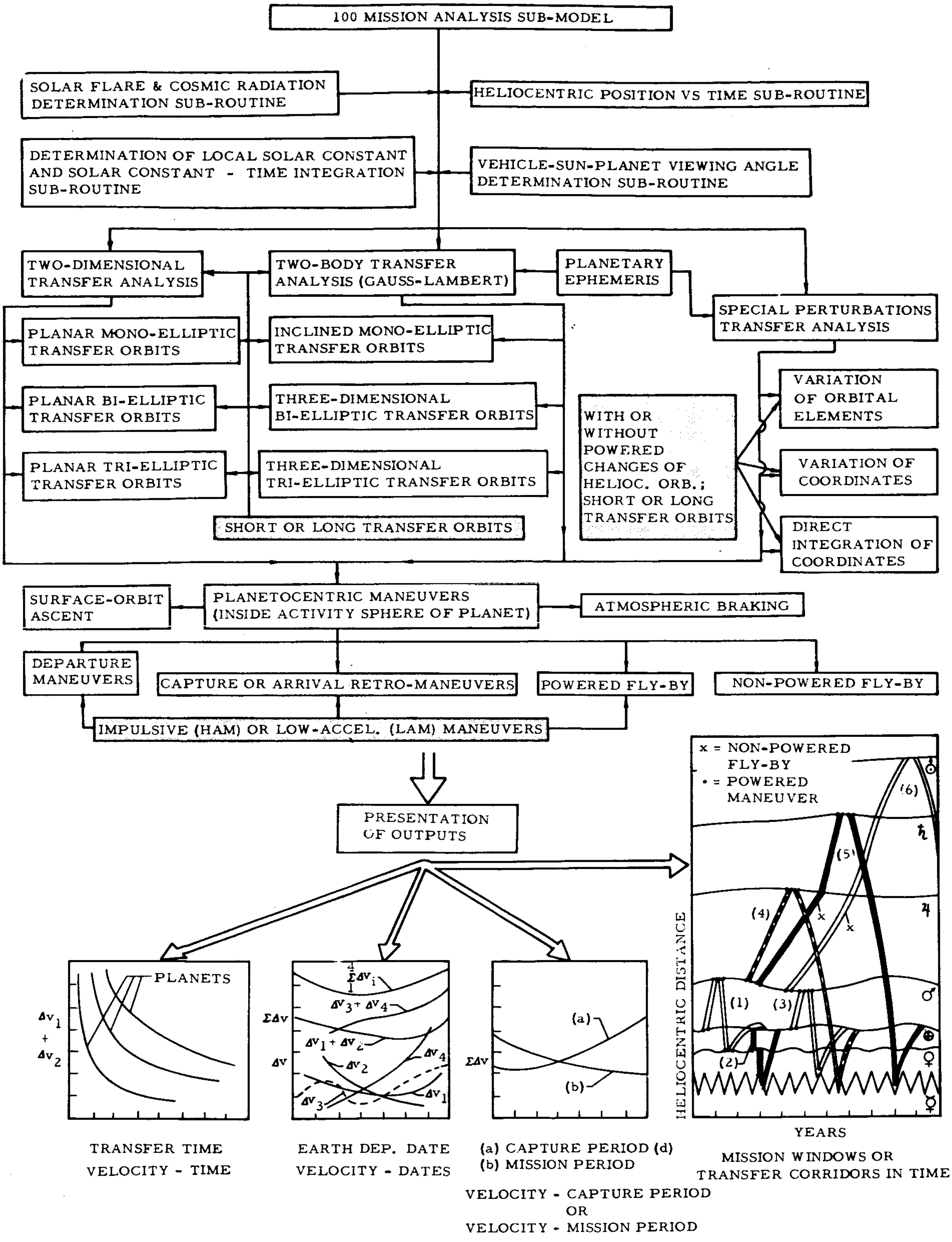


Fig. 4-5 PROGRAMS, SUB-ROUTINES AND PRESENTATION OF OUTPUTS OF THE MISSION ANALYSIS SUB-MODEL

Δv, or Δv_2 or $\Delta v_1 + \Delta v_2$; or to a certain maxi-entry velocity at Earth return; or to a maximum or minimum capture period, transfer period or mission period. Capture missions can be shown as well as fly-by missions; single-planet missions as well as bi-planet or tri-planet missions. It is obvious that in a fully developed planetary transportation system Earth no longer plays a preferred role, at least not celestial mechanically. The transportation system is by necessity heliocentrically oriented and transfers between any two planets are, in principle, acceptable (cf. cover of this report). Case (1) shows a capture mission to Mars with return to Earth via Venus where, depending upon the available energy a capture or a powered fly-by maneuver can be negotiated, resulting in additional scientific mission value as well as reduced Earth entry velocity in some cases. Case (2) shows a bi-planet capture mission to Venus and Mercury. Case (3) shows a Mars capture mission with a perihelion brake maneuver during return to Earth. Case (4) shows a typical Jupiter capture mission profile with perihelion brake. Case (5) illustrates a capture mission to Saturn, using the Jovian gravitational field in an unpowered fly-by maneuver to reduce the transfer period. Case (6) illustrates a similar mission to Uranus. A considerable variety of transfer corridor charts is in preparation, showing corridors selected from various points of view.

4.3 Mission Objectives and Payload Analysis

The purpose of this series (200) is

(a) to provide mission objective definitions and a mechanism for rating mission objectives,

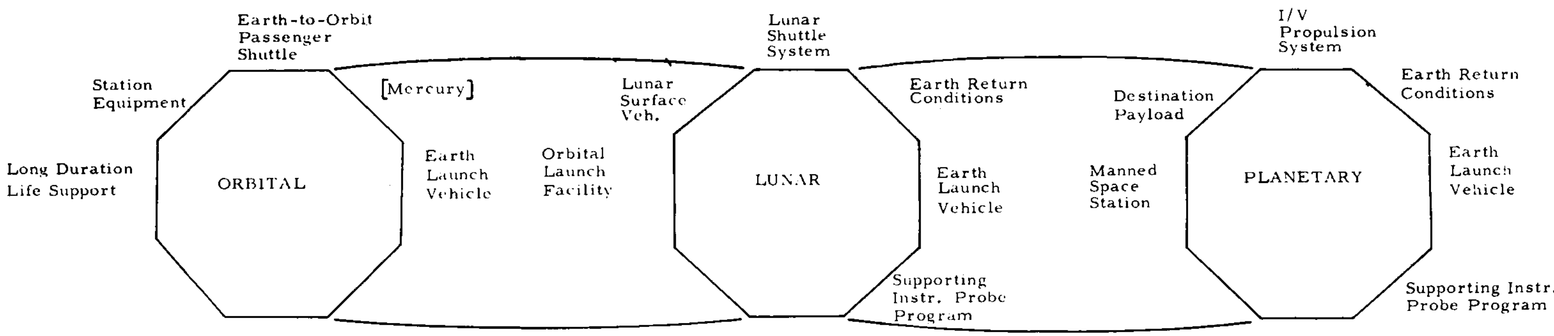
(b) to analyze state-of-the-art, pace-setting aspects, development schedules, development cost and production cost of payload

(c) to check on compatibility between transport vehicle and its intended payload.

Series 0 outlines the scale of extraterrestrial activities. Series 200 deals with the more specific objectives of these activities. Fig. 4-6 defines some of the principal planning criteria and objectives in the orbital, lunar and planetary categories. Around each category, important interfaces with advanced research and technology areas are indicated where the need for advancements in the state-of-the-art is apparent, in order to carry out the objectives indicated underneath each category.

Tab. 4-5 shows a typical breakdown of engineering-type rating parameters for the sustenance of extraterrestrial operations, i.e. the principal objective of transportation systems. The terms "primary" and "secondary" are not meant to indicate differences in importance so much as a sequence in the order of treatment. First, it is necessary to know which of the transport requirements (1) through (4), or which combination of these, is of greatest importance for the particular extraterrestrial operation, before any one or any combination of the parameters (A) through (K) can be selected as the principal modulating qualifications. Among the primary rating parameters, expendable cargo transport refers to material consumed in one form or another in the extraterrestrial facility. This includes not only food and hygienic and medical supplies, but also fuel and oxidizer consumed in various vehicles, worn clothing, worn-out equipment and losses due to evaporation and leakage. The non-expendable cargo includes housing, auxiliary vehicles, nuclear reactors, power conversion equipment and other items normally of long operating life. Transit cargo finally refers to cargo not needed at the extraterrestrial installation, but used to service customers of this installation; e.g. propellant and parts for interorbital space vehicles serviced at the particular installation. Depending on which of the four is most important, different secondary parameters will be of particular significance. For example, if (1) represents the primary transport requirement, i.e. if people

Fig. 4-6 DEFINITION OF PLANNING CRITERIA AND OBJECTIVES



Engineering R&D Lab

Geophysical Lab.

Biological & Astrophysical Lab.

Command, Control and Communication Station

Emergency Rescue Station

Orbital Launch Facility

Public Facilities

Military Reconnaissance

Military Base

Lunar Exploration Base

Lunar Scientific Res. Base

Lunar Planetary Command Control Base

Lunar Launch Facility

Lunar Public Facilities

Lunar Military Base

Orbital Planetophysical Laboratory

Exploration Base on Surface of Planet or Planet Moon

PRIMARY RATING PARAMETERS

(1) PERSONNEL TRANSPORT

(2) EXPENDABLE CARGO TRANSPORT

(3) NON-EXPENDABLE CARGO TRANSPORT

(4) TRANSIT CARGO TRANSPORT

SECONDARY RATING PARAMETERS

(A) NUMBER OF PERSONNEL AT BASE MAXIMIZED

(B) AMOUNT OF CARGO DELIVERY MAXIMIZED

(C) MINIMUM RELIABILITY OF TRANSPORT SPECIFIED

(D) RELIABILITY OF TRANSPORT MAXIMIZED

(E) MAXIMUM ACCEPTABLE TRANSFER TIME TO BASE SPECIFIED

(F) TRANSFER TIME TO BASE MINIMIZED

(G) MAXIMUM ACCEPTABLE TURN-AROUND TIME SPECIFIED

(H) TURN-AROUND TIME MINIMIZED

(J) COST MINIMIZED

(K) PROPELLANT CONSUMPTION MINIMIZED

(L) OPERATIONAL LIFE MAXIMIZED

Tab. 4-5 RATING PARAMETERS CHARACTERIZING THE SUSTENANCE OF AN EXTRA-TERRESTRIAL OPERATION

Principal Objectives of Lunar Operation	Lunar Exploration	Moon-Based Research Activities (Laboratories, Observatory etc.)	Lunar Space Port for Interplanetary Flights. (Fueling, Launching Repair & Maintenance of Interplanetary Vehicles. Passenger Transfer to Cislunar Shuttle)	Lunar Public Facilities (Hotels, Hospitals etc.)
Broad Objectives of Lunar Shuttle Operation in order of Importance (for support of operation, rather than establishment of base.	3 2 1	2 1 3	4 2 1 3	1 2 3

Tab. 4-6 RELATIVE IMPORTANCE OF BROAD OBJECTIVES OF LUNAR SHUTTLE OPERATION IN SUPPORT OF DIFFERENT LUNAR OPERATIONS

are needed most or to be serviced in the installation, parameters (A) and (D) may assume particular significance; and if the destination is planetary, (F) may be added. If (J) is important also, the need for a compromise between basically conflicting parameters (F) and (J) tends to influence decisions regarding the development of propulsion and vehicle systems. If (2) represents the primary transport requirement, emphasis is likely to be on smaller payload weights than for (3) or (4), but more routine flights over longer time periods, favoring more frequent flights with smaller payloads, provided (H), (K) and (L) can be highly developed. For (4) similar requirements exist as for (2), except that larger quantities might be needed, in which case (B) and (L) would prevail over a combination of (H), (K) and (L). Where (3) is the prime requirement, transport loads may be large, but flights infrequent. In this case, (L) and (H), both tending to reduce cost indirectly, are less important than (B), a non-ambitious specification of (E), to boost (K), and (J), i.e. direct cost minimization by avoiding expensive developments or systems of high direct operating cost.

Tab. 4-6 shows an example of typical arrangements of the primary rating parameters, shown in Fig. 4-7, in the order of decreasing importance for cases of supporting four types of lunar operations.

Tab. 4-7 shows typical ratings of the primary objectives over a wider range of activities; where NM = non-military, M = military activity.

Mission objectives, as listed in Fig. 4-6, and, in consequence, primary and secondary parameters which receive highest rating, from the basis for payload specifications.

Within the vehicle-oriented frame of reference, the payload has been divided into a mission payload weight group and a flight equipment weight group. In the operation-oriented frame of reference, the payload was classified as operational payload, intransit payload and destination payload. Fig. 4-7 shows the relation of payload weight breakdown to the overall vehicle weight breakdown. Terminal payload refers to the payload weight returned to Earth (e.g. the atmospheric entry vehicle). Transport payload includes scientific instrumentation, auxiliary vehicles, reconnaissance equipment etc. In the flight equipment weight group, the flight control system summarizes the weight of all-inertial guidance, autopilot, navigation system and flight control propulsion system for correction maneuvers and attitude control together with the necessary actuators. The vehicle control, checkout and maintenance system (VCCMS) consists of sensors, intra-ship information transmission and display, control systems, remotely controlled repair systems and harness weights. The power supply and distribution system (PSDS) consists of main power generation, main power distribution and emergency power generation and distribution. The spine system includes the structure, telescoping equipment and thermal and meteoroid shields of the boom separating the life support section from the propulsion modules.

Tab. 4-7

MATRIX OF EXTRA-TERRESTRIAL OPERATIONS
AND CORRELATED TRANSPORTATION CHARACTERISTICS
REQUIRED FOR SUSTAINING OPERATIONS

Extra-Terrestrial Operation		Characterizing Parameters: First Importance	Second Importance	Third Importance	Fourth Importance
NM - Engineering R&D	Earth Orbit	1	2	3	
NM - Recon. & Exploration	Earth Orbit	1	2	3	
	Moon Orbit	1	2	3	
	Moon Surface	3	2	1	
	Inner Planet Orbit	2	1	3	
	Inner Planet Surface	3	2	1	
	Mars Moon Surface	2	3	1	
	Outer Planet Orbit	2	1	3	
	Planetolunar Surface	3	2	1	
	Interplanetary Space	2	---	---	
NM - Scientific Research	Earth Orbit	1	2	3	
	Moon Surface	2	1	3	
NM - Relay Operations	Earth Orbit	2	1	3	
	Moon Surface	2	1	3	
NM - Emergency Rescue	Earth Orbit	4	2	1	3
	Moon Surface	4	2	1	3
NM - Launch Operations	Earth Orbit	4	2	1	3
	Moon Surface	4	2	1	3
NM - Public Operations	Earth Sub. Orb. Transp.	1	---	---	
	Earth Orbit	1	2	3	
	Moon Surface	1	2	3	
M - Reconnaissance	Earth Orbit	2	1	3	
M - Base	Earth Orbit	3	2	4	1
	Moon Surface	2	4	3	1
Interorbital Warfare	Earth Orbit	4	3	2	1
Space-to-Earth Warfare	Earth Orbit	4	3	2	1
	Moon Surface	4	3	2	1

Fig. 4-7 PAYLOAD BREAKDOWN AND ALLOCATION

① In crew vehicles used for initial exploration

② In service vehicles, accompanying crew vehicles

③ In transport vehicles used for base establishment or supply

Payload Weight Groups	Operational Payload	Intransit Payload	Destination Payload
1. Terminal Payload			
1.1 Earth Entry Module	①—③		
1.2 Crew	①—③		①
1.3 Earth Entry Support System	①—③		
2. Life Support Section			
2.1 Main Personnel Modules			
2.1.1 Command Module	①—③		
2.1.2 Radiation Shelter	①—③	③	
2.1.3 Equipment Module	①—③		
2.1.4 Food Storage Module	①—③	③	
2.1.5 Service Module	①—③		
2.1.6 Living Modules	①—③	③	
2.1.7			
. . .			
2.2 Extension Modules	①—③	③	
2.3 Space Taxis	①—③		
2.4 Abort System	①—③		
2.5 Spin System	①—③		
2.6 Ecological System	①—③	③	
2.7 Dehydrated Food	①—③	③	
2.8 Emergency & Reserve Life Support	①—③		
2.9 Miscellaneous	①—③		
3. Transport Payload			
3.1 Scientific Payload	①—②	①—②	①—②
3.2 Personnel (Passengers)			③
3.3 Cargo		②	② ③
3.4 Cargo Integration		②—③	
4. Communication System	①—②—③		
5. Flight Control System	①—②—③		
6. Vehicle Control, Checkout & Maintenance System (VCCMS)	①—②—③		
7. Power Supply and Distribution System (PSDS)	①—②—③		
8. Spine System	①—②—③		

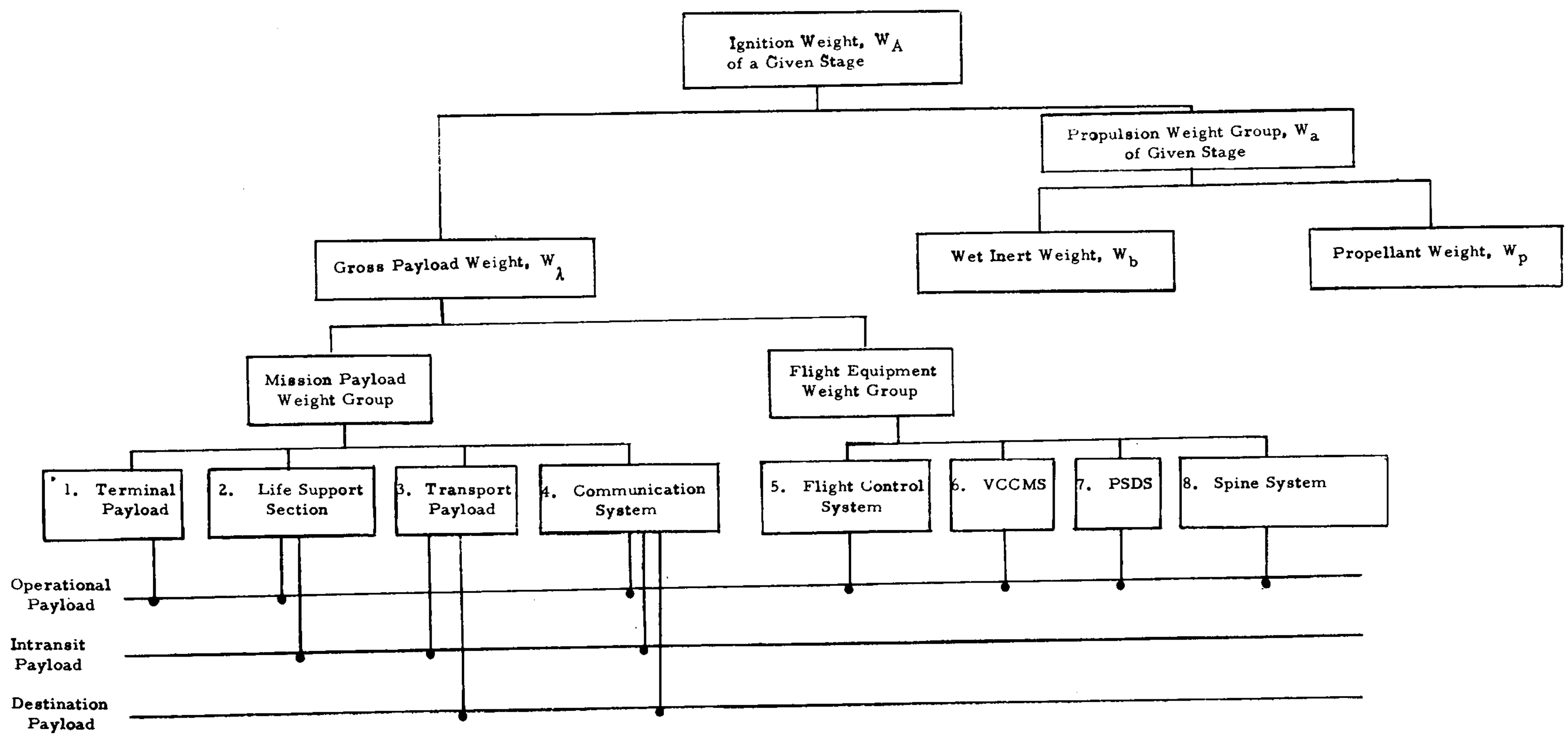
The operational payload contains all weight items and crew personnel required to operate the vehicle in the course of its mission. The intransit payload consists of all weight items needed to carry out research activities enroute and to maintain and handle destination payload (cargo and/or passengers) during transfer to the destination. The destination payload consists of cargo and personnel (= passengers) to be delivered to the destination. It is apparent that the allocation of a given payload weight item to either of the operational categories can change, as the mission objective changes. Fig. 4-8 shows the payload weight breakdown and variations in allocation for three basic types of interplanetary vehicles.

Payload determination proceeds in a systematic manner according to the breakdown shown in Fig. 4-7 and Fig. 4-9. Functional and relationships, graphs and tables are established for each item to serve as input into computer for weight and cost determinations.

The overall make-up of the mission objective and payload (MOP) submodel is shown in Fig. 4-10. The program provides a tie-in with the mission objective analysis (not shown in detail here) and permits to include or exclude various payload groups not necessarily needed for every mission (such as ascent into Earth orbit or descent to the surface of another body). All principal ecological systems are included as well as analytic relations for all important types of intransit payloads and destination payloads, from auxiliary vehicles and orbital reconnaissance equipment to landing vehicles, base equipment and surface locomotion equipment.

For longer mission, i. e. especially for planetary missions, spare parts and redundancies become a significant payload component which, in shuttle vehicles, can lead to significant reductions in destination payload capability, compared to lunar base supplies, for instance.

Fig. 4-8 RELATION OF PAYLOAD WEIGHT BREAKDOWN TO OVERALL WEIGHT BREAKDOWN



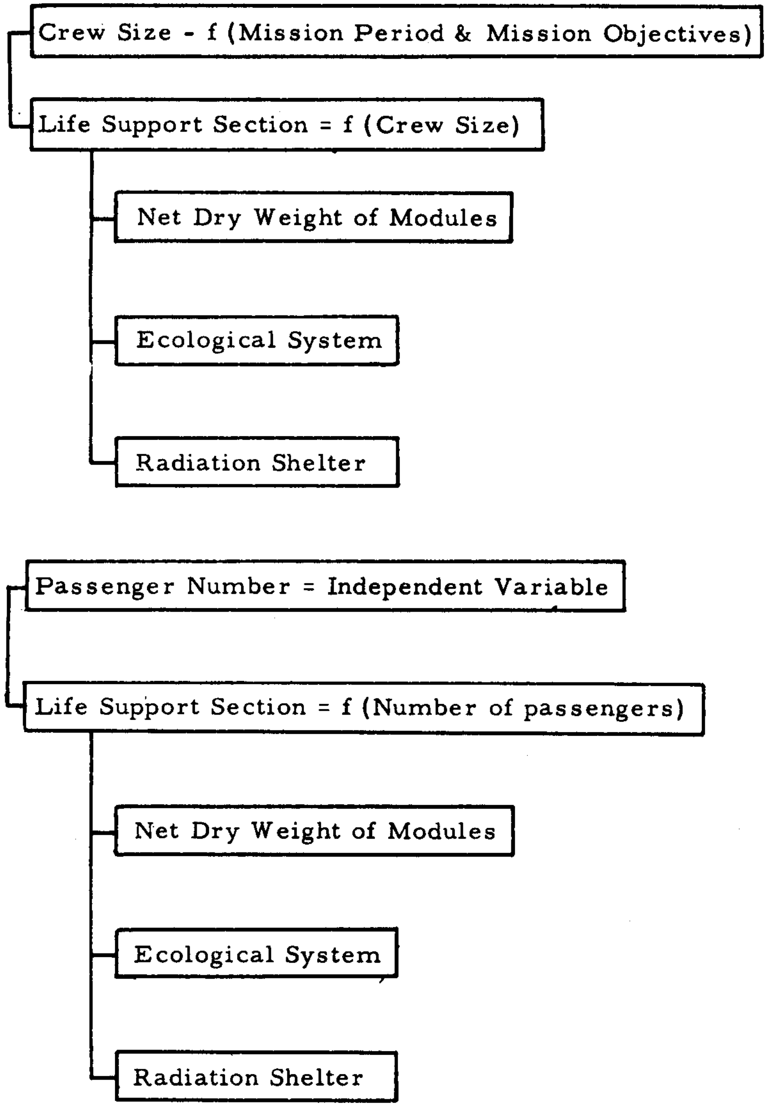


Fig. 4-9 PAYLOAD DETERMINATION

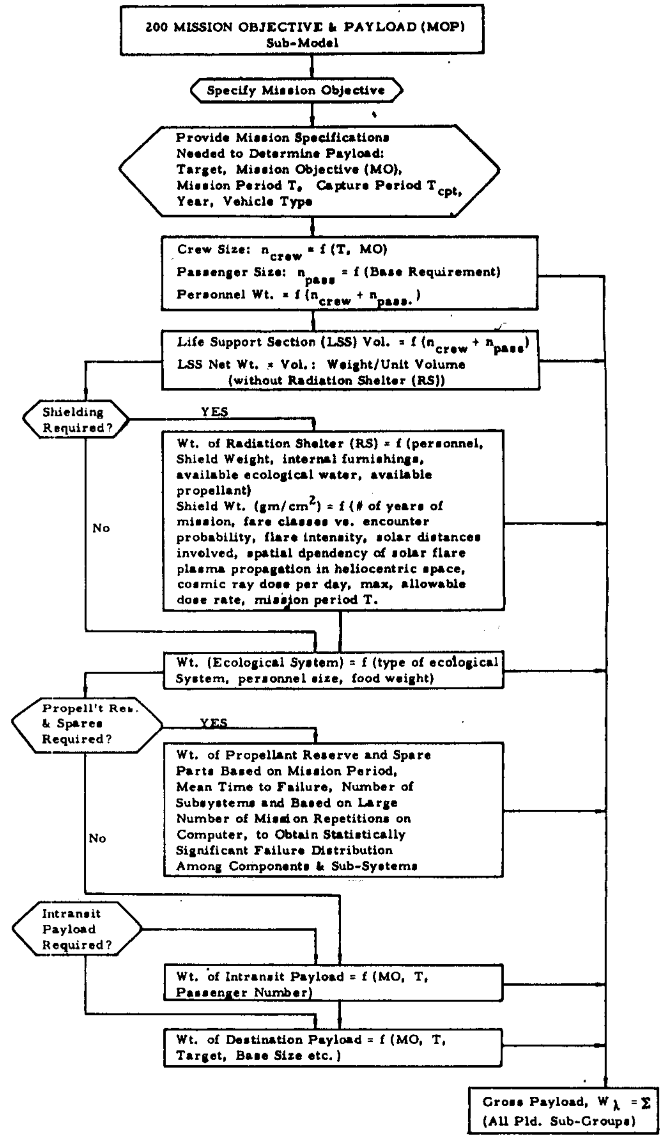


Fig. 4-10 MISSION OBJECTIVE AND PAYLOAD (MOP) SUB-MODEL

4.4 Propulsion Analysis

A more detailed breakdown of series 300, propulsion analysis, is presented in Tab. 4-8. In order to illustrate the approach the development of this sub-model, the interfaces between propulsion analysis, cost analysis and the sub-models 100 and 400 are shown in Fig. 4-11. Propulsion is the principal pace-setter in the development of a planetary transportation system. Basic inputs are, therefore, reliability of engine operation, performance, expressed in terms of specific impulse and thrust and design, expressed by the parameters k_e (engine weight to thrust ratio) and k_{ts}, the ratio of weight of the engine thrust structure to thrust of the engine. Practically all parameters for meaningful specification and evaluation of engines must involve one or more of these basic inputs.

These considerations are reflected in defining the functional relations and evaluation parameters listed sub 310 in Fig. 4-12. They define the pattern of data collection (320) to provide a systematized data file for computer analysis on display purposes. Historical data and estimates of future capabilities are included. The data cover all essential sub-systems as well, such as methods of thrust throttling, thrust vector control, feed systems, and others; as well as test facility requirements and other support requirements which may act as pace setters in the development of the particular propulsion system.

The collected data and relations are ordered with respect to two frames of reference: design performance oriented data and engine program oriented information. In this process the collected data and relationships are treated and evaluated consistently, gaps are filled by estimates or independent analyses and uniform presentations will be attempted for major engine groups, considering inherent differences in evaluating or specifying chemical, solid-core reactor on gaseous core reactor engines as compared to nuclear pulse engines or power-separated nuclear-electric engines. The data are applied to engine-vehicle integration analyses (350), many of which lead to potentially important computer programs. The final process is the evaluation of propulsion systems (360). This is the principal interface area with the other sub-models through which the adequacy of an engine system for a particular mission performance, or a particular transportation problem, or its availability (or availability with given reliability) for particular missions at specified time periods (if the space program or part of it is the independent variable) can be determined. Tab. 4-9 presents a few examples of evaluating processes leading to acceptance or rejection of a particular engine by the computer. Only one question is shown. Many questions must be considered.

An outline of the Series 300 sub-model is presented in Fig. 4-13 A large number of alternatives, including the one that submodel is not needed for certain problems, are incorporated to make the submodel adaptable to the widest possible range of problems. Tab. 4-10 shows a typical engine data file card.

300 Propulsion Analysis

- 301 Chemical (C)
- 302 Solid Core Reactor Nuclear (SCR)
  - 302.1 Graphite Base (SCR/G)
  - 302.2 Metal Base, Water Moderated (SCR/WM)
  - 302.3 Metal Base, Unmoderated, Fast Neutron (SCR/M)
- 303 Gaseous Core Reactor Nuclear (GCR)
  - 303.1 Coaxial (GCR/C)
  - 303.2 Glo-Plug (GCR/GP)
  - 303.3 Glo-Tube (GCR/GT)
  - 303.4 Single Vortex (GCR/SV)
  - 303.5 Multi-Vortex (GCR/MV)
  - 303.6 Twin Cell Vortex-Stabilized (GCR/TCVS)
  - 303.7 Vortex-Stabilized (single or multi-cell) (GCR/VS)
- 304 Nuclear Pulse Drive (N/P)
- 305 Nuclear/Electric Drive (N/E)
- 306 Controlled Thermonuclear Reactor Drive (CTR)
- 308 Air-Breathing Engines

310 Definitions (cf. Fig. 4-31 and Tab. 4-25)

320 State-of-Art

330 Design-Performance Characteristics

340 Engine Program Analysis

- 341 Development
  - 341.1 Critical Items
  - 341.2 Test Requirements (Facilities)
  - 341.3 Development Schedule
- 342 Cost
  - 342.1 Development Cost
  - 342.2 Production Cost
  - 342.3 Refurbishing Cost
- 343 Mission Value Analysis
- 344 Growth Potential

350 Engine-Vehicle Integration

360 Propulsion System Evaluation (cf. Tab. 4-24)

Tab. 4-8 BREAKDOWN OF PROPULSION ANALYSIS

Problem	Interface Area – Criterion: Problem Side	Interface Area – Criterion: Engine Side	Evaluation	Conclusion
Can engine E be applied to mission x?	$\frac{\Delta v_x}{g^*} = \tau_x$	I_{sp}, limiting upper value for $\frac{\tau}{I_{sp}}$	$\frac{\tau_x}{I_{sp}} <$ limiting value	Positive
	τ_x Initial vehicle weight, W_A	I_{sp}, F Limiting upper value for t_1	$F/W_A = n_o$ $\mu = \exp.\ (\tau_x/I_{sp})$ $\Lambda = (\mu - 1)/\mu$ $\frac{I_{sp}}{n_o}\Lambda = t_1$ = operating time $t_1 >$ limiting value	Negative (Engine rejected by computer)
	It should be $2000 \leq I_{sp} \leq 3000$ sec and $\frac{1}{k_e} > 2$	$\frac{1}{k_e}$ vs I_{sp} for given engine type	$\frac{1}{k_e} > 2$ for $I_{sp} \leq 2400$	Positive for $I_{sp} \leq 2400$ Negative for $I_{sp} \geq 2400$ (Engine rejected by computer)
	Mission has n maneuvers: $\sum_{i=1}^{n} x_i$ Mass fraction of stages to be $x_1, x_2 \ldots x_n$	Limiting lower value for overall payload weight fraction $\prod_{i=1}^{n} \lambda_i$	For each stage: $\Lambda_i = (\mu_i - 1)/\mu_i$ $\lambda_i = 1 - (\frac{\Lambda}{x})_i$ $\prod_{n=1}^{i} \lambda_i <$ limiting value	Negative (Engine rejected by computer)
	Mission to be flown between 1978 and 1982 with 80% probability	Development Schedule	Schedule evaluation indicates: engine operational 1978 (75% probability) 1982 (95% probability) 1980 (80% probability)	Negative for 1978 Positive for 1982 Earliest date 1980 (Engine accepted by computer for 1980-82 period)

Tab. 4-9 EXAMPLES FOR EVALUATING ACCEPTABILITY OF GIVEN PROPULSION SYSTEM

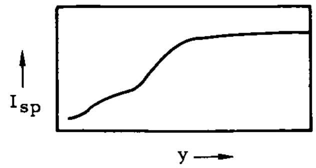
I Engine Designation

(I) Primary Mission Applications

(II) Specific Impulse: Sea level: Vacuum:
(Initial Values)

Altitude Compensation efficiency:

I_{sp} vs. Altitude:



III Thrust: Sea level: Vacuum:
Thrust Throttling Range:

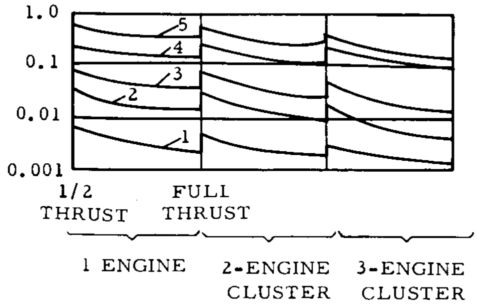
IV Scaling Coefficients:

1. Thrust Structure Insulation
2. Thrust Structure
3. Shielding against propellant heating

1 + 2 + 3 = overall thrust structure weight proportionality, k_{ts}

4 = k_e

5 = 1 + 2 + 3 + 4 = k_f



V Propellant or Expellant:

Propellant flow rate:

Propellant temperature:

Engine Power Output:

VI Operating Life:

VII Restart Capability:

VIII Clustering:

IX Availability: (90% Probability) Single: (Thrust) Cluster of: (Thrust)

Availability:	-	-	-
Probability:	-	-	-
Thrust:	-	-	-

X R&D Cost ($)
Year

XI

Production Cost ($):	-	-	-	-
Year:	-	-	-	-
Quantity:	-	-	-	-

XII Growth Potential:

Year	-	-	-	-	-
I_{sp}	-	-	-	-	-
Thrust	-	-	-	-	-
$1/k_e$	-	-	-	-	-
R&D Cost	-	-	-	-	-
Product. Cost	-	-	-	-	-
Reliability	-	-	-	-	-

Tab. 4-10 TYPICAL ENGINE DATA FILE

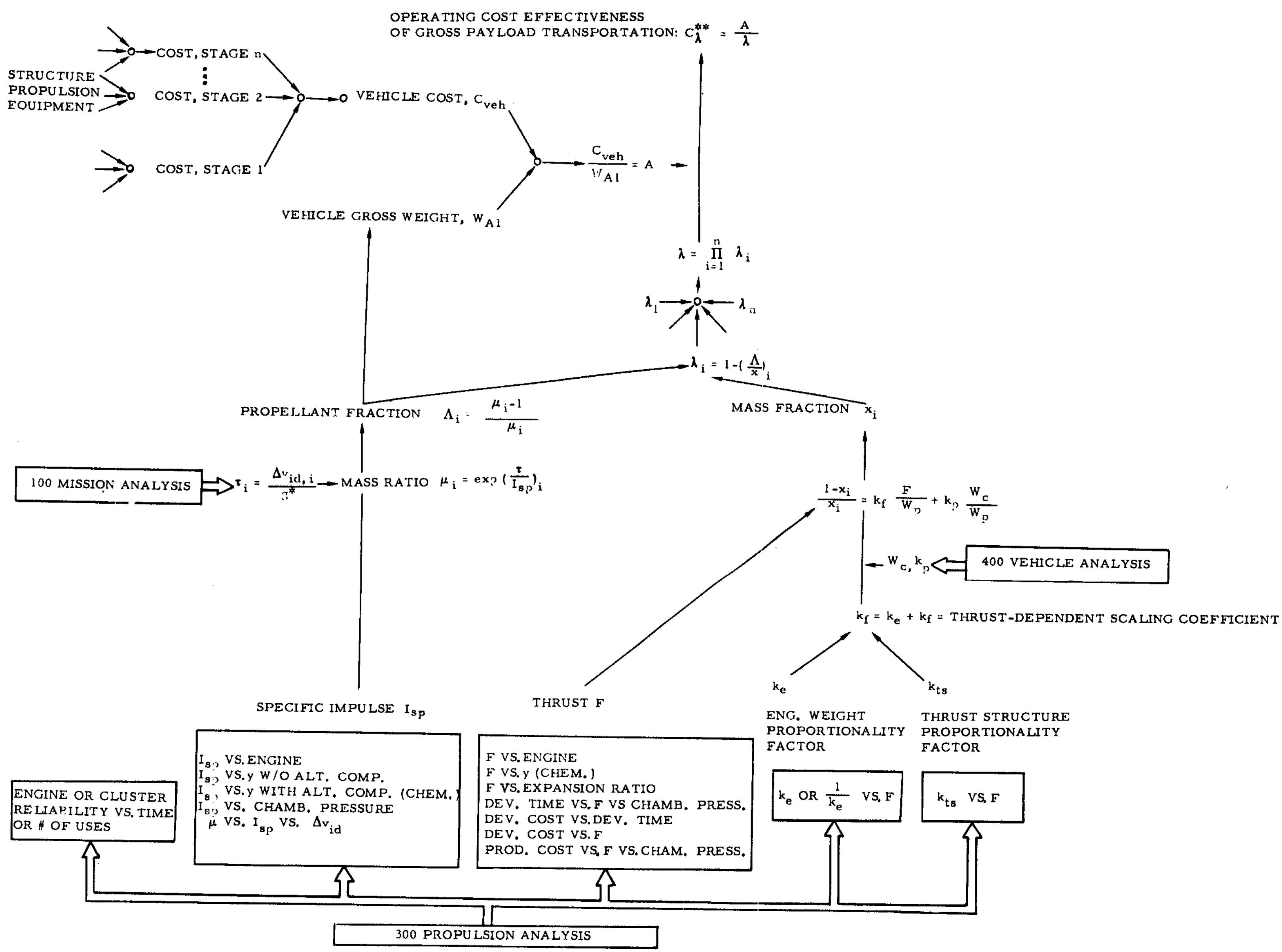


Fig. 4-11 INTERFACES BETWEEN PROPULSION ANALYSIS, COST ANALYSIS AND SUB-MODELS 100 & 400

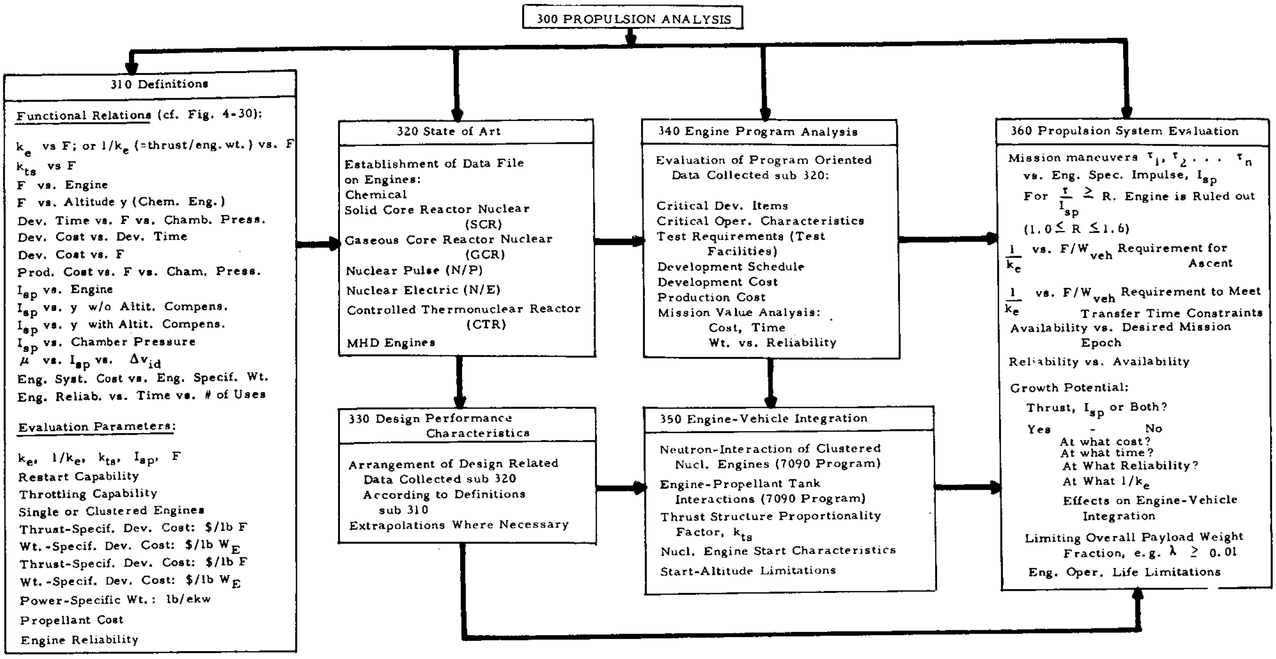


Fig. 4-12 PROPULSION ANALYSIS

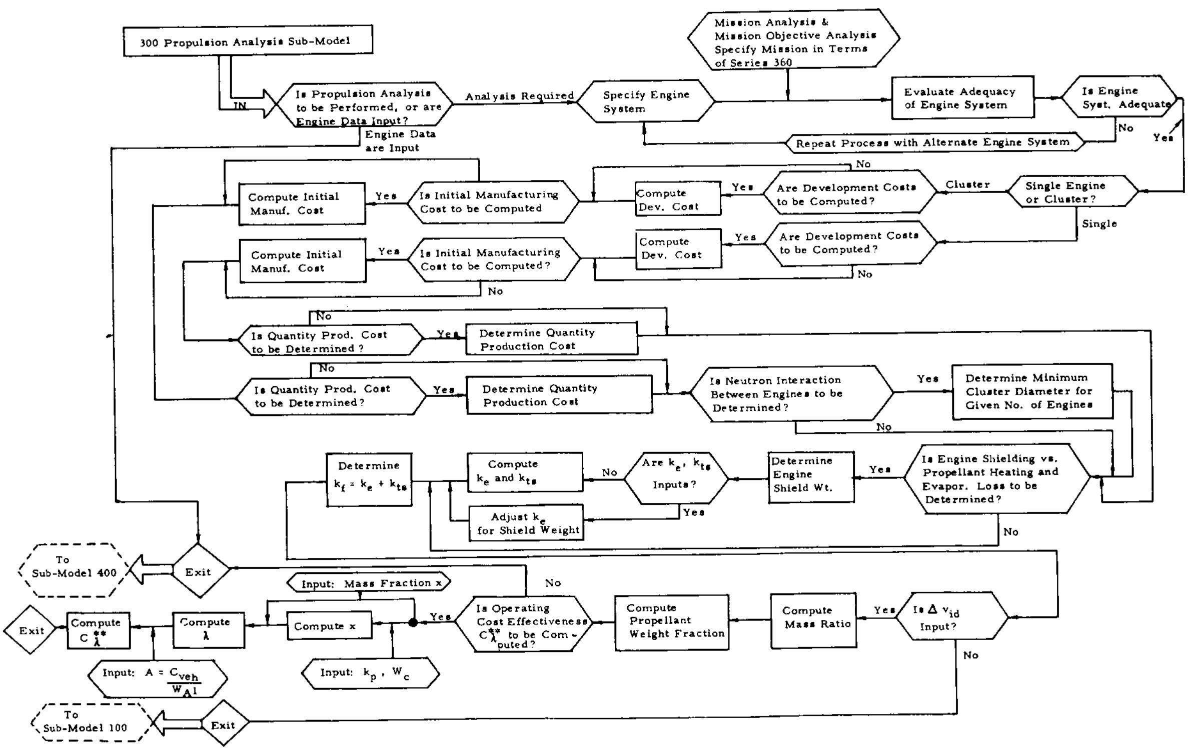


Fig. 4-13 SUB-MODEL 300, PROPULSION ANALYSIS

4.5 Vehicle Analysis

A breakdown of series 400, Vehicle Analysis, is presented in Tab. 4-11. A vehicle classification by principal functions, generally carrying significant differences in vehicle design, is shown in Fig. 4-14. The designation ELV merely indicates that the vehicle is Earth launched, not necessarily that its destination is restricted to Earth orbits. It may, in a process which will be referred to as direct delivery, deliver payloads into circumlunar or circumplanetary orbits, if its propulsion system is sufficiently powerful. But the term ELV does imply not only surface start, but also sub-orbital start for a second stage. Since this stage would continue the flight to the lunar or planetary destination, it represents a space ship whose engines and structural frame are designed for relatively high thrust/weight levels (around one). The interorbital space vehicles (ISV) may have any range of thrust/weight ratio from the ELV upper stage on down to the very low values characterizing the nuclear-electric category. Generally, even those ISV's which are powered by engines capable of high thrust, operate at the lowest thrust/weight ratios commensurate with the specific impulse of their engines (or other constraints, such as limited operating time). Their engines and structural frame are lighter; hence they have superior mass fractions and often possess a higher specific impulse, realizable because of lower engine thrust requirement. By definition, the ISV is not equipped to perform surface descent. If this type is reusable, it is a shuttle vehicle. The third basic group comprises destination space vehicles, i.e. vehicles designed to transfer payload from a planetocentric or selenocentric orbit (hyperbola or closed orbit) to the surface. The fourth group contains launch vehicle from the surface of the target body. The DSLV and the DSV could be one and the same, or the DSLV could be carried as payload by the DSV. In either case the design requirement for DSLV is more or less different from that of the DSV. Pick-up vehicles and reserve vehicles are classified in a separate group, because the requirements for their missions may cause them to be distinct from either of the preceding groups.

The procedure in the vehicle analysis is analogous to the one described for the propulsion analysis. Differences in the design aspects are often caused by differences in the propulsion systems used. This is particularly true for ISV's. In other cases, aerodynamic or heating considerations may play an important role also. In either case, however, the fundamental vehicle anatomy is necessarily the same in vehicles whose operation is independent of their environment. Each vehicle must possess a payload section, a storage section for its working fluid and an energy conversion system (propulsion system) in which the transport system for the working fluid from storage to the ejection device (thrust unit) is included. The sum of the weights of propellant, its storage section and the energy conversion section represent the net weight W_a of the

400 Vehicle Analysis

For vehicle classification by principal functions see Tab. 4-12

410 Definitions

420 Design Characteristics

430 Parametric Weight Determination

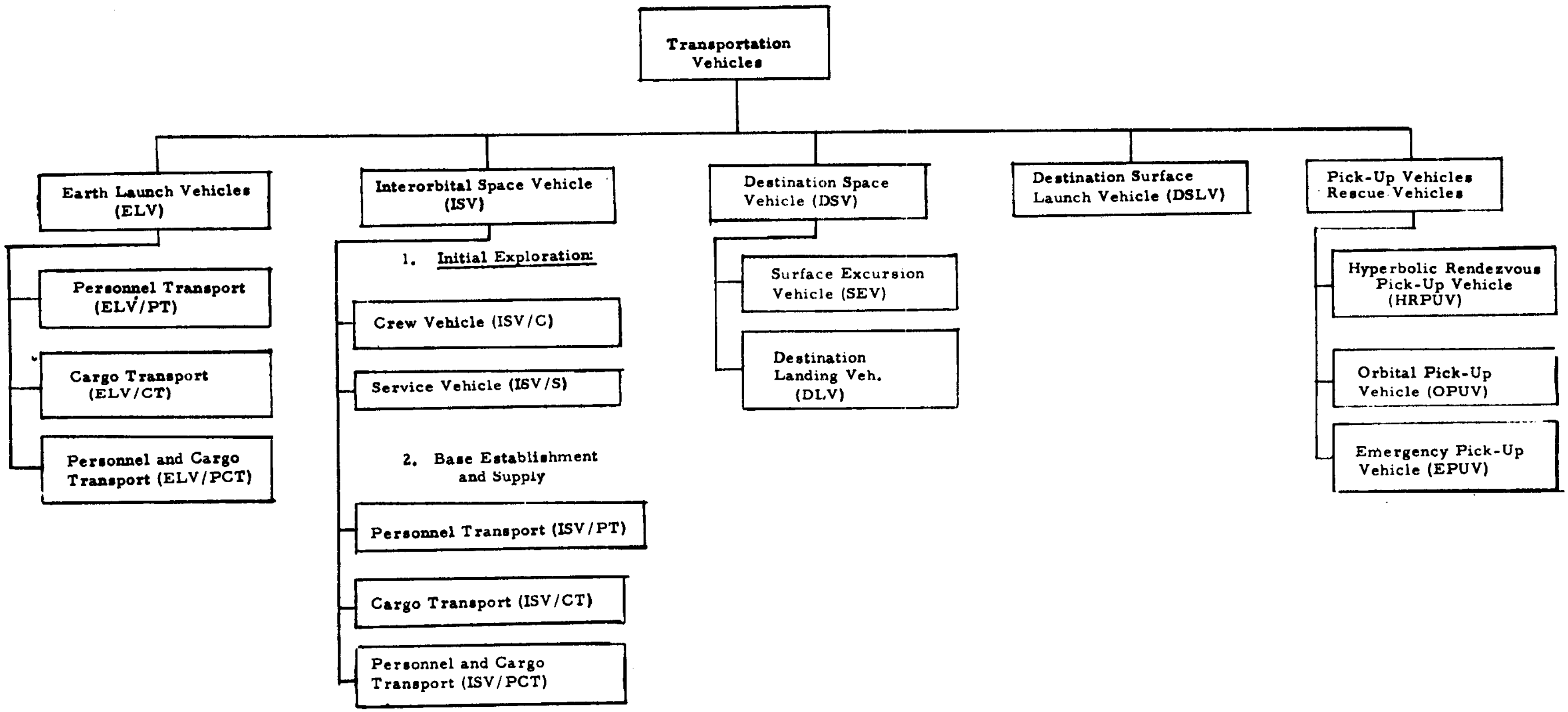
440 Program Analysis
- 441 Development
  - 4411 Critical Items
  - 4412 Facility & GSE Requirements
  - 4413 Development Schedule
- 442 Cost
  - 4421 Development Cost
  - 4422 Production Cost
  - 4423 Refurbishing Cost

450 Operations Analysis
- 451 Ground Operations (ELV)
- 452 Orbit Delivery Operations (ELV, I/V)
- 453 Mission Operations
- 454 Reliability Analysis
- 455 Orbit Delivery Success Analysis (ELV)
- 456 Mission Success Analysis

460 Vehicle Evaluation
- 461 Mission Worth
- 462 Service Reliability
- 463 Operational Availability
- 464 Operational Characteristics
- 465 Development Characteristics

Tab. 4-11 BREAKDOWN OF VEHICLE ANALYSIS

Fig. 4-14 VEHICLE CLASSIFICATION BY PRINCIPAL FUNCTIONS



vehicle or stage. Net weight and gross payload weight W_λ yield the gross weight, (or ignition weight) W_A, of the vehicle or stage (the gross payload can be the next higher stage or stages. The ratio gross payload of the final stage, $W_{\lambda n}$, referred to, simply, as λ, is, therefore, $\lambda = \prod_{i=1}^{n} \lambda i$. Inter-stage weights are counted as part of the next lower stage to the extent to which they are jettisoned with the lower stage.

In lunar and planetary missions which are long, compared to orbital ascent, long coast periods separate the periods of powered maneuver, where reasonably high thrust/weight ratios are used. During these coast periods, weight is eliminated for various reasons. The weight eliminated or jettisoned, for instance, between the (n-1)-th and n-th maneuver (only principal maneuvers are involved here, not small corrections) is referred to as $W_{j(n-1)(n)}$ (e.g. W_{j34}). Finally, prior to each principal maneuver, all weights no longer needed are jettisoned, such as, for instance the thermal and meteoroid shield system of the tanks about to be emptied. The weight jettisoned prior to the n-th maneuver is referred to as W_{jn}.

The four major weight constituents of every vehicle, namely, W_P, W_p, W_T and W_λ are of different relative dominance for different vehicles, mainly due to the influence of propulsion systems. Chemical and the lower energy nuclear vehicles are dominated by W_p, i.e. chemical propellants or hydrogen. These nuclear vehicles are therefore referred to as <u>hydrogen vehicles</u>. In the nuclear pulse vehicle the propulsion system weight is comparatively much higher, although propellant weight is still dominant. In vehicles driven by power-separated propulsion systems, such as the nuclear-electric vehicles, the propulsion system weight is particularly large due to the weight of its energy conversion system and its radiators. In these vehicle, as well as in CTR-driven vehicles (where the propulsion system is likewise a dominant weight item) payload is often the second-largest weight item except for very-high-energy missions. Fig. 4-15 illustrates these basic trends. The solid core reactor engine and gaseous core reactor engine powered vehicles belong to the class of hydrogen vehicles. The N/P vehicle is dense, using metallic propulsion material. The N/E vehicle, also using dense metallic expellants is dominated by its radiators, as perhaps the most critical component of this vehicle.

Definitions characterizing these vehicle groups are by necessity different. They are used in the vehicle card file to provide significant vehicle inputs for the computer model. Tab. 4-12 shows the data sheet for large ELV's as an example.

Design characteristics (420 in Tab. 4-11) are extracted from data collected or from point designs made. They form the basis for expansion into parametric weight determination systems (430).

Tab. 4-12 EARTH LAUNCH VEHICLE INPUTS

(Non - Manned)

ELV: Designation

Payload Section:	Dia. d	Useful Vol.	Pld. V_ω	Height Cyl. Sect. h_{cyl}	Height Cone Sect. h_{cone}	Vol. Cyl. Sect. V_{cyl}	Vol. Cone Sect. V_{cone}
	--	--	--	--	--	--	--

Payload Weight:

Delivery Altitude (km)	180	270	360	450	540
Net Pld., W_ω	--	--	--	--	--
Max. Pld. Density W_ω / V_ω	--	--	--	--	--
Net Pld., W_ω(Tanker)	--	--	--	--	--

Development Cost:	Year	1973	74	75	76	77	78	79
	\$	-	-	-	-	-	-	-

or Function $\frac{d\theta}{d\xi} = -\frac{1}{w}\frac{d}{d\xi}\frac{2}{\tau^{w-1}(1+\tau^2)}$

Initial Operational Availability:	Earliest Exp. Date T_E	Most Likely Date T_M	Latest Allowable Date T_L
	--	--	--

Mission Reliability:	Year	1979	80	81	 90
	P_D	-	-	-	

or

$$R = R_{max} - \frac{R_{max} - R_{1st\ Launch}}{\left[\sum_{\Delta t=0}^{n} N_{Act.\ Launch}\right]^{\beta}}$$

for indiv. launch of given ELV

or

$$R_t = R_{max} - \frac{1}{1-\beta}\,\frac{\left[\sum_{t=0}^{n} N_{Act.\ Launch}\right]^{1-\beta} - \left[\sum_{t=0}^{n-1} N_{Act.\ Launch}+1\right]^{1-\beta}}{\sum_{t=0}^{n} N_{Act.\ Launch} - \sum_{t=0}^{n} N_{Act.\ Launch} - 1}\left(R_{max} - R_{1st\ Launch}\right)$$

Reusable:	St. 1	Yes / No	St. 2	Yes / No

Turn-Around Time:

Direct Operating Cost Effectiveness per Launch: $C^{xx}_{O,D}$ (\$/lb Pld) $= \frac{K_{prod}}{0.9}\,\frac{W_d}{W}$; $K_{prod} = a^{-b} e$ $b = 1/6$(90% learning)

$z = 0.03\,(0.9Y - 0.3) + 0.00022\,\frac{v}{Y2}$

$0.00000008\,\frac{v^2}{Y6}$

y = Cumulative No. of Years from In. Oper. Av.

Indirect Operating Cost Effectiveness per Launch: $C^{xx}_{I,D}$ (\$/lb Pld) $= m\, C^{xx}_{O,D}$

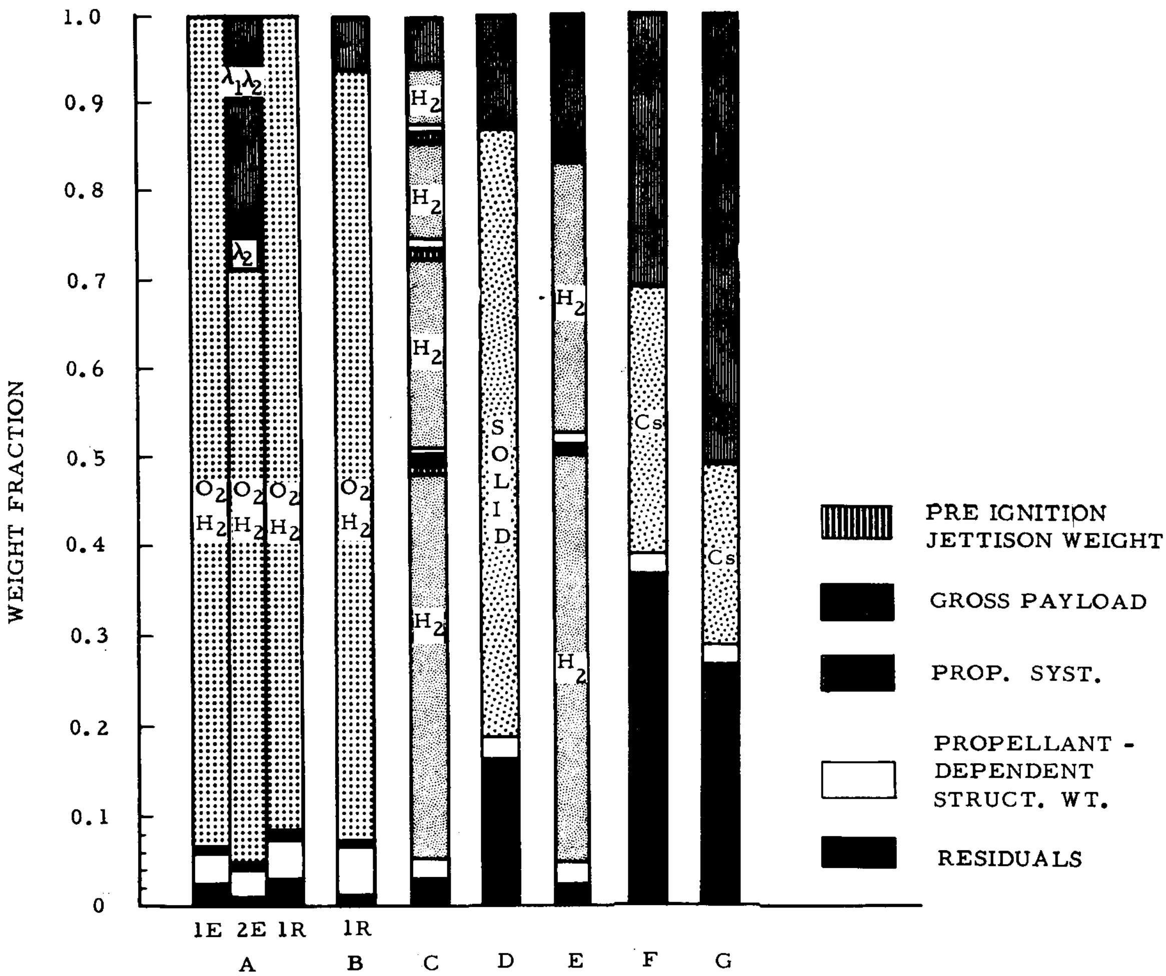
Total Operating Cost Effectiveness per Launch: C^{xx} (\$/lb Pld) $= \frac{K_{prod}}{0.9}\,\frac{W_d}{W}\,(1+m)$

Total Operating Cost per Launch: C' (\$/Launch) $= \frac{K_{prod}}{0.9}\,W_d\,(1+m)$

Total Cost of given Orbit Lift Operation: $C_{op} = N_{op}\,C'$; $N_{op} = N_{Act.\ Launches}$ during operation

OR

Annual Launch Cost: $C = N_Y\,C'$; $N_Y = N_{Act.\ Launches}$ during Year



A = ELV; Δv_{id} = 30,000 FT/SEC; 2 STAGES
1E STAGE 1 EXPENDABLE
2E STAGE 2 EXPENDABLE
1R STAGE 1 RECOVERABLE
B = ELV; Δv_{id} = 30,000 FT/SEC; 1 STAGE
C = 4-STAGE MARS VEHICLE; ST. 1: I_{sp} = 825 SEC; ST. 2, 3, 4: I_{sp} = 900 SEC; $\Sigma \Delta v_{id}$ = 80,000 FT/SEC
D = N/P MARS VEHICLE; I_{sp} = 1900 SEC; $\Sigma \Delta v_{id}$ = 73,000 FT/SEC
E = GCR MARS VEHICLE; 2 STAGES; I_{sp} = 2000 SEC; $\Sigma \Delta v_{id}$ = 90,000 FT/SEC
F = N/E ION VENUS VEHICLE; EARTH ORBIT LAUNCHED; I_{sp} = 14,000 SEC; $\Sigma \Delta v_{id}$ = 165,000 FT/SEC
G = N/E ION VENUS VEHICLE; INJECTED INTO PARABOLIC ORB.; I_{sp} = 13,000 SEC; $\Sigma \Delta v_{id}$ = 100,000 FT/SEC

Fig. 4-15 WEIGHT FRACTION OF PRINCIPAL WEIGHT GROUPS

The three major groups of weight calculations are shown in Fig. 4-16.

The second group of weight calculations shown in Fig. 4-16 often assumes a supporting role as part of payload weight calculations. In the third group it is found expedient to sdparate the computation of the Earth orbital departure weight from Earth return weight computations, because of the many options which exist for Earth return. Therefore, computation of Earth entry weight (if Earth entry takes place), Earth return retro-maneuver (if one takes place) and of weights jettisoned prior to Earth arrival (i. e. at termination of heliocentric return coast) and during Earth return coast are treated separately. Reason for separating Earth return coast is the possibility of considerable variation in number of persons occupying the vehicle on the way out and on return, if passenger delivery to a base is involved.

Series 440. Program Analysis, is treated corresponding to the subject items listed in Tab. 4-11. Comparing these listings with those in Fig. 4-1 it will be noted that only those items are shown in Tab. 4-11 which pertain to the individual vehicle and are not vehicle-mission-integration oriented.

Operations Analysis, series 450, is detailed in Tab. 4-11. In comparing the listings with those in Tab. 4-1 it is noted that only the subjects pertaining to the vehicle proper are listed in Tab. 4-11, whereas those requiring vehicle-mission integration, such as orbital operations, are omitted. Details are shown in Fig. 4-17. The data files contain reliability data in the form shown in 4-18. In the upper left corner is shown an example of presentation of individual reliabilities as function of time. To its right hand side typical delivery reliabilities are shown for the DFM-mode, i. e. the mode in which complete vehicles are delivered into orbit, ready to depart. For a realistic determination of the actual number of launch attempts, it is necessary to consider whether or not the vehicles are identical and interchangeable. In the lower part of Fig. 4-18 are shown typical matrices pertaining to the orbital vehicle-assembly mode (OVAM). Here also, the difference between interchangeability and lack thereof results in considerable differences in procurement, hence, in cost. These differences are especially important when dealing with individual missions rather than missions which recur on a routine basis. The data file, as shown in Fig. 4-18 makes no assumptions in this respect, being concerned with the individual vehicle performance only.

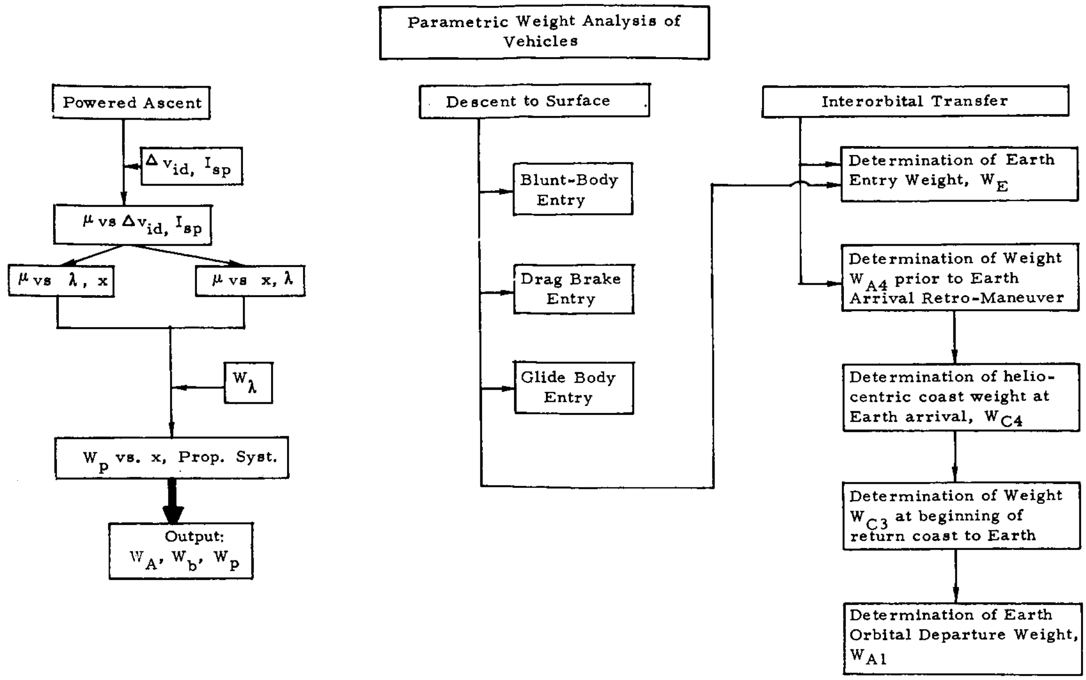


Fig. 4-16 THREE MAJOR GROUPS OF WEIGHT CALCULATIONS

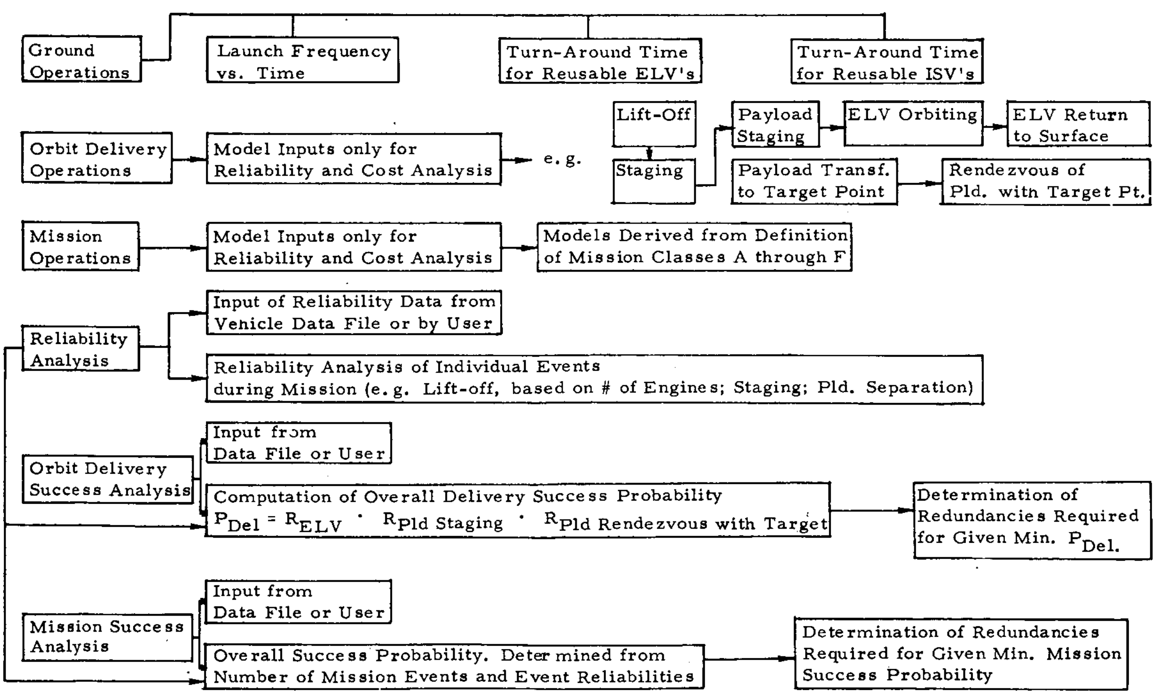
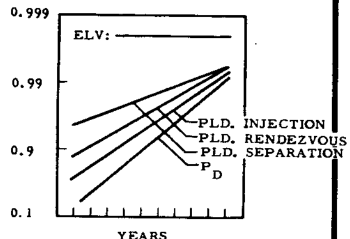
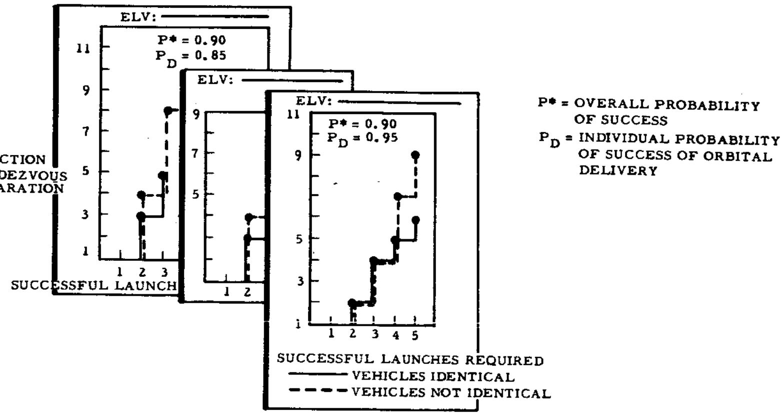


Fig. 4-17 VEHICLE OPERATIONS ANALYSIS (SERIES 450): DETAILS

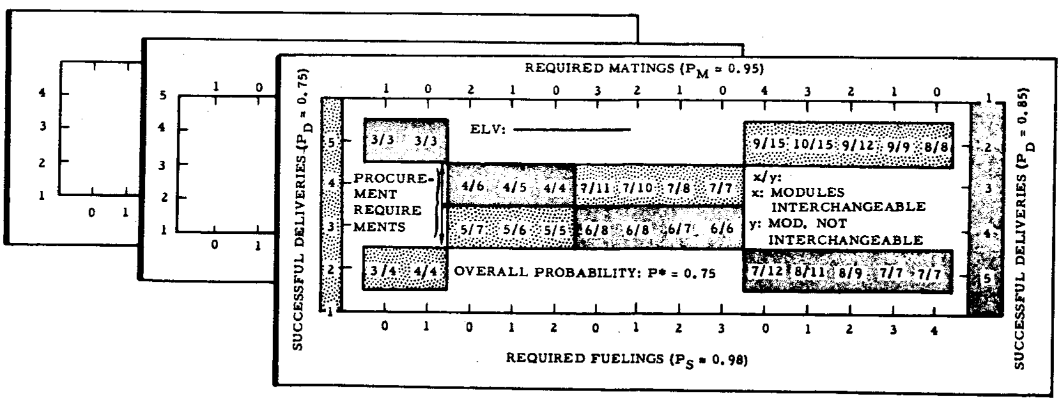


EARTH ORBIT DELIVERY(EOD):

INDIVIDUAL RELIABILITIES AND PROBABILITY OF SUCCESS OF DELIVERY



DIRECT FLIGHT MODE (DFM): NUMBER OF LAUNCH ATTEMPTS VERSUS NUMBER OF REQUIRED SUCCESSFUL LAUNCHES FOR SPECIFIC DELIVERY SUCCESS PROBABILITIES AND OVERALL SUCCESS PROBABILITIES FOR IDENTICAL AND NON-IDENTICAL VEHICLES



ORBITAL VEHICLE-ASSEMBLY PROCUREMENT MATRIX. FIGURES IN MATRIX SHOW ACTUAL PROCUREMENTS REQUIRED FOR SUCCESSFUL DELIVERIES OF 2, 3, 4, OR 5 ITEMS, DEPENDING ON P*, ON P_D (LEFT AND RIGHT ORDINATE), ON THE DISTRIBUTION OF MATING AND FUELING IN ASSEMBLING THE ISV (UPPER AND LOWER ABSCISSA), ON THE PROBABLITY OF SUCCESSFUL MATING OR FUELING AND ON WHETHER THE MODULES TO BE ASSEMBLED ARE INTERCHANGEABLE OR NOT (LEFT AND RIGHT NUMBER x/y)

Fig. 4-18 DATA FILE ON RELIABILITIES AND SUCCESS PROBABILITIES CHARACTERISTICS FOR GIVEN VEHICLE

4.6 Resources

The resources consist of funds, skilled manpower, facilities and existing technological capabilities where relevant.

The US Government expenditures have climbed from six million dollars per annum in 1800 to a requested 98,802 million dollars in FY 1964 while the population has grown from six million to 177 million in the same time.

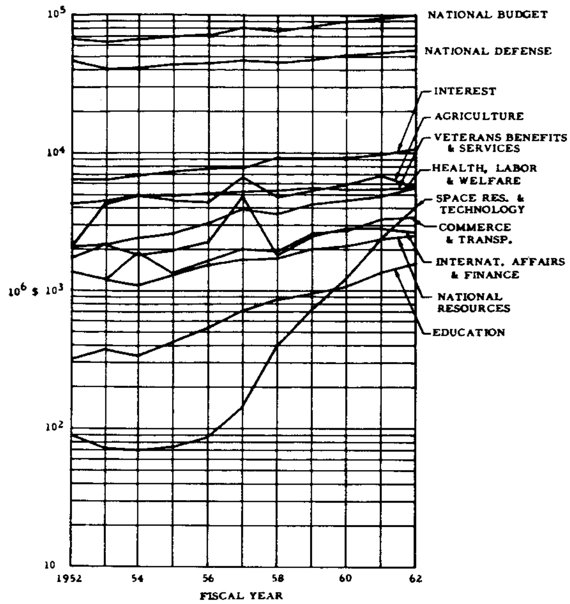
Fig. 4-19 shows the national budget breakdown in the period 1954-64. It is noteworthy that the majority of items lies in a band of one to ten billion dollars annually. Only the national defense layouts are significantly higher. Space Research and Technology and Education have more recently grown into this regime and are now likely to level off for the time being, especially the space budget. In the long run (through 1980), the only item which potentially might outgrow this band and reach proportions comparable to those of the national defense budget is the Health, Labor and Welfare budget, because it is most affected by the pressure resulting from population growth, automation and political considerations. The interest payments too, appear to climb beyond the ten billion dollar mark, but are not expected to grow at a rate comparable to the Health, Labor and Welfare budget. Based on the principle of equilibrium of objections (and interests) in peacetime, the other budget items are likely to remain at comparable levels, and eventually to rise as a group above the ten billion dollar mark.

The national budget as a whole approaches the 100 billion dollar level with national defense and interest payments remaining the largest single items. For the last ten years, however, the national budgets (federal payments) have stayed rather steadily around 20% of the gross national product (GNP). As a result of WW-II efforts, with federal expenditures exceeding 45% of the GNP, the public debt climbed to 125% of the GNP, but since then has decreased steadily (Fig. 4-20).

During the last ten years the federal layouts for R&D have increased by a factor of five. The principal increases were in the area of national defense and space programs. Of the total budget five to 6.5% have gone into research facilities proper; the rest was spent conducting R&D.

It is of interest to examine the more relevant budgets in some detail. The DoD budget is by far the largest and the most well established one. Fig. 4-21 shows that, in spite of considerable changes in the choice of weapons and philosophy of warfare during the 1954-64 period, the major cost items remained relatively constant, with the exception of R&D and Test Evaluation, reflecting

Fig. 4-19 BUDGET BREAKDOWN 1954-1964



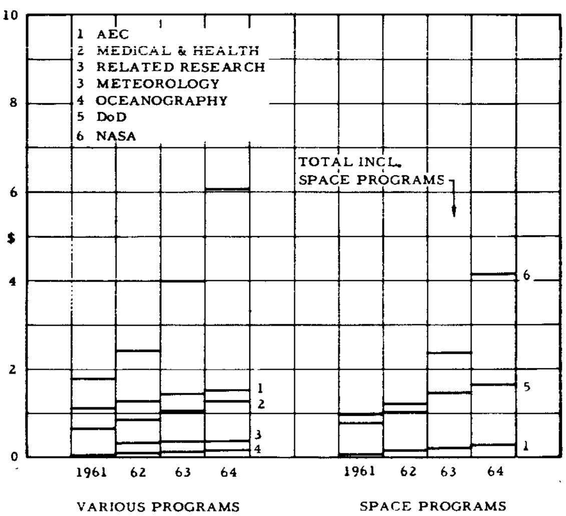


Fig. 4-20 FEDERAL R & D EXPENDITURES 1961-64 EXCLUSIVE NATIONAL DEFENSE R & D

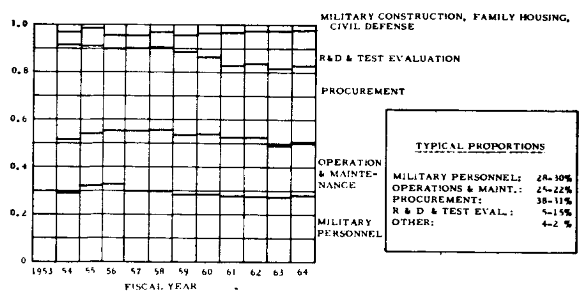


Fig. 4-21 BREAKDOWN OF BUDGET FOR DoD MILITARY FUNCTIONS

the growing scientific and technical sophistication of modern weapons. Nevertheless, the "big three" cost items are personnel, procurement and operation and maintenance, in that order. The national space program does not require a comparable number of personnel. The "big three" in that case are R&D and Test Evaluation, procurement and operation and maintenance. The space program is younger and far less well settled, since it has not retained the "flywheel" of "conventional" activities to the degree Dod had to maintain its conventional armed forces. The equivalent would be, for NASA, aircraft R&D which, however, had to be reduced drastically (Fig. 4-22) since 1959. The NASA program breakdown is therefore at this time governed by R&D activity, since very few "going systems" are in existence which would raise operations and maintenance cost and, to some degree, also procurement cost. The portion of the NASA budget designated as R&D budget in the Administration's budget reports to Congress is broken down in Fig. 4-23. Its main components are manned space flight, unmanned investigations of space, space research and technology and supporting operations. Their proportions are shown in Fig. 4-23. Additional items, not shown, are space applications and aircraft technology. The former had a budget of 16.2, 60.6, 82 and $123 \cdot 10^6$ \$, including R&D Facilities. While the manned program R&D activities vary between 13 and 20%, the budget for instrumented probes ranges from 3 to 5% so far. It is likely that this budget will have to be increased if the instrumented program is to provide timely support for a vigorous manned lunar and interplanetary program. In contrast to the Dod budget, the R&D Facilities percentage is higher, namely, 10 to 15%.

Fig. 4-24 compares the growth of the US population, the Government expenditures the GNP and the NASA budget. The solid lines are actual values, the dashed lines are extrapolations. Fig. 4-25 shows the rise in Government expenditures and the GNP per capita of the population. In 1959, the GNP amounted to 2700 \$/person. The increase between 1940 and 1960 is primarily due to automation. Since the potential of automation is far from exhausted, the per capita value of the GNP can be assumed to reach values between 10,000 and \$15,000 by 2000. As conservative values, 6000 and 3000 \$/person were added. Based on the three population points in 2000 in Fig. 4-24 and the four per capita values for GNP a total of 12 possible values for the GNP in 2000 is obtained. The spread, as shown ranges from 570 billion to 3750 billion dollars. Consequently, if the Government expenditures stay at 20% of the GNP, the resulting lower and upper extremes of the national budget in the year 2000 are 114 and 750 billion dollars.

Selecting two values, namely a GNP of 3000 \$/person and one of 6000 \$/person and selecting the lowest of the three population points in Fig. 4-24, two upper limits for the annual appropriations are established (5,700 and $11,400 \cdot 10^6$ p/a in 2000) and all programs must be fitted into this envelope (Fig. 5-26).

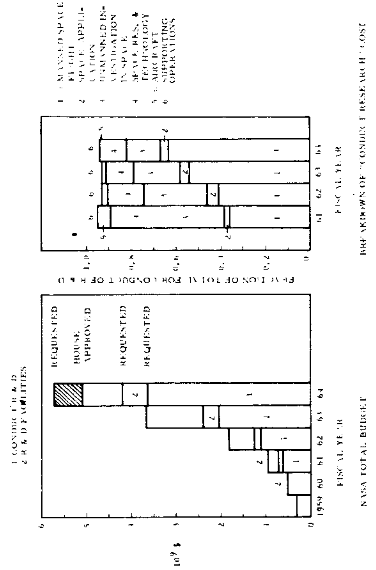


Fig. 4-22 NASA BUDGET

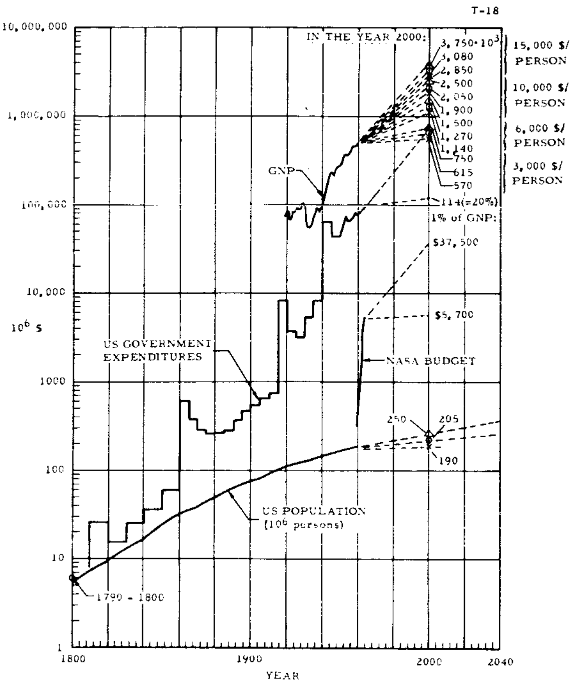
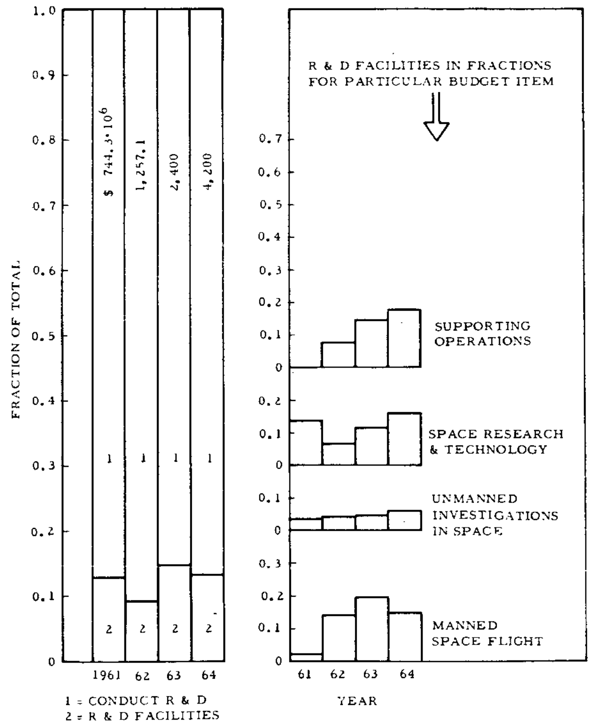


Fig. 4-24 COMPARISON OF US POPULATION, GOVERNMENT EXPENDITURES, GROSS NATIONAL PRODUCT AND THE NATIONAL SPACE BUDGET

Fig. 4-23 NASA R & D BUDGET



It is seen that the initial interplanetary mission could not be fitted into the lower of the two boundaries if all the other programs were carried out. The mission would either have to be postponed to the end of the eighties or another program must be curtailed. If, on the other hand, the upper of the two boundari[es] were available, the initial interplanetary mission could be carried out in the seventies. Fig. 4-27 shows the same model, but the funds are presented on a cumulative basis.

Thie model is crude and serves only to illustrate the method of establishing funding resources models for analysis and comparison.

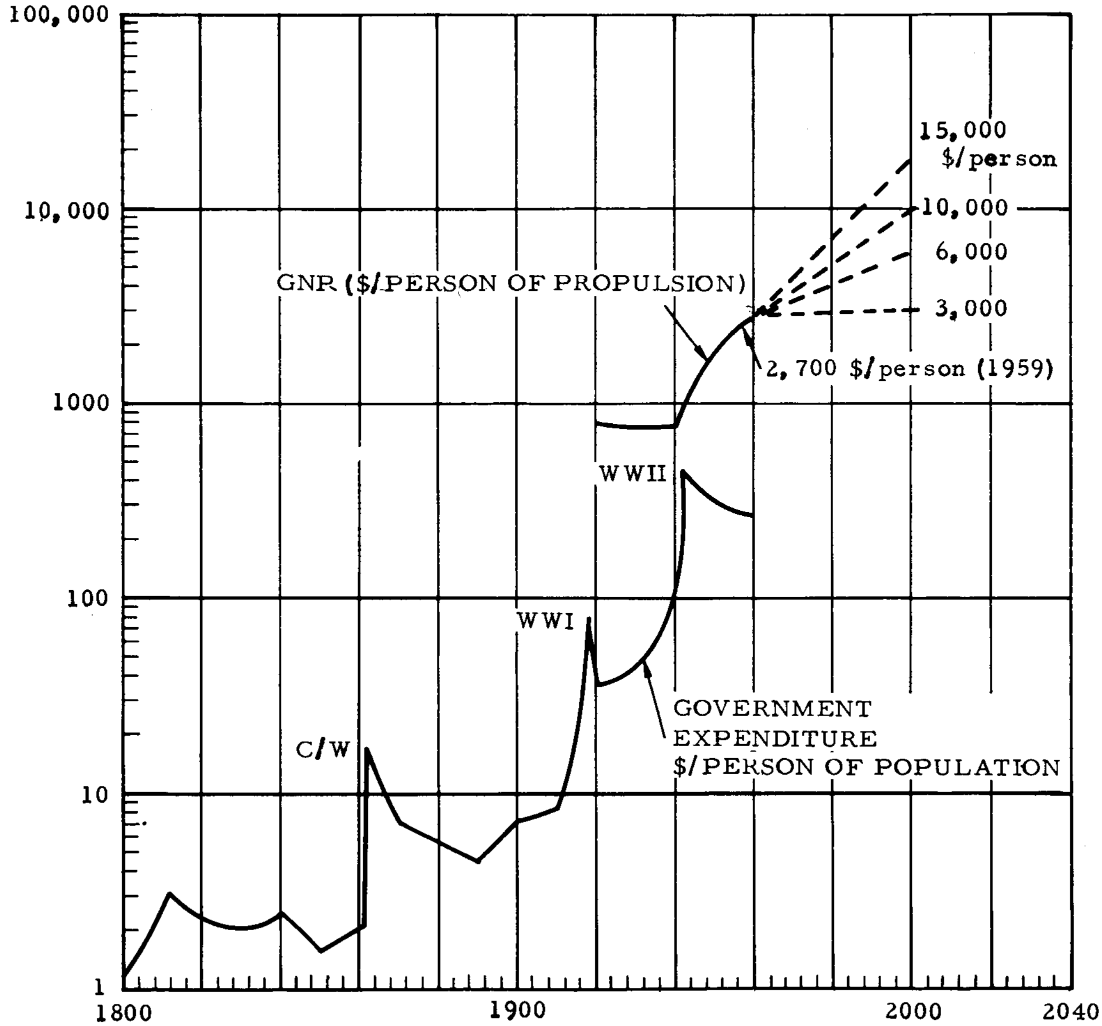


Fig. 4-25 GOVERNMENT EXPENDITURE AND GNP IN DOLLARS PER PERSON OF US POPULATION

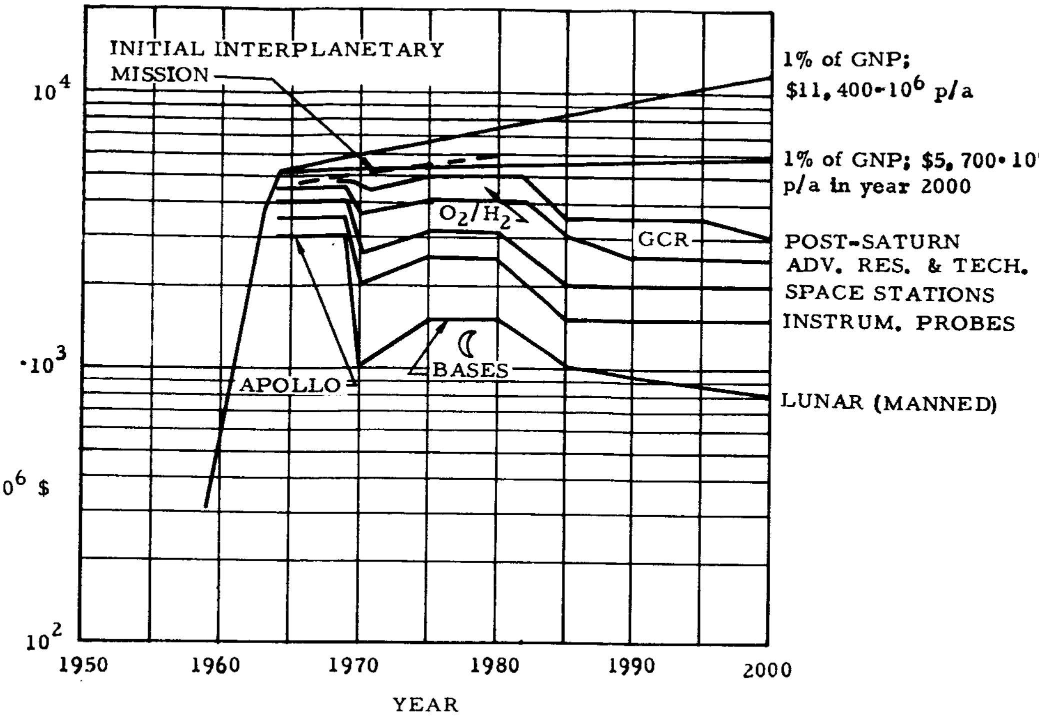


Fig. 4-26 MODEL OF YEARLY EXPENDITURES FOR VARIOUS SPACE PROGRAMS THROUGH 2000

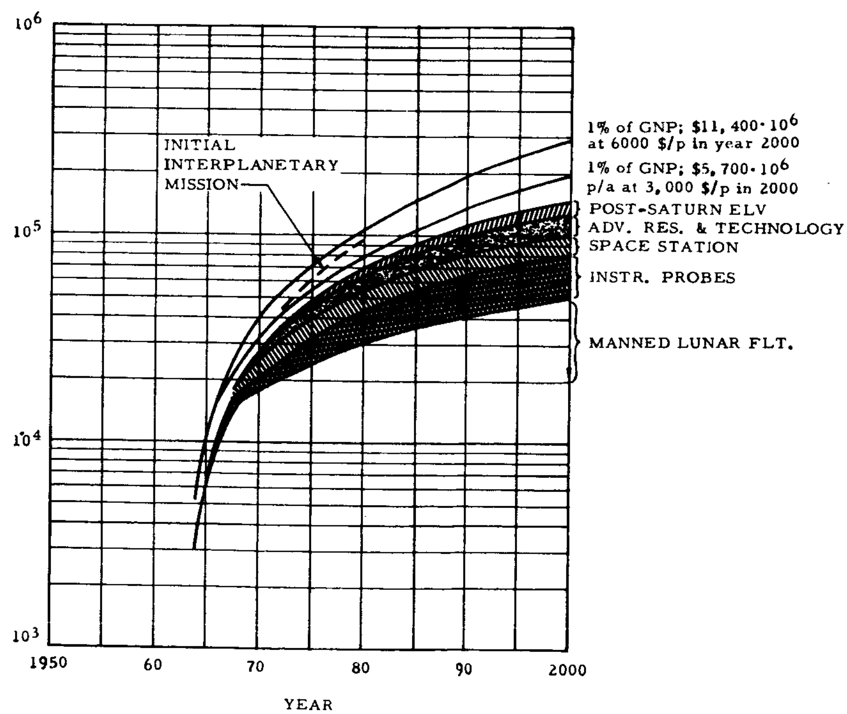


Fig. 4-27 EXAMPLE OF A FUNDING RESOURCES MODEL

5. Vehicle-Mission Integration (VMI) (600)

Objective of the VMI is to provide one or several models for the analysis of individual missions. The model has two major applications: (1) to provide decision data when the problem is to compare individual missions; e.g. comparison of a fly-by with a capture mission; or comparing different propulsion systems, launch vehicles, crew sizes or other important variables for a given mission; and (2) to generate data for characteristic missions of widely different mission groups or propulsion systems, in order to provide reference values in comparing alternatives in a program consisting of a large number of missions; i.e. to provide inputs into a technological program synthesis.

A vehicle-mission program model with emphasis on payload analysis is shown in Fig. 5-1.

Computation of the orbital burden rate is explained on an example given in Tab. 5-1.

The direct operating cost of a given mission consists, therefore, of the following key elements: (1) manufacturing cost of ISV; (2) launch cost of ELV; (3) planned number of launches, hence number of ELV's and payload (ISV secti and tankers) which must be procured, to assure a given overall probability of success of attaining orbital departure readiness; (4) orbital burden rate (OBR) In the long run, the high cost of the OBR will encourage investment of an ELV which is capable of launching ISV's into various missions via DFM rather than OVAM, although the initial investment in the form of development cost is high.

Initially, planetary missions are preceded by a considerable developm effort which not only adds to the overall mission cost, but also introduces a mission schedule uncertainty which is caused by the development schedule uncertainty of key systems and operational developments.

A refined vehicle-mission integration, therefore consists of the steps outlined in Fig. 5-1. These steps are subsequently discussed very briefly.

Step 1: By defining the mission class (Tab. A-2) the number and distributior of maneuvers is given. By defining the mission modes (Tab. A-8), the manner in which certain maneuvers are carried out is defined, from which basic development requirements can be derived. Missic energy requirement, mission period and mission objectives provide an additional basis for determining development requirements. In general, the planning of an interplanetary mission must consider the areas shown in Tab. 5-2 and its evaluation must follow an approach of the kind presented in Fig. 5-3.

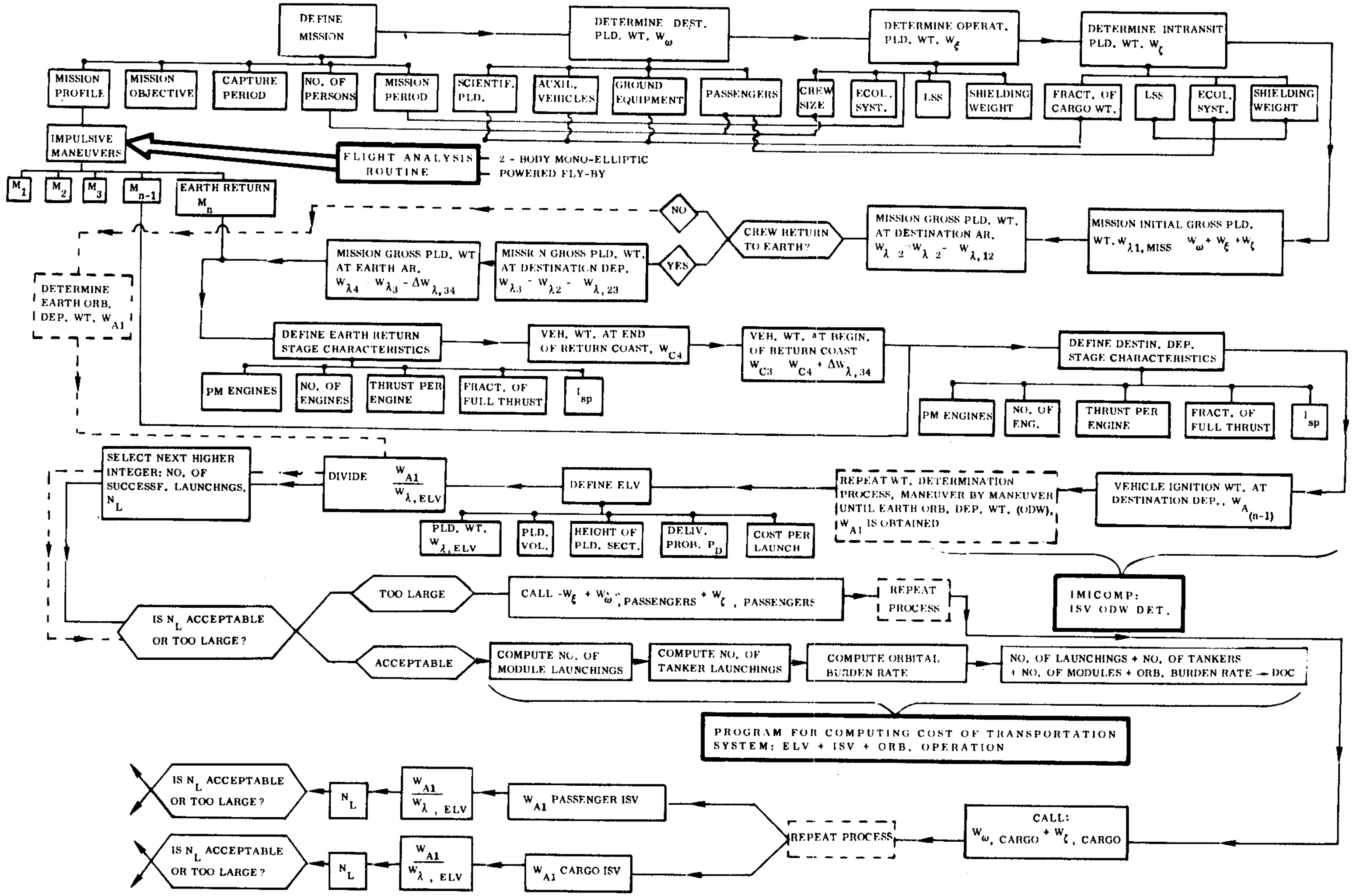


Fig. 5-1 VEHICLE-MISSION INTEGRATION MODEL

Tab. 5-1 COST ANALYSIS OF ORBITAL BURDEN (EXAMPLE)

For OVAM Cases, the Cost of Orbital Operation is Computed as follows:

Period of Operation: T_{op} (d) = 5 (n_L, act) + 10

Number of Personnel: N_p = 10 per interplanetary vehicle to be readied in orbit

Living Cost: C_L (\$/person) = T_{op} (d) · 170 (\$/lb) · 4.75 lb/d/person

Personnel Transportation into Orbit and back: $C_{Tr, P}$ = \$20,000/person

Special Training for Orbital Work: C_{Trng} (\$) = N_P · 800,000 \$

<u>Thus, cost of Orbital Labor:</u>

$$C_{OL} (\$) = N_P (C_L + C_{Tr, P} + C_{Trng})$$

Number of Orbital Labor Hours: h_L (hrs) = $\frac{1}{3} N_P \; T_{op} \cdot 24$

Hourly Cost of Orbital Labor: $C_{h, OL}$ (\$/hr) = $\frac{C_{OL}}{h_L}$

Material and Equipment (Expendable & Non-Expendable):

100 lb/person/day + 670 lb/person

Weight of Material and Equipment: $W_{M\&E}$ (lb) = N_P (100 T_{op} + 670)

Specific average Cost of M & E: 50 \$/lb

Earth-to-orbit transportation of M & E: 170 \$/lb

Overall M & E Cost: $C_{M\&E}$ (\$) = $W_{M\&E}$ (50 + 170)

<u>Overall Orbital Operations Cost</u>

$$C_{orb} (\$) = C_{OL} + C_{M\&E}$$

Tab. 5-2 **MANNED PLANETARY MISSION**

Mission Specification

1.1.1 Target planet and year: Venus, 1975

1.1.2 Mission group: Capture, elliptic orbit ($n = 8$; $r^* = 1.1$)

1.1.3 Mission objectives: MO_2

1.1.4 Mission data:

Dep Earth	5-20-75		
J. D.	244 2552.5		
$T_1(d)$	140		
v^*_1	0.1327		
v^*_2	0.1400		
Ar Venus	10-7-75		
J. D.	2692.5		
$T_{cpt}(d)$	20		
V_{DW}	-		
Dep Venus	10-27-75		
J. D.	2712.5		
$T_2(d)$	240		
v^*_3	0.2455		
v^*_4	0.2824		
	km/sec	ft/sec	
$1.03\,\Delta v_1$	4.24	12,900	
$1.03\,\Delta v_2$	1.61	4,900	
$1.03\,\Delta v_A$	1.40	4,300	
$1.03\,\Delta v_P$	3.24	10,200	
Δv_4			
Earth-Venus Δv_{corr}	0.091	300	
Venus capt. orb. Δv_{corr}	0.122	400	
Venus-Earth Δv_{corr}	0.091	300	
Plane change allowance Δv_{pl}	0.61	$\geq$ 2,000	
$1.03\,(\Delta v_1 + \Delta v_2 \cdot \Delta v_A + \Delta v_P)$	11.89	36,000	

2 **Destination Payload**

	Availability
Radar mapping equipment	1967
Environmental satellites	As needed
Atmospheric entry probes	as needed
Atmospheric buoyant probes	as needed
Landing probes	as needed

3 **I/V Propulsion and Design**

1.3.1 Propulsion System: Compatible with mission year

1.3.2 Vehicle Assembly Mode: Compatible with prop. syst. & ELV

1.3.3 Structural Design: Follows from prop. syst. & vehicle assembly mode

1.4 **Earth Entry Conditions**

1.4.1 Earth return mode: Compatible with mission year

1.4.2 Earth entry velocity: $v_E^* = 0.47$ EMOS (without retro-braking)

1.4.3 Whether or not requirement for Earth return retro-braking (M-4) exists or not, depends on mission profile and selected Earth entry condition

1.5 **Earth Launch Vehicle (ELV)**

1.5.1 I/V orbital departure weight follows from 1.1 through 1.4

1.5.2 Technological adequacy of existing ELV is evaluated against orb. dep. wt. and I/V assembly mode selected sub 1.3. The minimum number of launchings and the distribution of matings and fuelings are determined

1.5.3 Economic and Operational acceptability of existing ELV is evaluated on the basis of: ELV orb. delivery success probability for the mission year. Actual number of launchings derived from minimum mission number, vehicle assy mode, orb. delivery success probability. Associated orbital burden rate.

1.5.4 If ELV is found adequate and acceptable on both counts, it is removed from the list of critical influence factors

1.5.5 If ELV is found inadequate and/or unacceptable it must be decided whether a modification of the existing ELV can meet requirements

1.5.6 Or, whether a new ELV must be developed first

1.5.7 Selection of alternative must be compatible with mission year

1.6 **Supporting Instrumented Probe Program**

1.6.1 Depending upon mission objectives, a larger or smaller amount of information about the environment at the destination is required. Pre-mission information requirements are specified.

1.6.2 According to these requirements one can specify the probes required in support of the manned mission preparation:
- Roving interplanetary probe
- Non-optical fly-by probe
- Non-optical capture probe
- Optical fly-by probe
- Optical capture probe
- Combined fly-by & landing probe
- Combined capture & landing probe

1.6.3 A pre-mission instrumented probe program is derived from the requirements sub 1.6.1 and 1.6.2. The program is defined by:
- Type or types of probes required (1.6.2)
- The Earth launch vehicle for the probe (PELV)

1.6.4 The selected program is technically compatible with the selected mission year if development and deployment of probes and PELV are compatible with the lead times for the I/V development; and if there is a reasonable probability of timely and successful execution of this program

1.6.5 The selected program is economically acceptable within the frame work of the overall mission budget, if its cost does not exceed a specified percentage of this overall budget.

1.7 **Manned Space Station Program**

1.7.1 An orbital training and test program is considered a necessary part of the mission development program. Details of the program must be specified

1.7.2 For this program an orbital laboratory is required which preferably should consist of the I/V's life support section (LSS)

1.7.3 If this is not feasible, an available orbital laboratory may be usable while final proofing of the LSS takes place later as a separate orbital flight test. The orbital training requirements are similar for early Venus and Mars missions (powered or non-powered fly-by; elliptic or circular capture). The orbital laboratory required is a function of crew size and equipment to be orbit tested on a long-term basis. The adequacy of existing orbital facilities must be evaluated in the light of the requirements.

1.7.4 If existing orbital facilities are inadequate, a new pre-mission orbital laboratory program must be derived.

1.7.5 The selected program must be compatible with the lead times required for the I/V development and with the ELV's available.

Step 2: In analyzing likely pace setters, the characteristics and the desired time period of the mission must be taken into consideration. Usually, alternatives exist. Dependent on these, the pace setters will be different. A space program planning board has been developed which aides in assessing these factors. In a simplified form it is shown in 5-4. The upper portion contains a breakdown by mission; the lower portion shows a breakdown by major development areas versus time. The black dots represent operational availabilities or capabilities or supporting flights to be counted upon in the particular year with a given probability. Four program alternatives are shown. Obviously, the pace-setters are those, whose availability or occurrence materializes too close to the desired mission date. These critical items have been circled. It is seen that for the Mars powered fly-by mission (Ma-1), the technological capability of entering the Earth atmosphere at 15 km/sec represents the pace setting problem. For the Ve-2 mission this problem is avoided, but at the expense of two pace setters in the engine field (development of a second generation thermal neutron and a fast neutron engine). The alternative still exists to avoid making the engines the "battlefield" and instead working the mission around the first-generation solid core reactor engine (SCR-1) and a 15 km/sec Earth entry capability. Thus, even if mission plan and mission period are given, the pace setting tasks are not necessarily defined conclusivel

Step 3: Taking the most critical tasks so determined, their schedule and schedule confidence must be analyzed. This is done by defining the major development tasks and the supporting tasks, such as facility and tooling development where and if necessary. The schedule confidence is determined by PERT-methods. To assure consistent treatment of development schedules on the subsystem and system level (including auxiliary vehicles) it is necessary to standardize the development program scheme, as indicated, for example in Fig. 5-5.

Step 4: The overall mission preparation schedule is analyzed in a manner analogous to that described in Step 3; only that now all developments contributing to the attainment of mission readiness at a given time must be taken into account. Typical results are shown in Fig. 5-6. In this example the probability that a given mission preparation schedule is successful is shown as function of time for the following missions: (1) Venus, elliptic capture, (2) Mars, powered fly-by, (3) Mars, circular capture, (4) Mars, surface excursion, (5-A) Mars, synodic base, based on gaseous core reactor (GCR) engine powered I/V, (5-B) Mars, synodic base, based on nuclear pulse (N/P) powered I/V, (6-A) and (6-B) Mars, long-term base, based on GCR and N/P powered I/V's, respectively.

ep 5: Now, that the mission schedule confidence level is established, a mission year can be chosen for a given success probability, or a mission time period, ranging, for example, from 75 to 90 percent success probability.

p 6: Based on the so determined mission time period, the development cost can be determined. Again, one can either engage in a detailed analysis of the development cost, or, as is often sufficient, assure a reasonable expected development cost and study the effect of deviations from this cost.

For the latter case, a new approach has been adopted. The spending rate presented in non-dimensional form by a suitable analytical function f (ζ), ere $0 \leq \zeta \leq 1$ is the fraction of the overall time available for the particular elopment and $0 \leq f(\zeta) \leq 1$ if the fraction of the total development fund, spent ime ζ . Perturbations can be introduced in the original function (f_1 (ζ)) to ulate overruns or other economic effects of changes or complications in the gram, altering the spending from f_1 (ζ) to f_2 (ζ), as shown in an example Fig. 5-8. This analytical approach simplifies particularily the integration of eral missions, or of entire mission programs such as lunar or planetary sion programs into the national space budget with reasonable accuracy.

 7: Finally, for the mission time or mission time period selected in Step 5, a somewhat more accurate determination of the direct operating cost can be made, based on time dependent systems reliability and operational probability of success data.

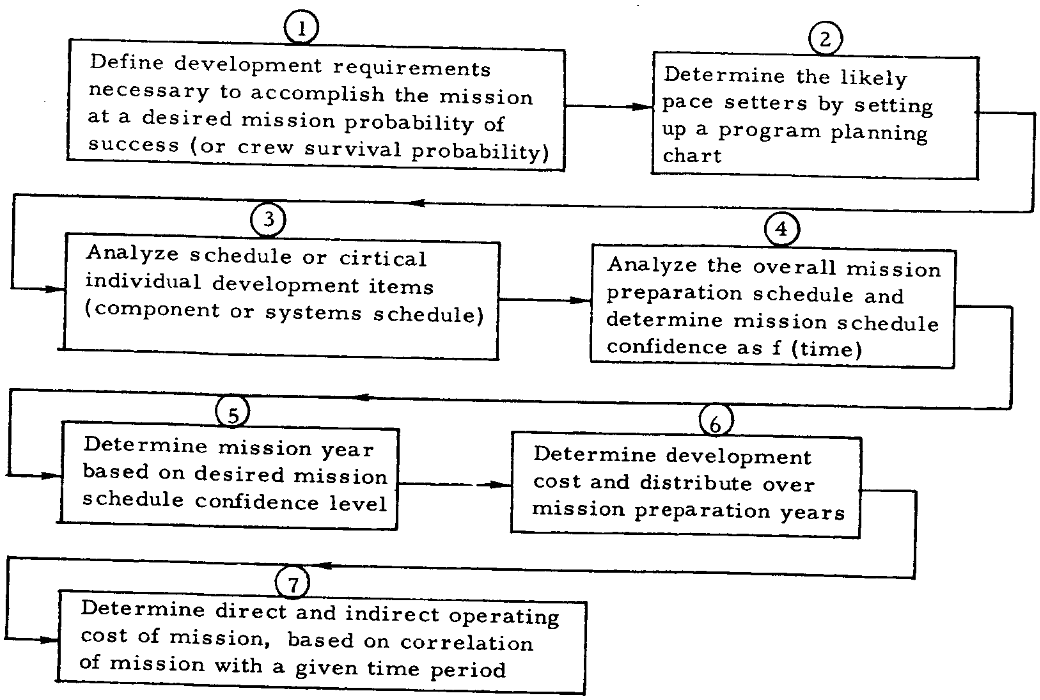


Fig. 5-2 STEPS IN REFINED VEHICLE-MISSION INTEGRATION

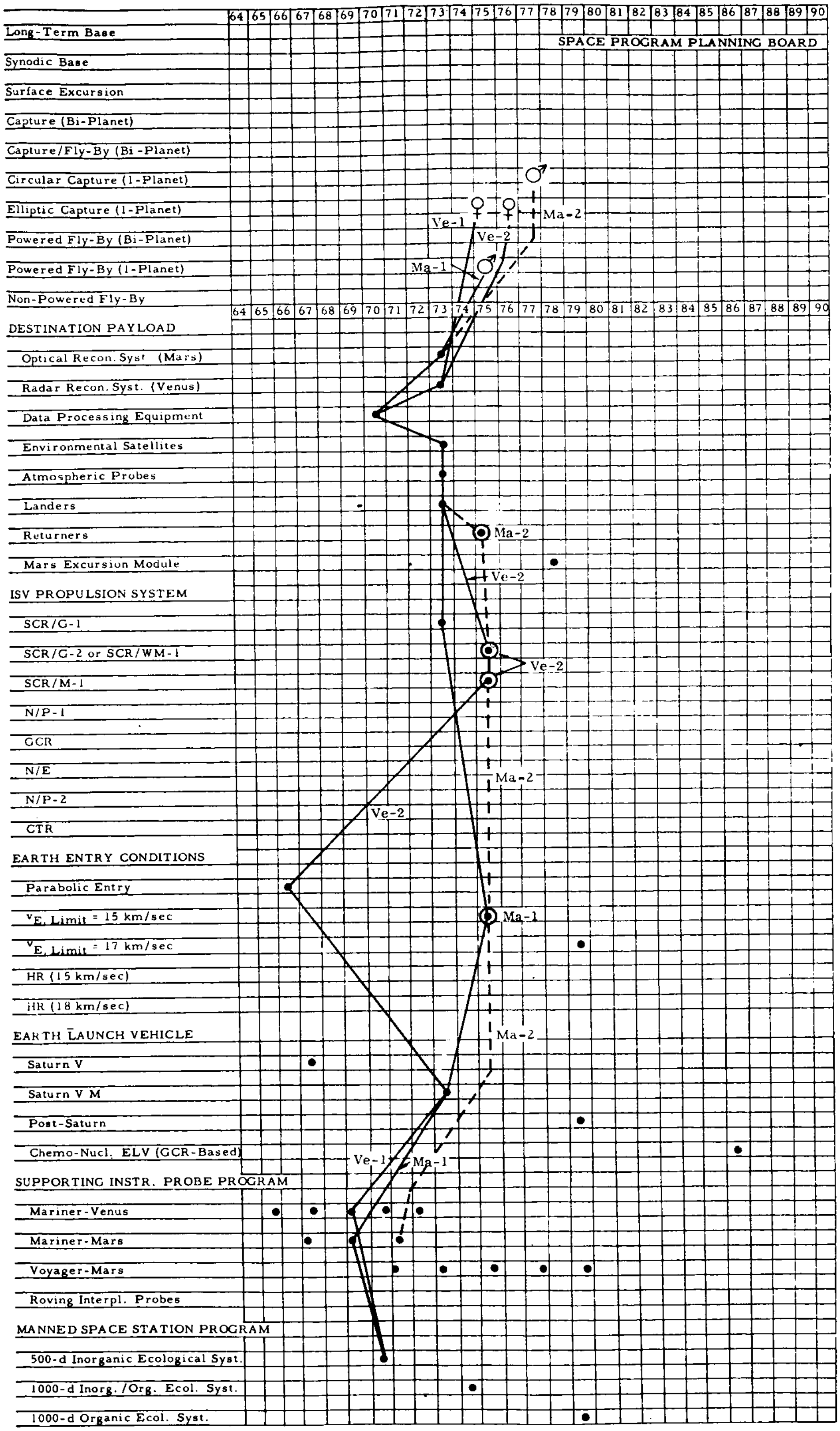


Fig. 5-4 Four Program Alternatives for Early Missions

Dev. Steps	Activity	Objective
D-1	Conceptual studies	Concept definition
D-2	Preliminary design	Baseline specifications
D-3	Supporting research and data acquisition	Provide necessary engineering "hard" data info.
D-4	Component dev. & testing	Establish component fabrication basis
D-5	Subsystem design & fab.	Establish tooling & fabrication experience
D-6	Subsystem testing	Shake down subsystems & provide data for final design
D-7	Subsystem prototype design and fabrication	Establish prototype fabrication capability
D-8	System design & fabrication	Establish basis for assembly, fabrication and checkout of system
D-9	System testing	Testing (incl. flight testing where re quired) and shake down of entire system. Facility checkout. Final changes.
D-10	Final design of system subsystem & components	Freezing design and discontinuing changes for the time being. Specification of operational hardware.
D-11	Fabrication	Manufacturing of operational hardware
D-12	Final checkout	Achievement of initial operational capability
D-1/D-2 = Progr. Definition Phase; D-3/D-8 = Dev. & Testing Phase; D-9/D-12 = Qualification Phase		

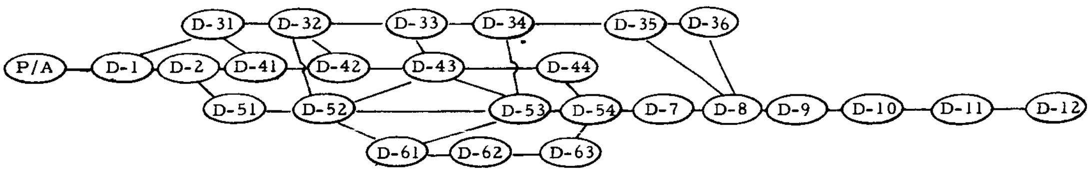


Fig. 5-5

ANDARDIZATION OF DEVELOPMENT PROCESS FOR PLANNING EVALUATION
JRPOSES AND SAMPLE NETWORK FOR SCHEDULE PROBABILITY ANALYSIS

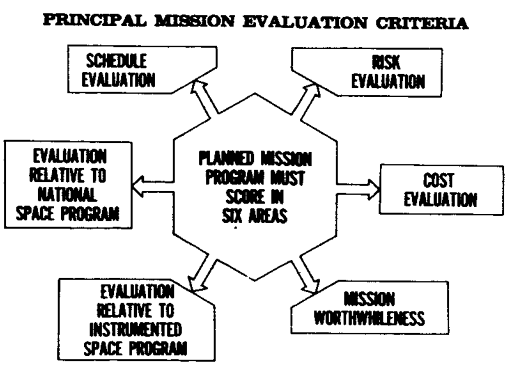


Fig. 5-3

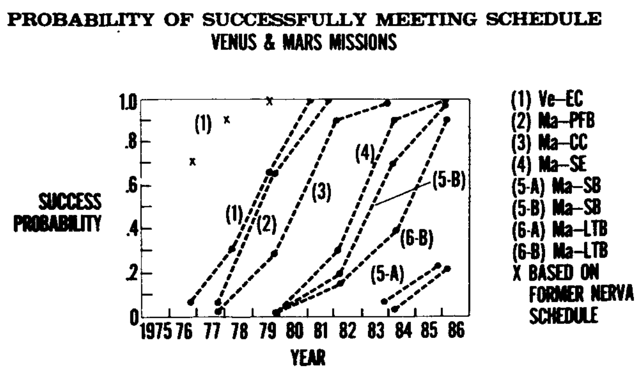


Fig. 5-6

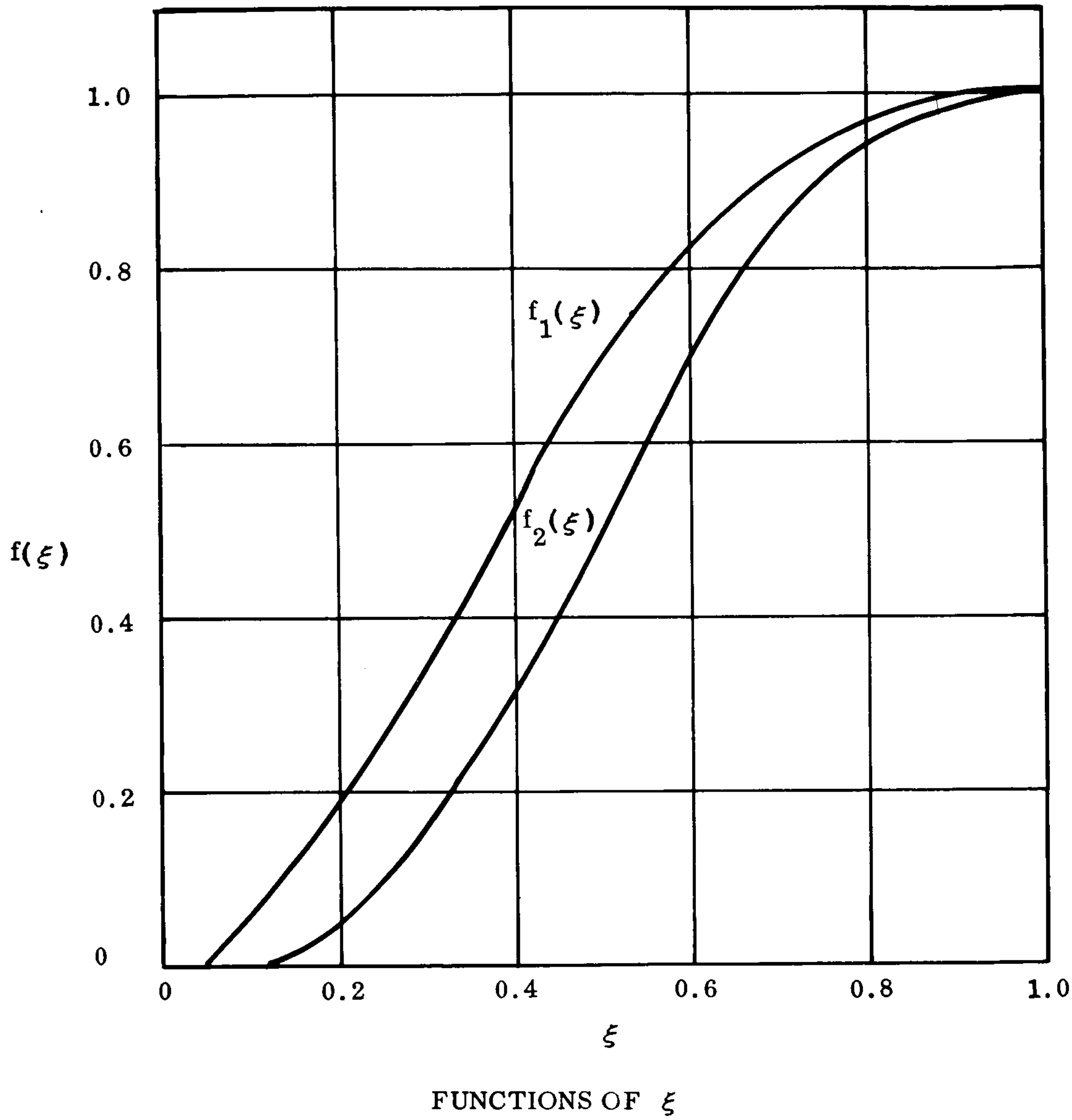


Fig. 5-7 SPENDING VS TIME FUNCTIONS

6 Vehicle-Payload Integration

The purpose of the vehicle-payload integration program is to provide the computer with a selective capacity wherever there is a situation in which the selection of a given vehicle depends not only upon mission energy and schedules, but also on compatibility with the payload it must carry. If the payload weight exceeds the carrying capability of a given vehicle, either the vehicle must be rejected or the payload changed. But this is only part of the problem. In some cases the payload weight carrying capability of a given vehicle is adequate, but not the volume of its payload section. Sometimes both, weight and volume are sufficient, but not the length of the payload section. For example, Saturn V imposes some volume and length constraints upon bulky, but not too heavy, payloads, such as sections of hydrogen carrying nuclear I/V's. In general, vehicles which have to traverse an atmosphere are more sensitive in this respect than are ISV's. In the nearer future, this places the primary emphasis on payload integration with ELV's. It requires that the payload sections of ELV's be evaluated with respect to three parameters, rather than weight alone, namely, weight, volume and length (or height) of payload section. It is further required that effective density data are derived from vehicle designs for sections of vehicles or entire vehicles, so that the "fit" (either weightwise or length-wise or volume-wise) of a given payload relative to a given ELV can be determined.

For orbital activities, especially OVAM of lunar and planetary vehicles, the question of choice of ELV can arise. Fig. 6-1 shows an example. Following flight analysis and payload weight determination, the orbital departure weight (ODW) of the ISV can be determined. If the ODW is too large for the available ELV's to permit DFM, the orbital vehicle-assembly mode (OVAM) must be employed. The choice may be, first between mating of two fueled modules in orbit or of fueling two modules which were delivered into orbit in already mated condition. In the latter case, one larger ELV may deliver the mated but empty modules and smaller ELV's may be used to fuel them. In the former case only one ELV-type needs to be employed, but that one may be larger, more expensive, not yet as reliable as the older smaller ones and it causes more orbital work in orbit.

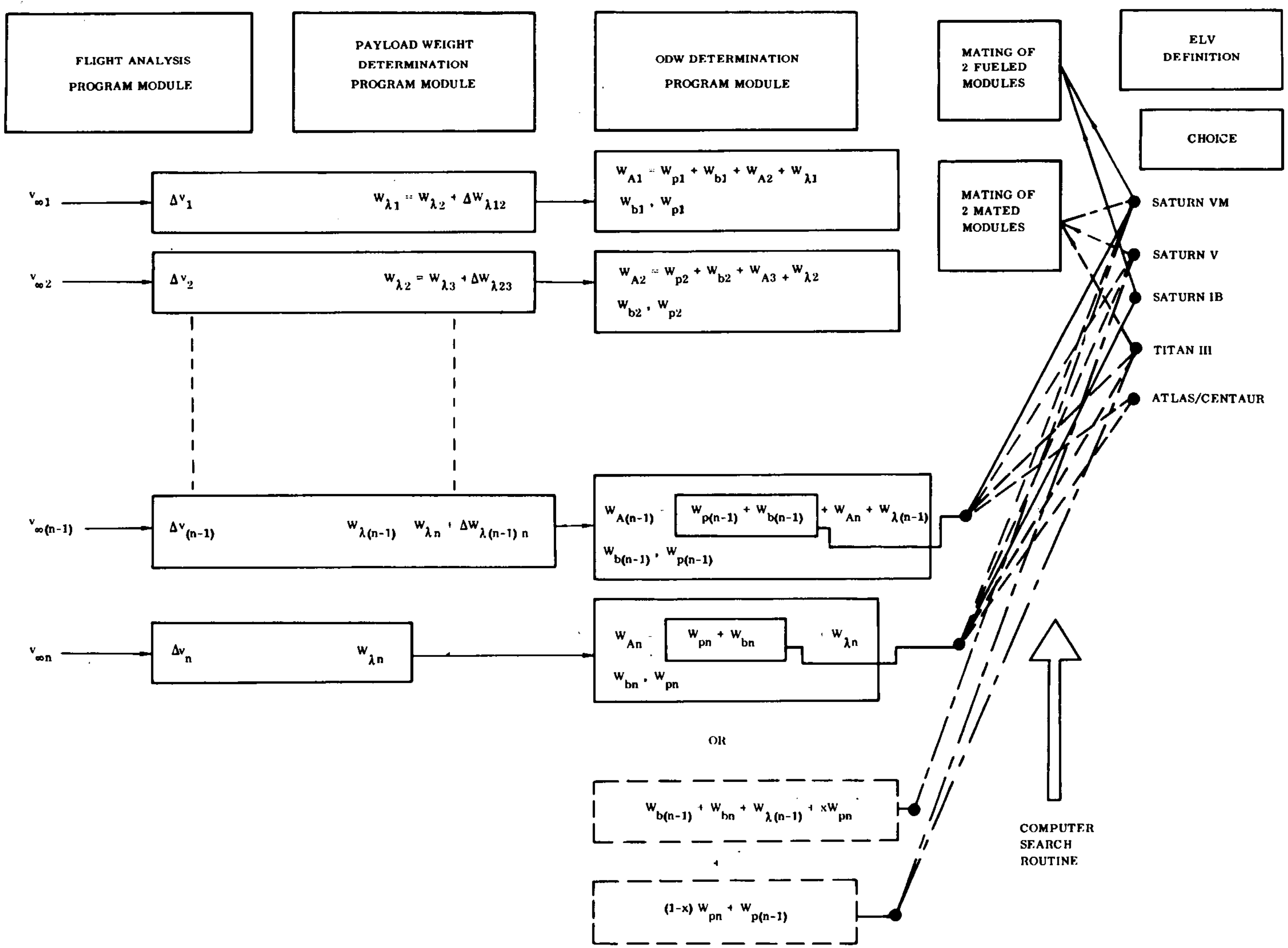


Fig. 6-1 PROGRAM MODULE TIE-UP

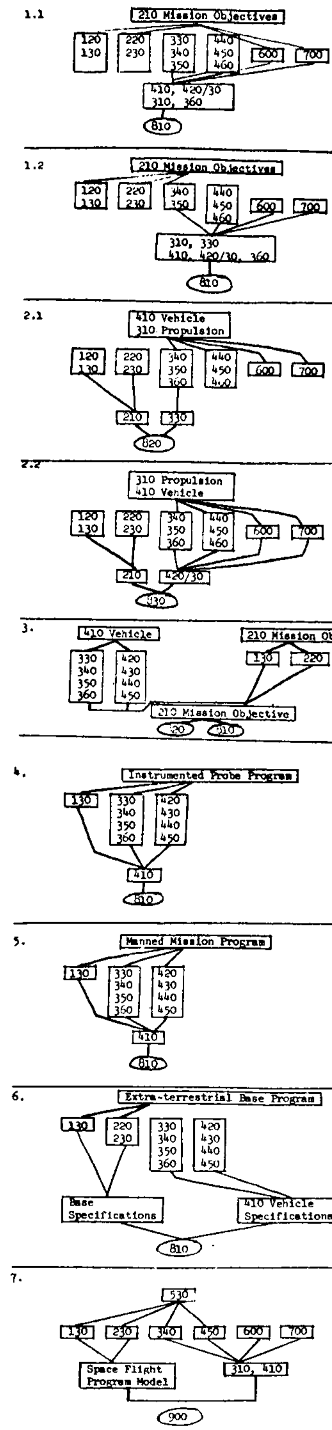
7 Technological Program Synthesis (TPS, Series 800) and Resources Program Synthesis (RPS, Series 900)

The objective of the TPS (Series 800) is to synthesize individual missions into a program and eventually into the model of a national space program. The term program is, therefore, used here to designate a sequence of missions representing an orderly progression in the exploration and utilization of space. An example of such a program model, concerned with the planetary mission program is shown in Fig. 7-1. By definition, series 800 deals with the technological aspects.

The TPS and RPS can be approached in several ways. In each case, however, one must distinguish between independent variable, parametric variable(s) and dependent variable(s). In this manner, differences in approach can most easily be defined and computer programs be arranged most effectively.

Tab. 7-1 presents a simplified summary of this approach. The TPS is divided into six groups. In group (1), the mission objectives represent the independent variable (that is, a given sequence of missions and mission objectives); and vehicles and propulsion systems are the parametric variables. In group (2) the situation is reversed. The effect on the evaluation objectives are pointed out. In group (3) a given vehicle model is evaluated against a series of reference velocities which constitute typical performance plateaus for mission groups; i. e. instead of being specified explicitly, mission groups are represented by overall velocity levels. In group (4) and (5) a distinction is made between unmanned and manned solar system exploration models; and in group (6) the independent variable refers to extra-terrestrial acitivities rather than missions. Group (7) finally comprises program syntheses based on resources models to check the feasibility of accomplishing a certain series of missions within a given budget. The schematics for these various approaches to mission synthesis are shown in Figs. 7-2 through 7-4.

No.	Independent Variables	Parametric Variables Primary Variable	 Secondary Variable	Dependent Variables	Valuation Objective
1.1	Mission Objectives Model	Vehicle	Propulsion	Mission Performance Characteristics Payload Requirements Program Cost Requirements Development Operation Schedule	Compare individual vehicles against given mission objectives
1.2	Mission Objectives Model	Propulsion	Vehicle	Mission Performance Characteristics Payload Requirements Program Cost Requirements Development Operations Schedule	Compare individual propulsion systems or engine systems against given mission objectives
2.1	Vehicle Model	Mission Objectives	Detail Propulsion Characteristics and/or Initial Vehicle Weight Scaling coefficients or I_{sp}	Mission Performance Characteristics Vehicle Payload Fraction Payload Requirements Operational Requirements Program Cost Requirements Schedule	Determine suitability of given vehicle - propulsion system combination for various missions and detail variations of the engine system
2.2	Propulsion System Model or Thrust System Model and Associated Vehicle Model	Mission Objectives	Vehicle Structure (scaling coefficient) or other suitable characteristic parameter, e.g. power generation weight (for nuclear electrical space vehicles)	Mission Performance Characteristics Vehicle Payload Fraction Payload Requirements Operation Requirements Program Cost Requirements Schedule	Determine the suitability of given propulsion system for various missions and variations of characteristic vehicle data
3.	Vehicle Model	Reference Velocities and Specific Impulse Ref. Velocities (a) 30,000 ft/sec (b) 45,000 ft/sec (c) 60,000 ft/sec (d) 90,000 ft/sec (e) 120,000 ft/sec (f) 150,000 ft/sec (g) 180,000 ft/sec	Payload Weight for Reference Velocities	Development Cost Development Schedule Probability of achieving IOC in certain year Mission capabilities Direct operations cost, based on given number of launches	Determine the mission worth of a given single vehicle
4.	Solar System Exploration Models	Interplanetary Vehicle		Minimum Orbital Departure Weight Development Cost Direct Operations Cost Availability	Determine desired progression in I/V development on the basis of desired limitation in orbital departure weight. Assess effect of resulting orbital departure weight upon ELV payload requirements.
	Solar System Exploration Models - Manned	Interplanetary Vehicle	Variations of vehicle and propulsion system characteristics	Minimum Orbital Departure Weight Development Cost Direct Operating Cost Availability	As above except related to manned mission requirements
6.	Extra-terrestrial Base Models Location: Earth Orbit Moon Planet Orbit Planet Surface Planetary Moon Surface	Life and degree of self-sufficiency of Base	Interplanetary Vehicle-Propulsion System Combination	Mission Performance Characteristics Supply Requirements (Life) Transport Requirements (number of launches) Transportation Cost Running Cost of Maintaining Respective Base	
7.	Resources Model (Man-hours; funds)	Space Flight Program Model	Vehicle & Propulsion Systems	Vehicle & propulsion systems cost Development Operation Orbital operations cost Extra-terrestrial Base Cost	Determine the adequacy of given resources against various space programs and their principal vehicle cost and operational cost



Tab. 7-1 INDIVIDUAL PROGRAM SYNTHESIS

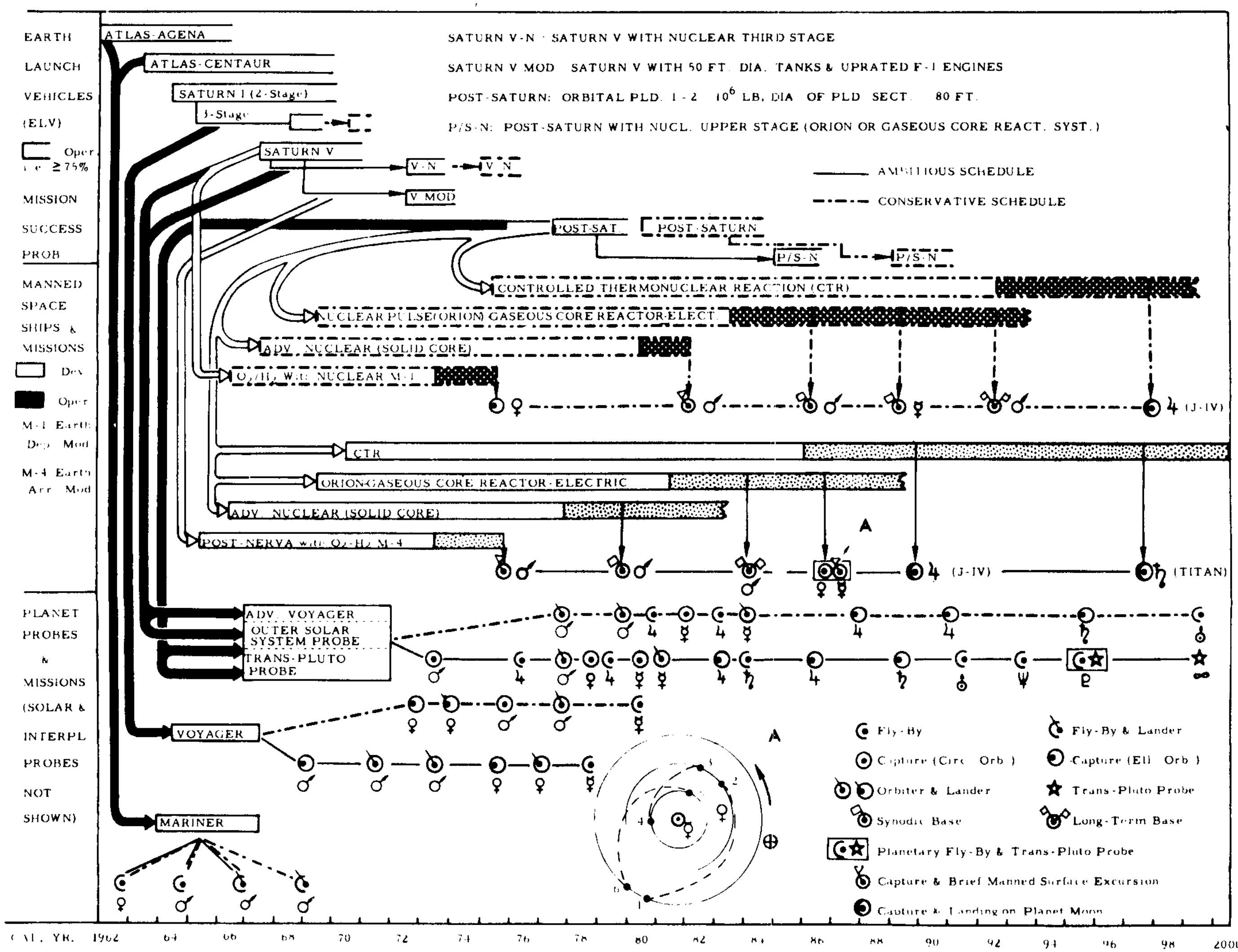


Fig. 7-1 **Example of an Integral Planetary Program Model, Correlating Development of ELV, Planet Probes, Manned Space Ships and Mission Groups as Function of Time in Two Schedules**

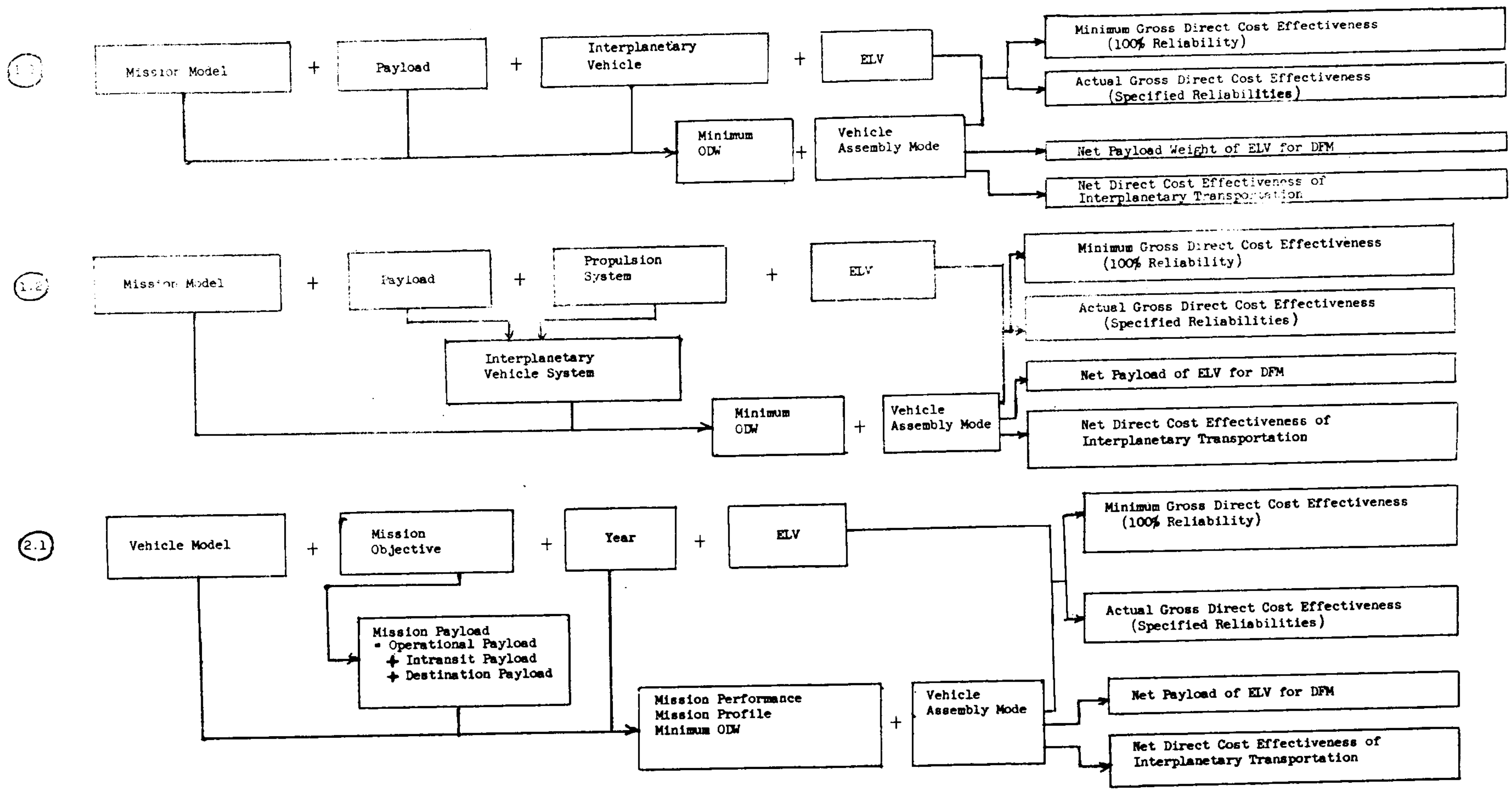


Fig. 7-2 INDIVIDUAL PROGRAM SYNTHESIS MODELS

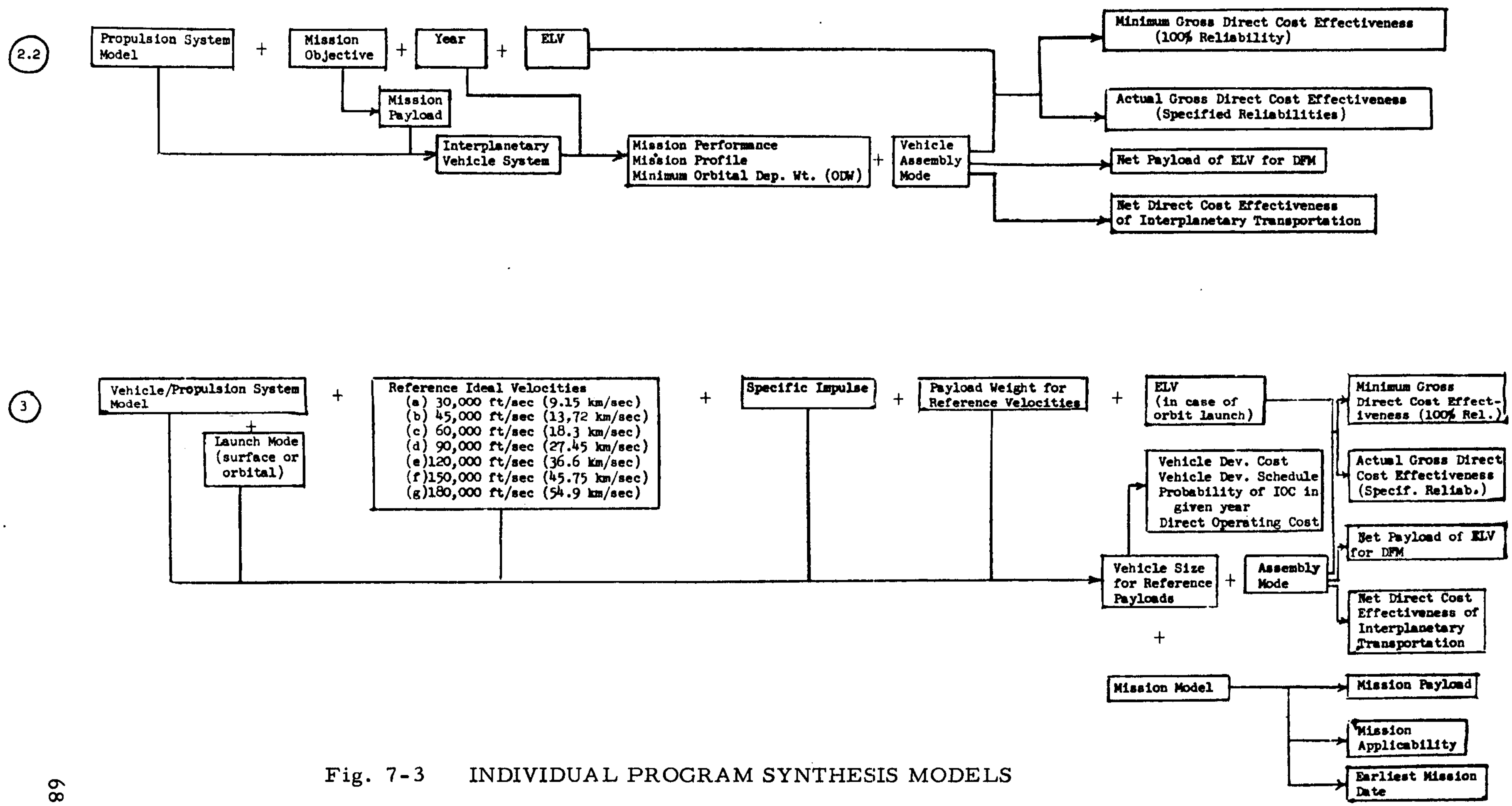


Fig. 7-3 INDIVIDUAL PROGRAM SYNTHESIS MODELS

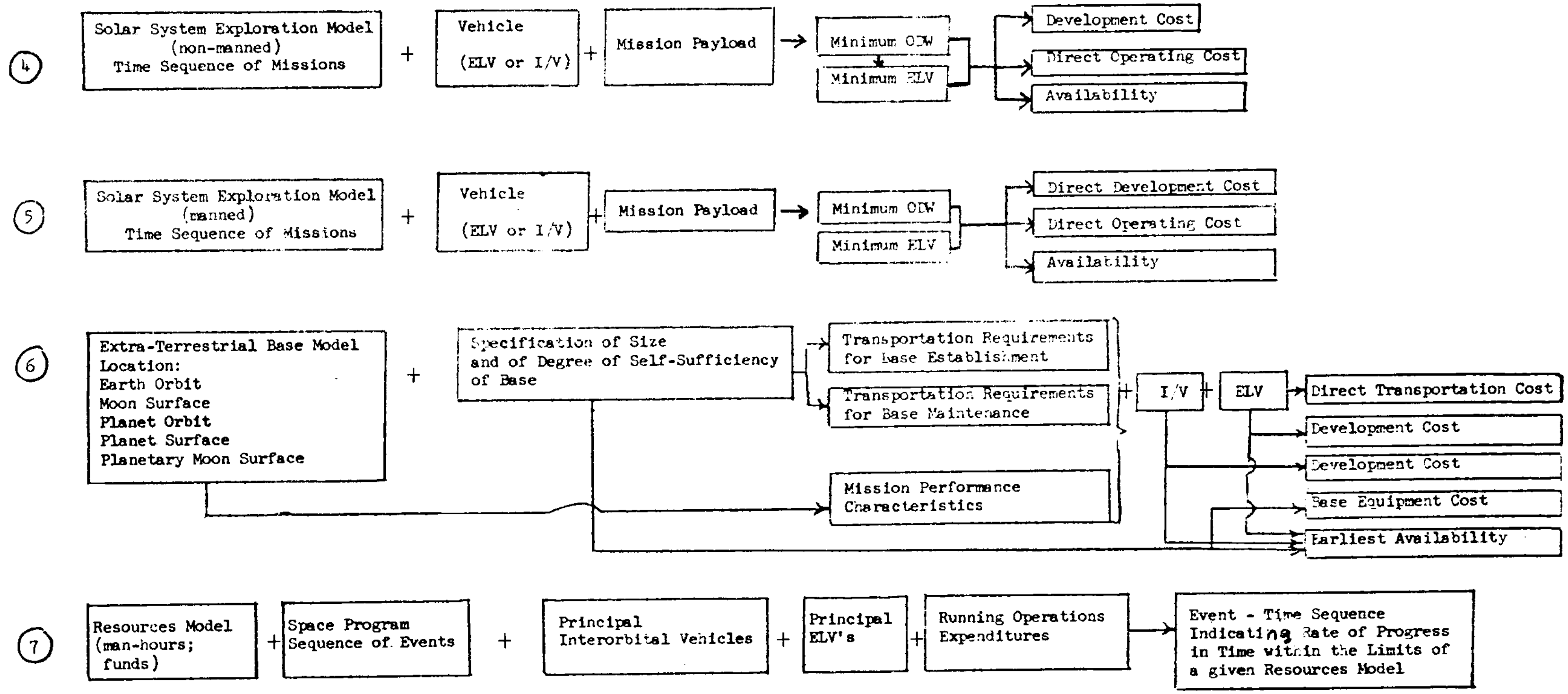


Fig. 7-4 INDIVIDUAL PROGRAM SYNTHESIS MODELS

Appendix A MISSION ORIENTED DEFINITIONS AND CODES

Tab. A-1 CODE OF DESTINATIONS

A. Bodies

He	Sun
Mo	Planet moon generally
Me	Mercury
Ve	Venus
Ea	Earth
Lu	Earth Moon
Ma	Mars
Ma-I	Phobos
Ma-II	Deimos
Ju	Jupiter
Ju-V	Jupiter moon V
Ju-I	Jo
Ju-II	Europa
Ju-III	Gangmode
Ju-IV	Callisto
⋮	etc.
Sa	Saturn
Sa-I	Mimas
⋮	etc.
Sa-VI	Titan
⋮	etc.
Ur	Uranus
Ur-I	Ariel
⋮	etc.
Ne	Neptune
Ne-I	Triton
Ne-II	Nereid
Pl	Pluto
Co	Comet in general
Co(Ju)	Comet of the Jupiter family
Co(Sa)	Comet of the Saturn family
⋮	etc.
Name	Specific comet
As-#	Asteroid (identified by number or name with As as prefix)

B. Space Regions

InMe	Intra-Mercury space
MeVe	Mercury-Venus space
VeEa	Venus-Earth space
EaMa	Earth-Mars space
⋮	etc.
As B	Asteroid belt (practically identical with MaJu space)
TrPl	Trans-Pluto space
ISS	Inner solar system
OSS	Outer solar system
XE	Extra-ecliptic space
IS	Inter-stellar space
IG	Inter-galactic space

Tab. A-2 PLANETARY ROUND-TRIP MISSION CLASSIFICATION

Class A	One-Planet Fly-By (Powered or Powerless)	
	A-1	Mono-elliptic both ways
	A-2	Mono-elliptic one way; bi-elliptic the other
	A-3	Bi-elliptic both ways
Class B	Bi-Planet Fly-By (Powered or Powerless)	
	B-1	All three transfers mono-elliptic
	B-2	One transfer bi-elliptic
	B-3	Two transfers bi-elliptic
	B-4	All three transfers bi-elliptic
Class C	One-Planet Capture	
	C-1 through C-3 as A-1 through A-3	
Class D	Bi-Planet Combination of Capture and Fly-By	
	D-1 through D-4 as B-1 through B-4	
Class E	Bi-Planet Capture	
	E-1 through E-4 as B-1 through B-4	
Class F	Tri-Planet Combinations of Fly-By and Capture	
	F-1	All four transfers mono-elliptic
	F-2	One transfer bi-elliptic
	F-3	Two transfers bi-elliptic
	F-4	Three transfers bi-elliptic
	F-5	All transfers bi-elliptic

Tab. A-3 DEFINITION OF MANEUVERS

Planetocentric (0-9)

0	Ascent to orbit
1	De-orbit and hovering; or from hovering into orbit
2	Plane change maneuver
3	Periapsis maneuver
4	Apoapsis maneuver
5	Intermediate (non-apsidal) Orbit change (planar)
6	De-orbit and landing
7	Rendezvous maneuver (with bodies of negligible gravitational field)
8	Establishment of circular orbit near planet
9	Establishment of elliptic orbit near planet

Planet (Primary) - Moon Maneuvers (10-19)

10	Injection into transfer orbit to Moon
11	Direct landing on Moon
12	Moon capture maneuver
13	De-orbit and hovering; or from hovering into orbit
14	De-orbit and landing (continuous thrust maneuver)
15	Ascent from Moon surface into Moon satellite re-orbit
16	Lunar orbital rendezvous maneuver
17	Selenocentric plane change maneuver
18	Selenocentric planar maneuver
19	Injection into transfer orbit to primary

Planetocentric-Heliocentric Maneuvers (20-29)

20	Injection into hyperbolic escape orbit from orbit about planet
21	Capture in circular orbit
22	Capture in elliptic orbit
23	Powered fly-by maneuver
24	Non-powered fly-by
25	Planet-approach retro-maneuver
26	Hyperbolic orbit to surface by retro-maneuver
27	Injection into hyperbolic escape orbit from orbit about a moon of not negligible gravitational field
28	Atmospheric braking
29	Atmospheric entry and descent

Tab. A-3 Definition of Maneuvers (Concluded)

Heliocentric Maneuvers

30	Plane change maneuver
31	Perihelion maneuver
32	Aphelion maneuver
33	Rendezvous maneuver (with bodies of negligible gravitational field)

Intra-Cluster (vehicles) or Intra-Convoy Maneuvers

40	Vernier maneuver of a vehicle relative to a reference vehicle (for maintanance of cluster or convoy conditions)
41	Vehicle to vehicle transfer (taxi or tug-boat flight)
42	Docking maneuver
43	Mating maneuver

Correction Maneuvers

HeC - # = Heliocentric correction maneuver No. . . . .
PlC - # = Planetocentric correction maneuver No. . . .

A given maneuver, if referred to in general, is referred to by number with the prefix M - , in order to avoid confusion (e.g. M - 20).

If a mission is described by the sequence of maneuvers, only the numbers need to be given, connected by plus-signs, provided no error or confusion is possible.

Example: Apollo - LEM - Mission: 0 + 10 + C-1 + C-2 + 12 + 14 + 15 + 16 + C-3 + C-4

If a mission involves several bodies of the same type, e.g. two planets, the same numbers (maneuvers) may have to be used several times. In this case, the planetary code designation will be used as prefix.

Example: a round-trip capture mission to Venus, involving circular capture and return to the Earth's surface with a retro-maneuver preceding atmospheric entry: Ea-20 + EaC-1 + EaC-2 + HeC-1 + HeC-2 + VeC-1 + Ve-21 + Ve20 + VeC-2 + HeC-3 + HeC-4 + EaC-3 + Ea-25. This mission is seen to contain two correction maneuvers while the vehicle is still in a geocentric (hyperbolic) escape orbit, two heliocentric correction maneuvers, one Venus approach correction maneuver and a number of correction maneuvers on the return flight.

Tab. A-4 ACCELERATION LEVELS

HAM	High Acceleration Mission
LAM	Low Acceleration Mission
PAM	Powered All-the-way Mission

Tab. A-5 PRINCIPAL MISSION OPERATIONS

PSL	=	Planet surface launch (e.g. EaSL = ascent from Earth surface)
SD	=	Surface descent (e.g. MaSD = descent to Mars surface; LuSD = descent to surface of Moon)
OL(#)	=	Orbit launch (# = orbit distance or periapsis distance from planet or moon in terms of radii of the body; e.g. VeOL (1.1) = launch from orbit about Venus at 1.1 Venus radii distance). If necessary, distinguish between COL (circular orbit launch) and EOL (elliptic orbit launch)
CC (#)	=	Circular (orbit) capture (# = capture distance in orbit radii; e.g. MaCC (1.3)
EC (#/#)	=	Elliptic (orbit) capture (#/# = periapsis distance in orbit radii and ratio n of apoapsis to periapsis; e.g. Ve EC (1.1/8) = Venus elliptic capture ($r_p^* = 1.1$; $r_A^* = 8.1.1 = 8.8$)
FB (#)	=	Fly-by (non-powered) (# = periapsis distance in orbit radii; e.g. Ju FB (1.5))
PFB (#)	=	Powered fly-by (# = periapsis distance in orbit radii)
HE (#)	=	Hyperbolic entry into atmosphere (# = limiting entry velocity in EMOS (Earth Moon Orbital Speed)
PE (#)	=	Parabolic entry into atmosphere
RM	=	Retro-maneuver
Hov (#)	=	Hovering (i.e. elimination of circumferential speed in orbit and falling slowly or preventing fall by supporting thrust) (# = hovering period in minutes)
Lu SL	=	Launch from surface of Moon

Tab. A-5 Principal Mission Operations (Concluded)

SE	=	Surface excursion
SS	=	Hard landing
S	=	Soft landing
RV	=	Rendezvous
IC'	=	Intercept (destructively)
OVAM	=	Orbital vehicle-assembly mode
COVAM	=	Capture orbit vehicle-assembly mode
IVAM	=	Interorbital vehicle-assembly mode
A B ( )	=	Atmospheric braking; ( ) = designation of orbital energy or ratio of apoapsis to periapsis of orbit after atmospheric brakin
He OC	=	Heliocentric orbit change, generally
PB	=	Perihelion braking (maneuver)
PA	=	Perihelion acceleration
AB	=	Aphelion brake
AA	=	Aphelion acceleration

Tab. A-6 SYMBOLS USED IN SCHEDULES

x	Fly-By
⊗	Powered fly-by
◬	Elliptic capture
⊙	Circular capture
◓	Surface excursion
◑	Synodic base
●	Long-Term base
•	Supply flight for LTB

Tab. A-7 EXAMPLES OF CODIFIED MISSION DESCRIPTIONS

1. Mission to Mars, circular capture at 1.3 radii distance, surface excursion, perihelion brake at return transfer: (C-2)-Ma CC (1.3) SE - PB

2. Mission to Mars, perihelion acceleration in Mars-bound transfer, elliptic capture $r_p^* = 1.3$, $r_A^* = 13$: (C-2) - PA - Ma EC (1.3/10)

3. Mission to Venus, mono-elliptic both ways, fly-by at two radii distance: (A-1) - Ve FB (2)

4. Bi-planet mission: mono-elliptic transfer to Mars, circular captu at $r^* = 1.3$, mono-elliptic transfer to Venus, powered fly-by at r^* mono-elliptic transfer to Earth and hyperbolic entry at limiting sp of 0.45 EMOS: (D-1) - Ma CC (1.3) - Ve PFB (1.5) - Ea HE (0.45)

5. Bi-planet capture mission: Heliocentric orbit change on way to Ma circular capture at Mars at $r^* = 1.3$, monoelliptic transfer to Ver elliptic capture, $r_p^* = 1.1$, n = 8, Earth-bound transfer with apheli acceleration and hyperbolic entry into Earth atmosphere at 0.4 EM limiting velocity:

(B-3) - He OC - Ma CC (1.3) - Ve EC (1.1/8) - AA - HE (0.4).

Tab. A-8 MISSION MODE DEFINITIONS

<u>Mission Mode</u>

- By Vehicle Assembly
  - Convoy
    - Direct Flight Mode (DFM)
    - Orbital Vehicle-Assembly Mode (OVAM)
    - Interorbital Vehicle Assembly Mode (IVAM)
    - Capture Orbit Vehicle Assembly Mode (COVAM)
  - Multiplex
    - Duplex
    - Other
- By Target Planet Capture
  - Retro-Thrust Maneuver (RTM)
  - Atmospheric Braking (AB) to either $\epsilon = -.04$ ($n = 49$); (AB(49)) or to $n \sim 1.5$ (AB(1.5))
- By Heliocentric Transfer
  - Mono-Elliptic
  - Bi-Elliptic
  - Tri-Elliptic
    - Powered Maneuver
    - Powered Fly-By
    - Fly-By
- By Earth Return
  - Retro-Maneuver to Parabolic
  - Retro-Maneuver to Hyperbolic
  - Direct Hyperbolic Entry
  - Hyperbolic Rendezvous in Unretarded Hyperbola
  - Hyperbolic Rendezvous in Retarded Hyperbola
  - Elliptic Rendezvous
  - Circular Capture

26993

PART 16

A STUDY OF ELECTRICAL PROPULSION IN THE SPACE PROGRAM

by

B. Pinkel and D. L. Trapp
The RAND Corporation
Contract No. NAS8-11081

A STUDY OF ELECTRICAL PROPULSION IN THE SPACE PROGRAM

B. Pinkel and Donald L. Trapp*

The RAND Corporation, Santa Monica, California

INTRODUCTION

We are reporting on a study of the place of electrical propulsion in the space program.** This study is proceeding in two phases. In Phase I, we were asked to delineate the problem. The objective in this phase was twofold: (1) to determine what conclusions could already be drawn from the existing information and (2) to list additional studies that should be made to complete the analysis. The report on Phase I has been issued, and we are now in Phase II doing the recommended studies.

Let us refer now to Fig. 1. In our study of the existing literature we found that there were a large number of space missions described where electrical propulsion had a weight advantage over other propulsion systems; namely, that it could carry the required payload to the required destination with less initial gross weight in Earth orbit. However, a large part of the weight of the vehicle propelled by the electrical propulsion system was contributed by the nuclear-turboelectric machinery which had a very high cost, whereas in the competing nuclear rocket system the bulk of the vehicle weight was made up of low cost propellant. It was not clear, therefore, that a weight advantage also translated into a cost advantage, and we decided that in the Phase II study it would be desirable to look at the cost comparison associated with the various mission studies. We were asked in the contract to

---

*Any views expressed in this paper are those of the authors. They should not be interpreted as reflecting the views of The RAND Corporation or the official opinion or policy of any of its governmental or private research sponsors. Papers are reproduced by The RAND Corporation as a courtesy to members of its staff.

**Under contract with Marshall Space Flight Center, NASA number NAS 8-11081.

1. MISSION COMPARISONS
   - LUNAR TRANSPORT
   - DEEP SPACE PROBES
   - MANNED MARS EXPEDITION
2. DEVELOPMENT PROGRAM AND COST
3. RELIABILITY
   - ENGINE RELIABILITY
   - MISSION RELIABILITY
4. IMPROVED POWER CYCLES

STUDY ITEMS

Fig. 1

emphasize the Manned Mars Expedition. However, we felt that the case for electrical propulsion rests on its utility over the spectrum of possible space missions and decided to look very briefly also at the lunar logistic transport and the deep space probe missions.

In the course of the Phase I study, a rough cut at the development program and cost was made. This analysis is being refined in Phase II. Even on the basis of a preliminary consideration of the development program, it was apparent that an expensive program would provide a system of only moderate reliability. This came from the fact that the electrical propulsion system is required to have a 10,000-hour operational duration (i.e. 400 days). Many full duration tests, each running 400 days, cannot be bought even with an expensive and lengthy program; e.g. about $3 billion and 12 years, respectively. Thus while reliability is an important problem in all propulsion systems, it takes on an added dimension in the electrical propulsion case and raises a number of interesting questions: (1) Can a high mission reliability utilizing propulsion system components of moderate reliability be obtained and what is the cost? (2) Can one derive more reliability information out of a development program than is the current practice by a more complete utilization of the data? (3) What does reliability analysis indicate about how development programs for long endurance systems should be planned?

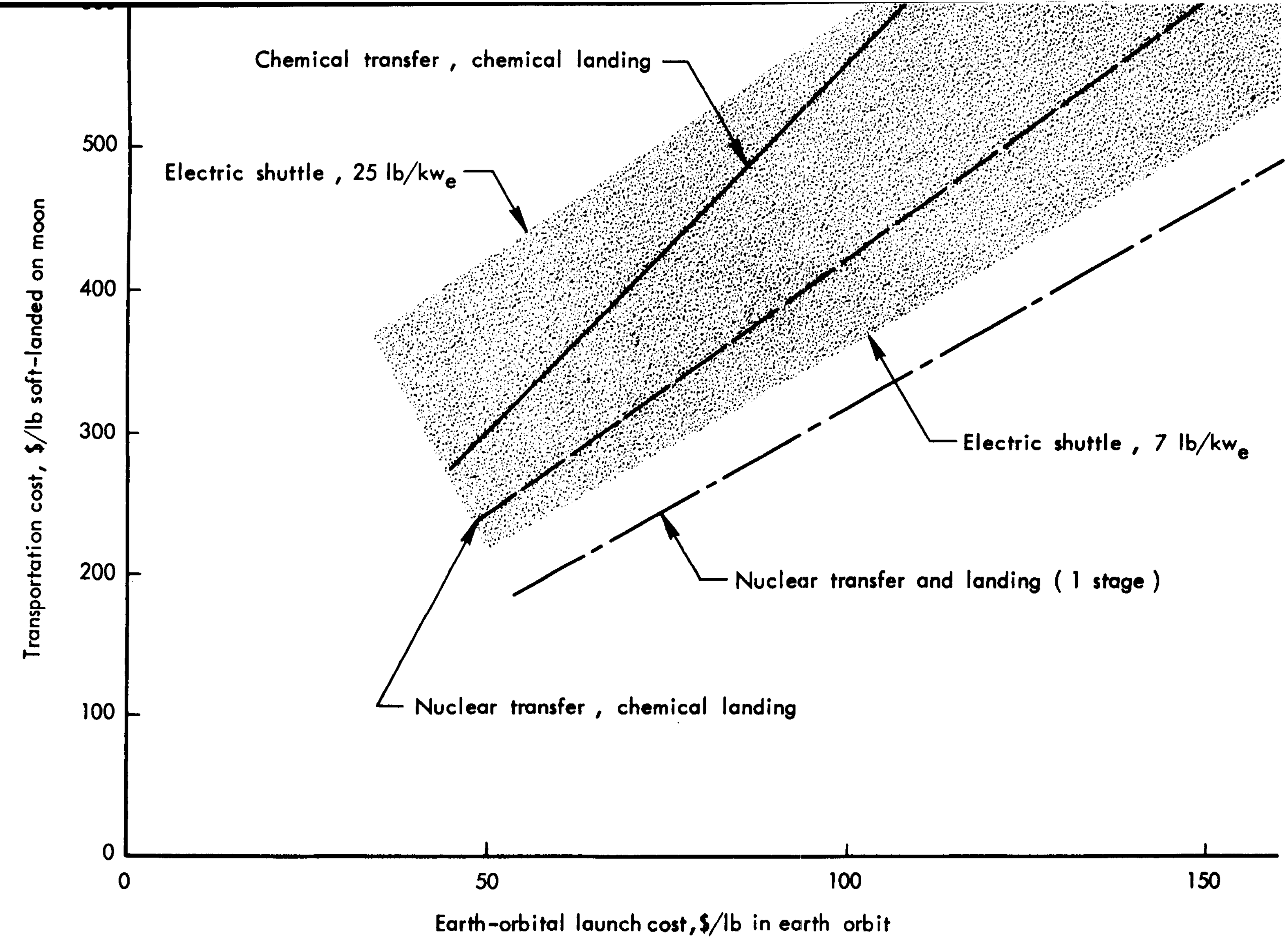
Finally, the current development effort on large space power systems emphasizes the Rankine cycle; it appeared advisable to look at the potential of some of the other power systems. This in essence is the program for our Phase II study and also for our presentation here today. Since we have not completed this study, our conclusions should be regarded as purely tentative.

LUNAR LOGISTICS TRANSPORT

With reference to the lunar logistic transport mission, it has been suggested that the electrical propulsion system be employed to power a shuttle vehicle that goes back and forth from Earth orbit to Moon orbit. Payload is transported from Earth to this shuttle by means

of a chemical rocket system, and then transferred from the shuttle to the Moon by means of either a chemical or nuclear rocket. The shuttle continues to operate back and forth until the engine has exhausted its 10,000 hours of life. This system is contrasted with a more conventional type utilizing either chemical or nuclear rockets for the space transport and lunar landing operations. The electrical propulsion system shows a payload advantage over the other systems; i.e. for the same weight lifted from the surface of the Earth, it would transport 40 per cent more payload to the Moon than the nuclear rocket. The advantage was even greater relative to the chemical system.

Figure 2 shows the transport cost of soft-landing payload on the Moon in dollars per pound of payload landed. The transport cost includes the cost of boosting the payload into Earth orbit and the costs of the transport vehicle, its propulsion system, propellant, and payload. This cost is plotted against the cost of placing mass in initial Earth orbit in dollars per pound. The shaded area represents the electrical propulsion system with chemical rockets used for the lunar landing portion of the operations. The lower edge of that area corresponds to a power plant weight of 7 lb/kw of electrical energy. The case of the nuclear rocket system for the space transport and lunar landing operations is represented by the bottom line. A specific impulse of 800 sec was assumed. (In the other figures shown here, a specific impulse of 800 sec will also apply for the nuclear rocket unless the chart otherwise specifies.) It appears that the nuclear rocket system provides a lower cost than even the most optimistic electrical system. The nuclear transfer to the vicinity of the moon coupled with the chemical landing system and the chemical transfer with the chemical landing fall within the shaded band for the electrical system. It may be argued with some justice that these latter curves for the nuclear and chemical systems represent a more equitable comparison with the electric system in that they also use a chemical system for the lunar landing operation; a nuclear landing may impose radiation shielding difficulties. Even in this case, Fig. 2 supports the conclusion that the electrical propulsion system does not appear to promise an important economic advantage in the lunar transport operation.



COMPARISON OF CHEMICAL, NUCLEAR, AND ELECTRIC LUNAR TRANSPORT SYSTEMS

Fig. 2

A brief discussion will now be made of the specific weight of the system; namely, the pounds per kilowatt of electrical energy. This is a parameter that will recur throughout the figures. The values indicated as parameters in Fig. 2 are the pounds per kilowatt of just the electrical power source. During the period of the Phase I study, the contractors concerned with large space power system development were estimating a value of 10 lb/kw for the power generator. When to this are added the weights of the power conditioner (which transforms the electrical current to accommodate the thruster), the thruster, and the shield, and when the inefficiencies of the power conditioner and thruster are also considered, the 10 lb/kw for the power source increases to 20 lb of over-all system weight per kilowatt of jet power. From this point on, reference will be made to this over-all system specific weight which in some of the figures will be labeled α_j. Thus 20 lb/kw of jet power represented the estimates at the time that Phase I was being performed. Since that time, the contractors have been making more detailed design studies leading to more sophisticated configurations from the standpoint of improving reliability, facilitating development and meeting operational loads and other requirements. Their present specific weight estimates for the over-all system are between 30 and 40 lb/kw of jet power, and may approach 40 lb/kw of jet power in the transition from the design phase to developed hardware. On the other hand, space power systems are in their infancy and a technological breakthrough might reverse this trend.

SPACE PROBE MISSION

The next mission to be discussed is the deep space probe. The literature indicates that the advantage of electrical propulsion over competing systems increases as the mission becomes more difficult. Although we are covering a spectrum of such missions in the study, an example of a difficult mission is presented here; namely, that of placing a large payload in a near Jupiter orbit. The results of this study are displayed in Fig. 3. Two boosters were considered for the probe--the Saturn 1B and the Saturn 5. The payloads in low-Jupiter orbit for

	TOTAL TIME (DAYS)	PAYLOAD IN JUPITER ORBIT (1000 lb)			
		ELECTRICAL PROPULSION			NUCLEAR ROCKET
		$\alpha_i = 10$	$\alpha_i = 20$	$\alpha_i = 40$	
SATURN IB	400	0	0	0	0
	800	8.3	2.8[(a)]	0	0
SATURN V	400	0	0	0	1.2
	800	62.	21.[(b)]	0	6.7[(c)]

COST PER POUND OF PAYLOAD IN JUPITER ORBIT
(a) $9,200 (b) $3,710 (c) $9,300

JUPITER ORBITER MISSION (ORBIT AT 2 JUPITER RADII)

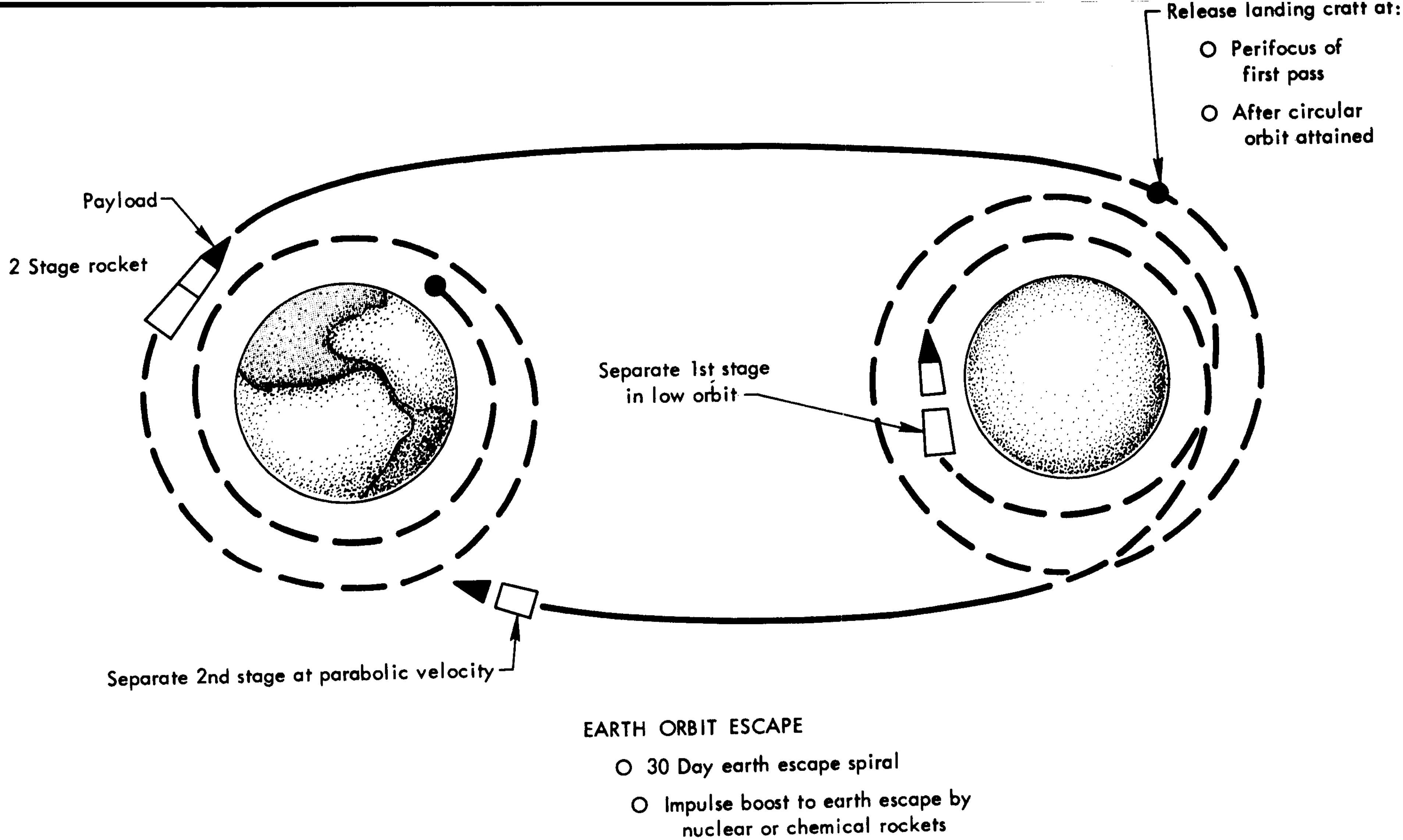
Fig. 3

The electrical propulsion case for several values of the parameter α_j are compared with the corresponding payloads for the nuclear rocket case. Using a Saturn 5 as the booster and α_j as 20 lb/kw_j, we see that the electrical propulsion promises to transport three times as much payload in Jupiter orbit as does the nuclear rocket at a cost per pound of payload in Jupiter orbit of about a third for the electrical propulsion relative to the nuclear rocket. The cost advantage reflects the 3 to 1 payload advantage because most of the cost is in the Saturn 5 booster. As the specific weight of the electrical system degrades to 40 lb/kw_j, its payload and cost advantage disappears. Thus, the payload and cost advantage for the electrical propulsion system is very uncertain in this space mission.

MANNED MARS EXPEDITION

Assumptions

The Manned Mars Expedition was considered in much greater detail, and Fig. 4 displays some of the assumptions that were made in this study. The objective was to transport a six-man crew to Mars and back with 86,000 lb allowed for the operation at Mars. Starting in initial Earth orbit, three alternative means of propelling the vehicle to Earth escape were considered. The first alternative was to use the electrical propulsion system. This results in a slow spiral which takes 30 days to attain the escape velocity and places the vehicle for an appreciable period of time in the Van Allen radiation belts. The second alternative was to project the vehicle to escape velocity by means of a high thrust nuclear rocket. It will be shown later that this results in a lower initial gross weight and a lower cost. The third alternative was to use a chemical rocket for the Earth escape trip and this also seemed to be advantageous in cost, but not in initial weight in Earth orbit. The remainder of the mission in each case is performed by the electrical propulsion system. On the first pass around Mars, the landing craft are separated and the exploratory operation takes place simultaneously with the spiraling in of the vehicle to a parking orbit. Because 86,000 lb



TYPICAL ELECTRIC ROCKET MISSION PROFILES

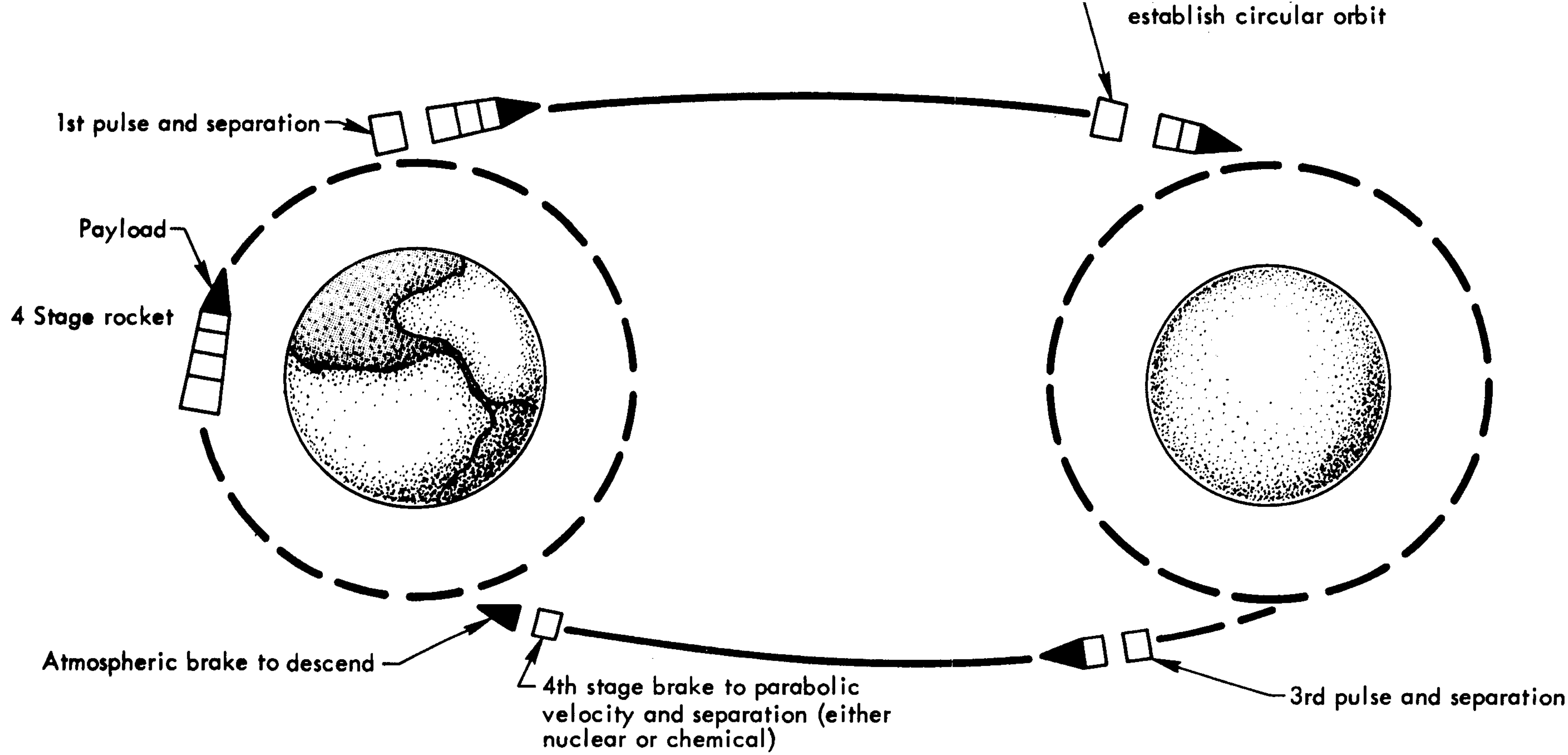
Fig. 4

are expended in the Mars operation and because the vehicle weight is also decreased by the propellant consumed in the trip to Mars, it pays to reoptimize the vehicle weight distribution at Mars departure by discarding a portion of the propulsion system; e.g. from 25 to 50 per cent of the propulsion system is discarded at the start of the return leg, depending on the specific mission under examination. The vehicle is brought into the vicinity of the Earth atmosphere at parabolic velocity, the landing capsule is separated and utilizes atmospheric braking for the descent.

Figure 5 shows the flight profile for the nuclear rocket system in the same Mars mission. Again, we are transporting a six-man crew, and 86,000 lb will be utilized in the Mars exploratory operation. The vehicle, in initial Earth orbit, is assumed to have a four-stage nuclear rocket. The first stage fires, boosting the vehicle into its trajectory to Mars. The second one fires to establish the vehicle into Mars orbit; the third one fires on the return trip; and the fourth one fires to slow the vehicle to parabolic velocity for expediting entry into the atmosphere. Each stage separates after firing to avoid afterheating and radiation problems.

Performance Comparisons

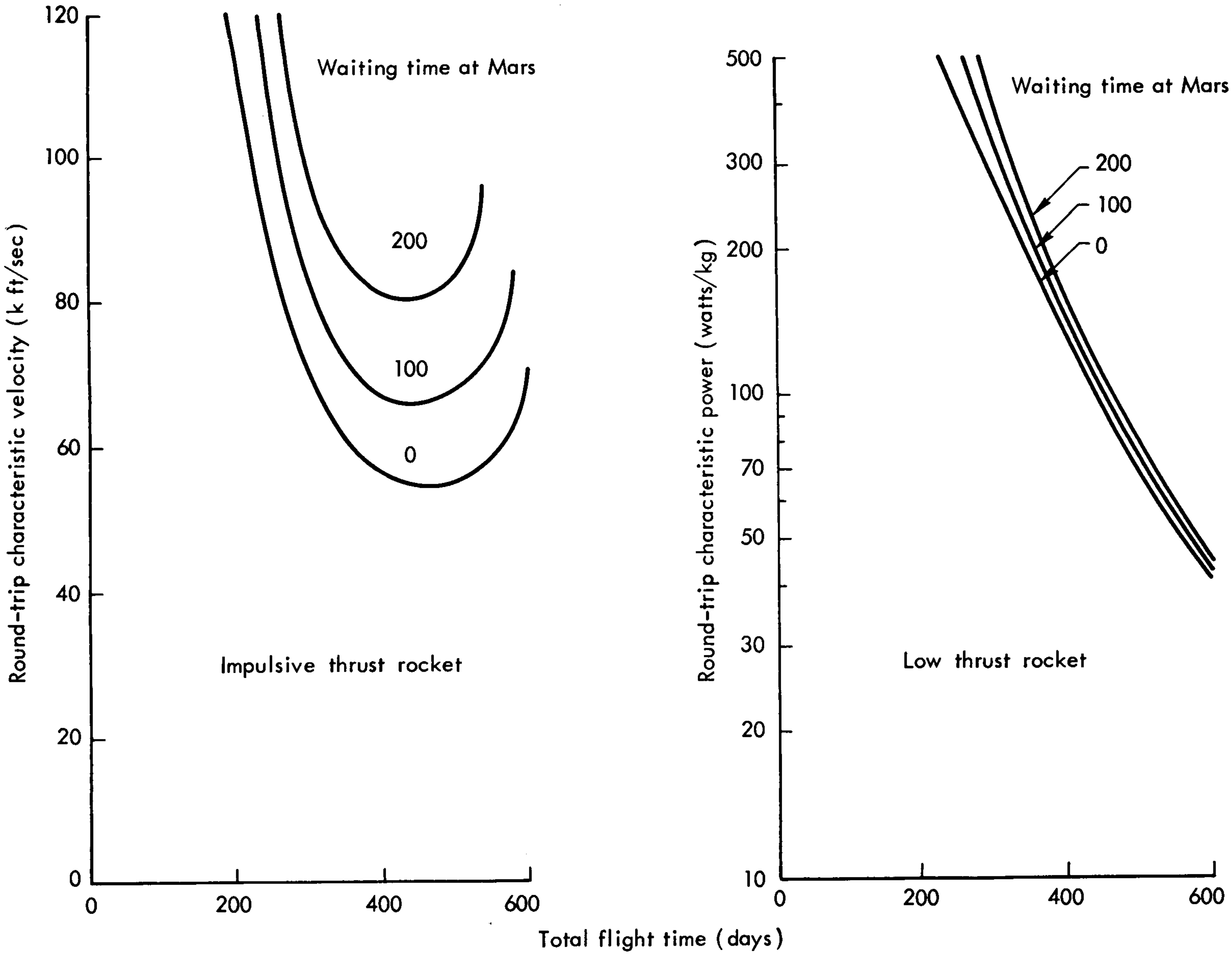
Figure 6 contrasts some of the important performance parameters associated with these two modes of performing the Mars mission. The characteristic round-trip velocity for the nuclear rocket system is shown plotted against the total flight time for the trip, for various stay times at Mars. The lower the characteristic velocity, of course, the lighter is the initial weight of the vehicle required to perform the mission so that it is advantageous to design for the minimum of these curves which occurs between 400 and 500 days of total flight time. We have considered only flights that do not require excessive stay time at Mars. The characteristic velocity and hence the gross weight of the vehicle is very sensitive to the waiting time at Mars. On the right-hand side of Fig. 6, is plotted a quantity we call characteristic power. Mathematically, it is the integral of the acceleration squared of the vehicle with respect to the flight time. It plays the same role, with respect to the electrical propulsion system, that characteristic velocity



Landing Operation
- O Two landing craft to surface
- O Wait time 0 to 200 days
- O Weight change 86,000 lb
- O Rendezvous by chemical rocket

TYPICAL NUCLEAR ROCKET MISSION PROFILES

Fig. 5


Waiting time at Mars
200
100
0
Impulsive thrust rocket
Round-trip characteristic velocity (k ft/sec)
120
100
80
60
40
20
0
0
200
400
600
Waiting time at Mars
200
100
0
Low thrust rocket
Round-trip characteristic power (watts/kg)
500
300
200
100
70
50
30
20
10
0
200
400
600
Total flight time (days)

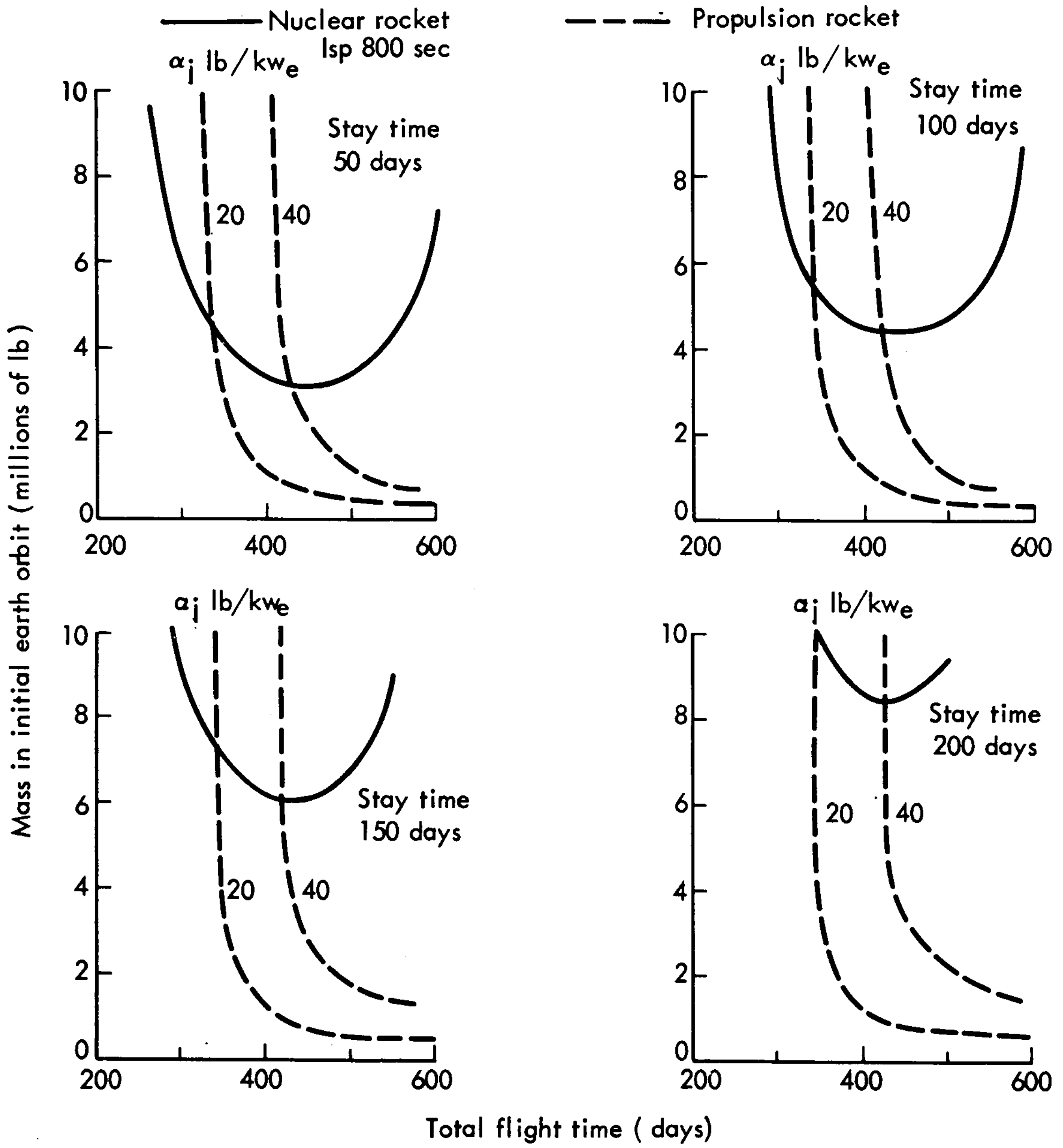
does in the case of the high-thrust propulsion systems, in that the lower the characteristic power, the less is the initial gross weight of the vehicle required to do the job. An interesting differences between the curves for characteristic power and characteristic velocity is evident. The characteristic power does not exhibit a minimum; the gross weight of the vehicle can be continually reduced by increasing the flight time for the trip. We will take advantage of this characteristic in our discussion of mission reliability. We also note that this system is much less sensitive to stay time at Mars than is the nuclear rocket system.

Figure 7 shows a comparison of the initial vehicle weight in Earth orbit for electrical and nuclear rocket systems for the Mars mission plotted against the total flight time and various stay times at Mars. A minimum initial weight in Earth orbit is evident for the nuclear rocket at a flight time of about 500 days, whereas for the electrical rocket case the initial vehicle weight continuously decreases as the flight time increases. Also as we increase the stay time to 50, 100, 150, and 200 days, there is a very substantial increase in the weight of the vehicle for the nuclear system whereas for the electrical system it changes only slightly.

Cost Comparisons

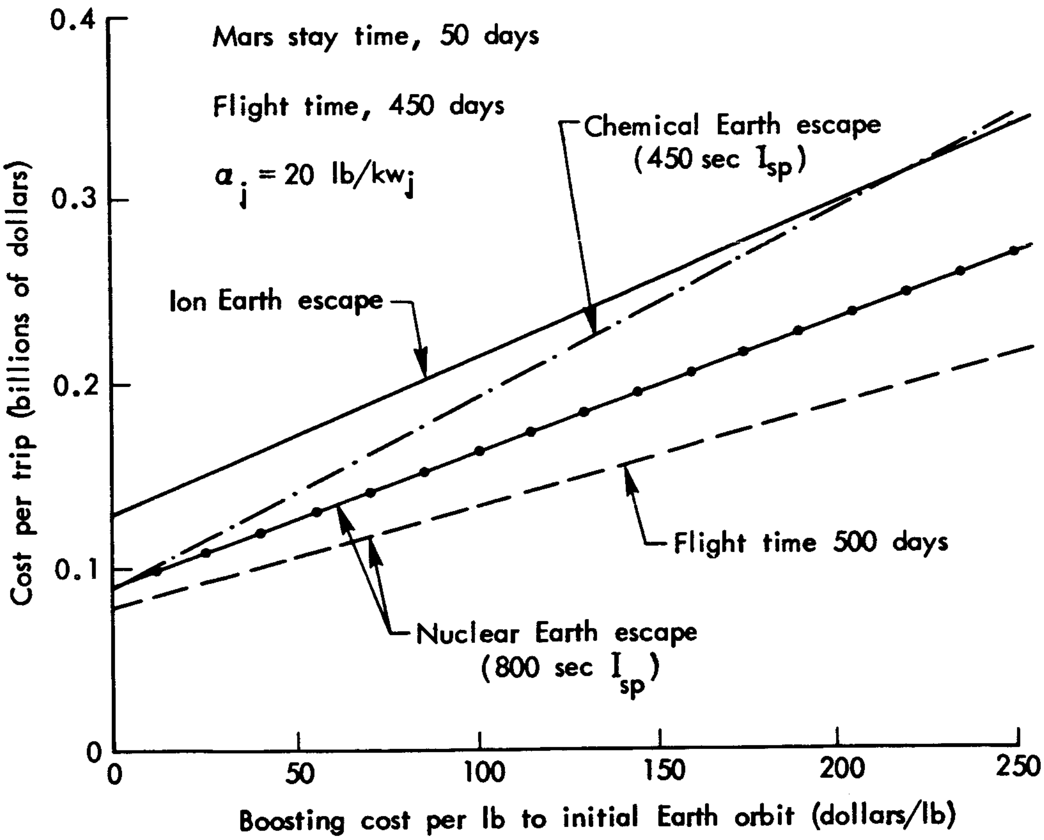
In Figs. 8 to 11, the operational cost of a Mars trip is given with respect to several parameters. This operationsl cost includes vehicle and system fabrication, propellants, and boosting cost to initial Eatth orbit, but no research and development costs.

The cost per trip for the electrically propelled vehicle is given as a function of the boosting cost to initial Earth orbit in Fig. 8 assuming a trip time of 450 days, waiting time at Mars of 50 days, and electrical propulsion specific weight of 20 lb/kw_j. Various methods of escape from initial Earth orbit are shown; a nuclear rocket used for Earth escape is seen to be better than chemical or ion propulsion used for Earth escape. The effect of increased specific weight for electrical propulsion is shown in Fig. 9, where the bottom of each band is nuclear



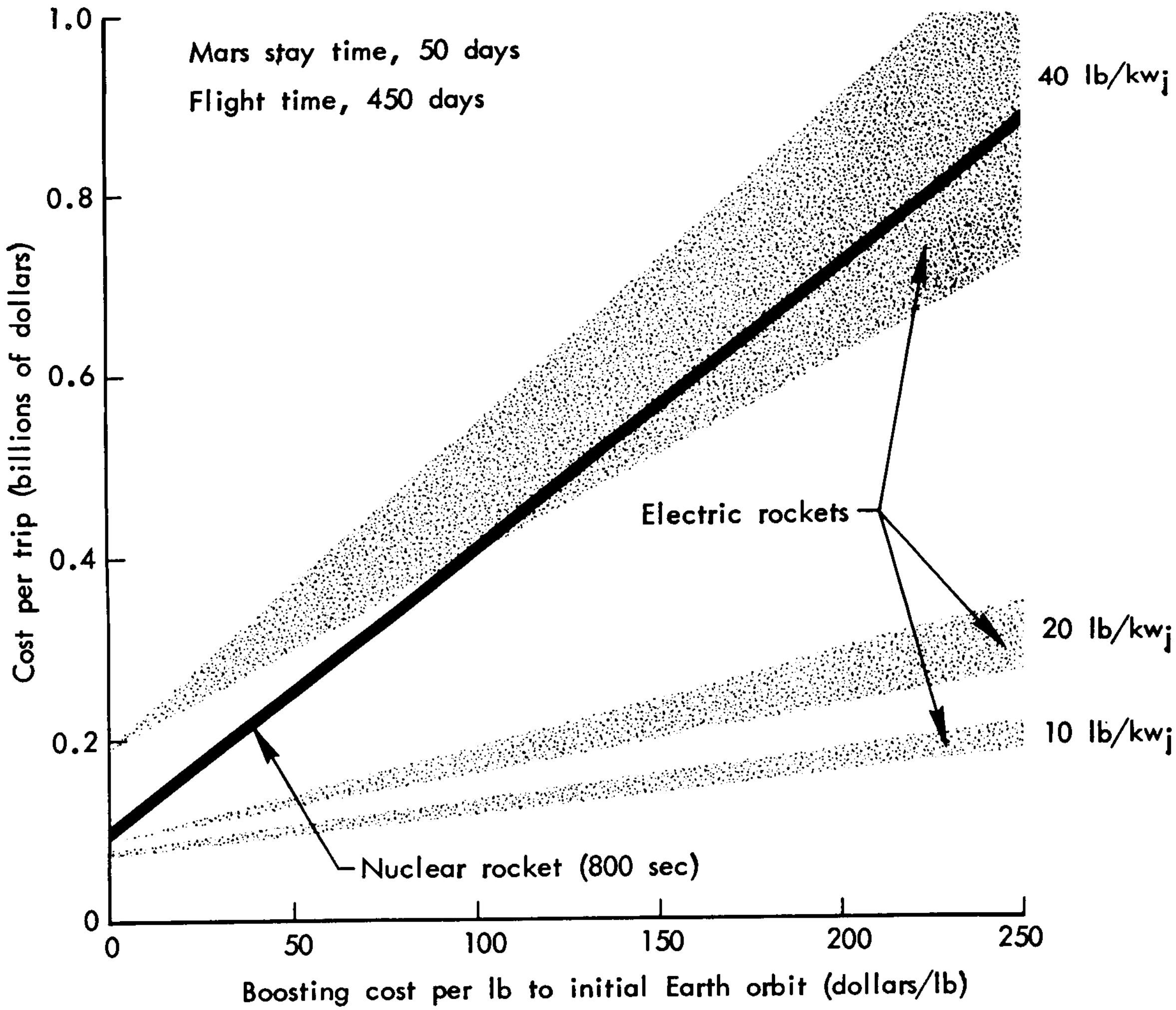
COMPARISON OF MASS IN INITIAL EARTH ORBIT FOR ALL-NUCLEAR ROCKET AND FOR ELECTRIC PROPULSION ASSISTED BY NUCLEAR ROCKET FOR EARTH ESCAPE

Fig. 7



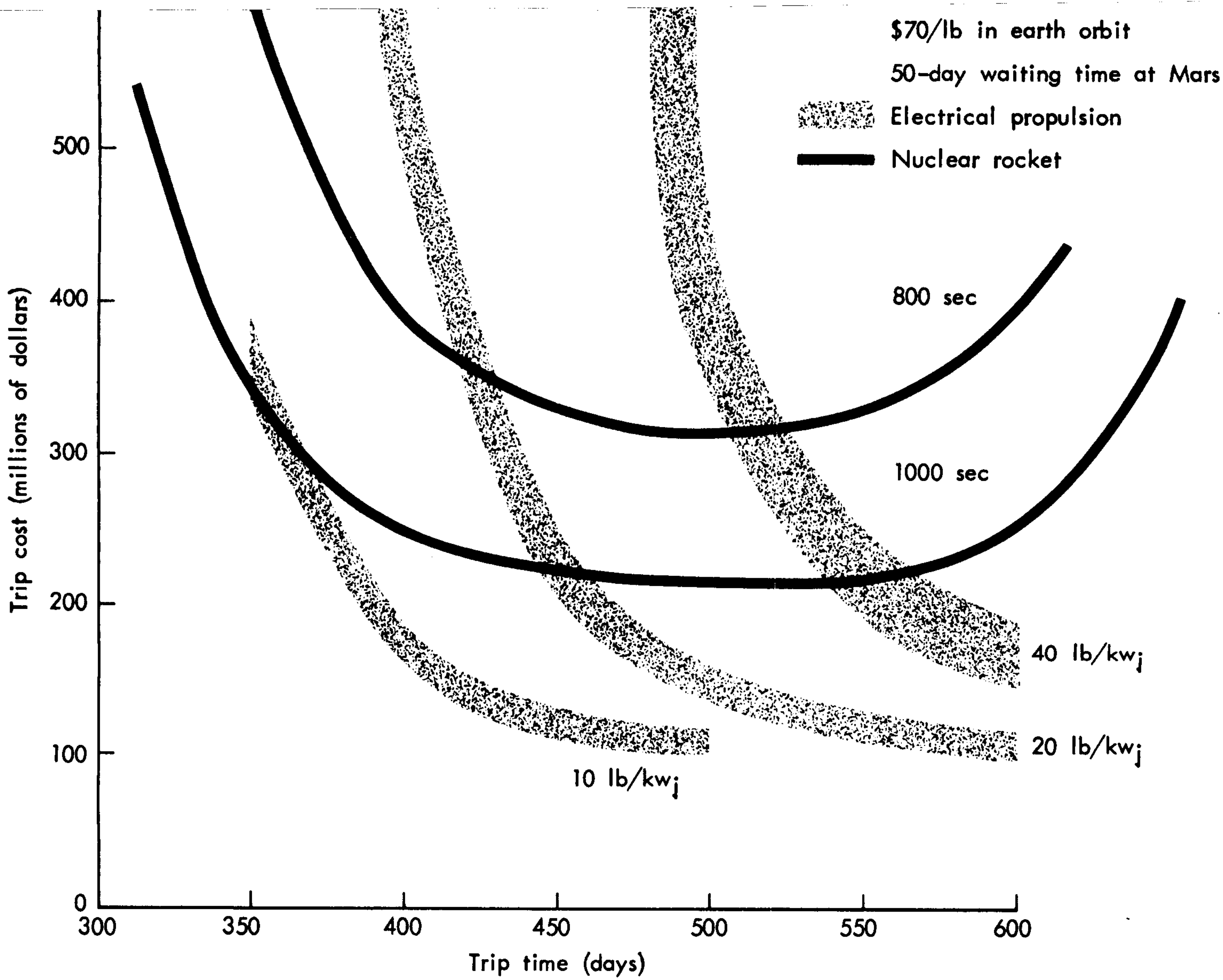
TRIP COST OF VARIOUS ELECTRIC PROPULSION SYSTEMS VERSUS BOOSTING COST
(EXCLUDES R & D COSTS)

Fig. 8



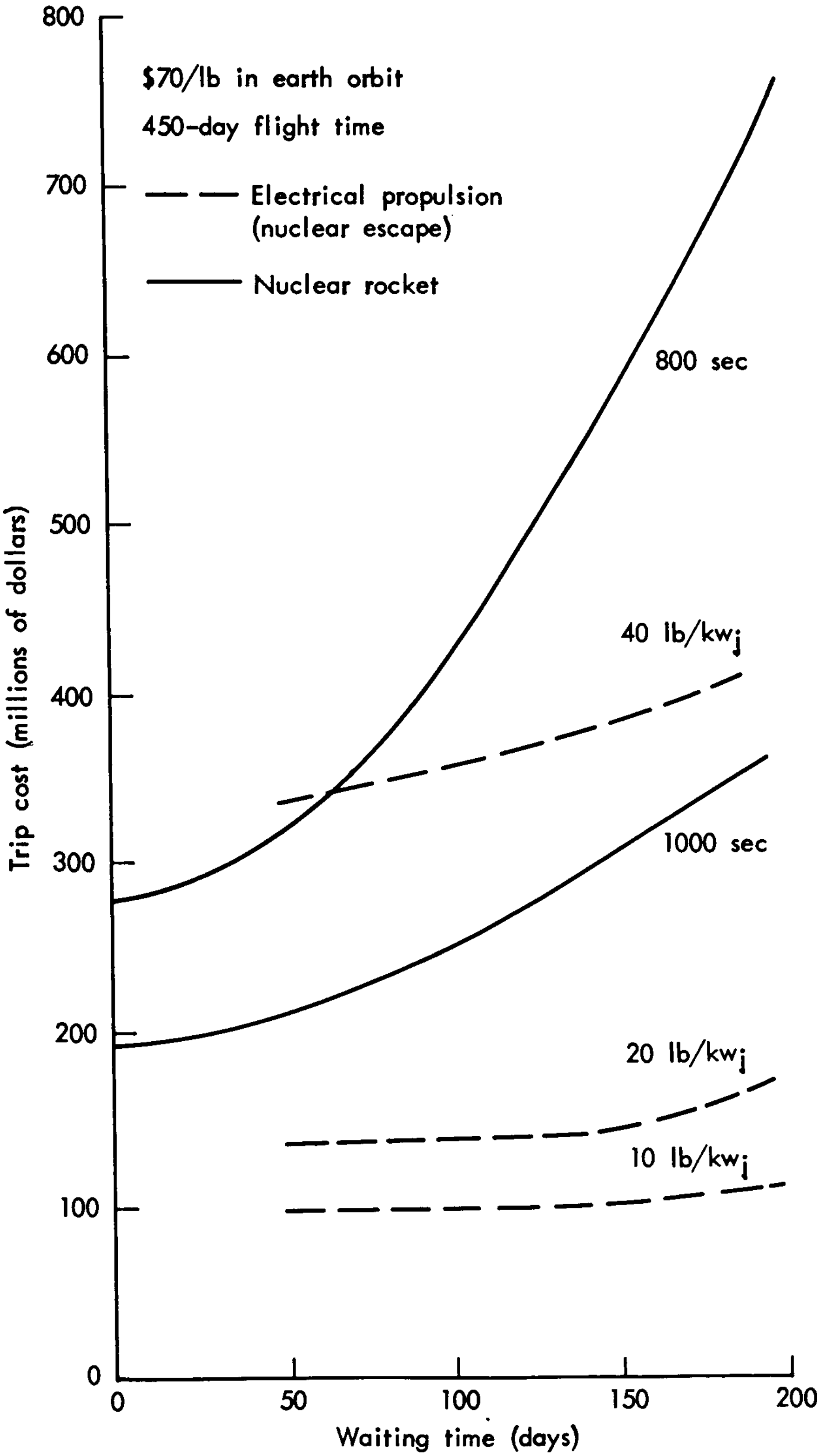
TRIP COST OF ELECTRIC PROPULSION AND NUCLEAR ROCKET VERSUS BOOSTING COST

Fig. 9



EFFECT OF TRIP TIME ON COST

Fig. 10



EFFECT OF WAITING TIME AT MARS ON COST

Fig. 11

rocket Earth orbit escape and the top is ion propulsion Earth orbit escape. Trip costs for a nuclear rocket are also shown in Fig. 9. Electrical propulsion has a cost advantage over the nuclear rocket for 20 lb/kw_j and any boosting cost to Earth orbit. If, however, the specific weight degrades to 40 lb/kw_j, then electrical propulsion has a cost advantage only if the boosting cost is greater than $100/lb in Earth orbit.

The trip cost is given as a function of the trip time in Fig. 10, assuming a 50-day Mars waiting time and $70/lb boosting cost to Earth orbit. In the previous two figures, a total trip time of 500 days was used (450-day flight time and 50-day waiting time); for the case of the nuclear rocket with 800 sec or 1000 sec specific impulse, this 500-day trip time is optimum. Several different specific weights are shown for electrical propulsion, where the top of each band represents chemical escape from Earth orbit and the bottom represents nuclear rocket escape. Electrical propulsion shows a very different trend than the nuclear rocket, for as the trip time is increased the cost for electrical propulsion continues to drop until it reaches some limiting value which is determined by the cost of the payload.

The Mars trip cost for an 800-sec nuclear rocket on an optimum 500-day trip is about $280 million. An electrical propulsion device on a 450-day trip, which corresponds to a 10,000-hour powerplant lifetime, will have a trip cost of $200 million if the propulsion device weighs 20 lb/kw_j and a trip cost of $1300 if the weight is 40 lb/kw_j. If the lifetime is increased to 11,300 hours, corresponding to a 520-day trip time, then a 40 lb/kw_j electrical propulsion device will be competitive with a nuclear rocket. Longer endurance than 11,300 hours would allow further increase in trip time giving mission costs for the electrical propulsion systems lower than for the nuclear system even for $\alpha_j = 40\ lb/kw_j$.

The foregoing remarks on endurance are based on the employment of the continuously variable thrust system which gives the minimum characteristic power and hence minimum vehicle weight in initial Earth orbit. As pointed out by Melbourne of JPL, the substitution of a constant thrust (constant specific impulse) electrical propulsion system in place of the

continuously variable system would result in a 15 per cent increase in the required characteristic power and a 5 per cent increase in vehicle weight in initial Earth orbit. Several studies have indicated that this constant thrust system should be programmed to operate power-on for about 2/3 of the flight time and power-off for 1/3 of the flight time for optimum performance. Thus, a 600-day flight time for this system would represent 400 days of system operation which is the current goal for the development program. Figure 10 indicates that for a 600-day flight time the electrical propulsion system has a large cost advantage over the nuclear rocket systems even for the specific weight of 40 lb/kw_j.

However, if 10,000-hour operation endurance is not achieved, for example, if only 5000 hours is obtained, then the cost advantage of the electrical propulsion system disappears.

Trip cost as a function of the waiting time at Mars is given in Fig. 11 assuming a 450-day flight time and $70/lb boosting cost to Earth orbit. Electrical propulsion is not very sensitive to the waiting time in contrast to the nuclear rocket.

RESEARCH AND DEVELOPMENT COST

In considering the economic feasibility of a Mars trip or any mission using electrical propulsion, we must determine not only the operational or recurring cost, but more importantly the research and development cost.

The amount the government has already funded for electrical propulsion and associated power sources is of interest and is shown in Fig. 12. The cumulative funding through FY64 is estimated at $650 million, distributed one-quarter for electrical propulsion and three-quarters for nuclear reactor SNAP systems. This is a substantial sum, and the various programs are still in the early development stages.

For purposes of estimating the R&D costs which may be illustrative of the magnitude to be expected if and when an electrical system for space propulsion is developed, a 2 Mw_e turboelectric system with an ion accelerator was selected as illustrated in Fig. 13. Except that a power level of 2 Mw_e was used, the powerplant is similar in many respects to the SNAP-50. Four power conversion systems are connected to each reactor system; the power conditioners increase the generator voltage for operating the ion thrusters and provide rectification to D.C. The power conditioners and thrusters are pooled in order to improve the reliability of the system.

A propulsion system such as this may prove to be very difficult to develop because the state of the art is being pushed. In order to obtain a low specific weight, the temperatures and efficiencies must be kept high; since the powerplant operates continuously, a long life of the order of 10,000 hours is desired.

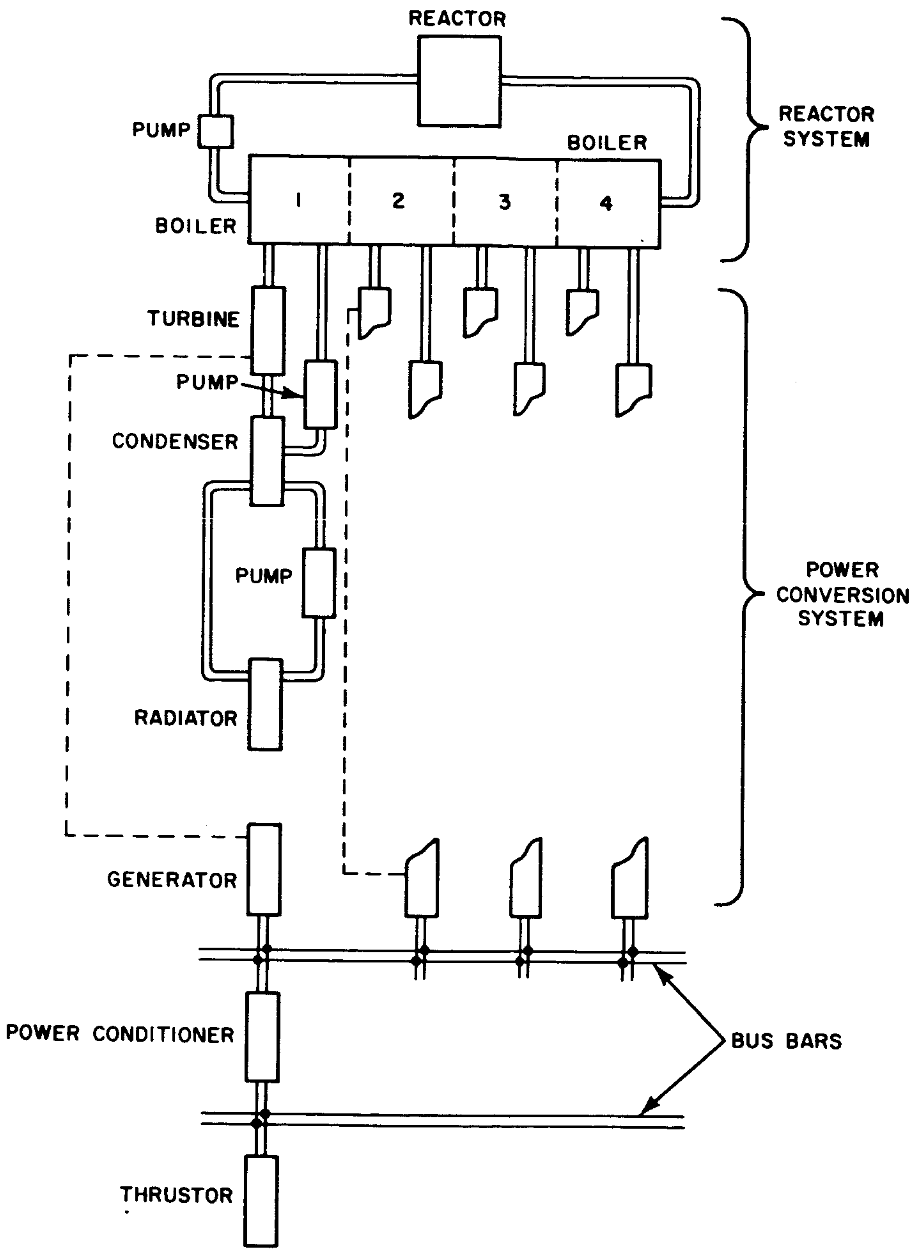
Estimating the R&D cost of such an advanced system is obviously very uncertain. As a basis for the estimate, experience with ballistic missile programs and available data on the SNAP programs were used. Three principal categories of cost were used: design and development, flight test, and system management and technical direction.

A development program was worked out in some detail, and Fig. 14 shows this program conceptually. Testing on components and subsystems was arbitrarily cut off after 10 years. Each component was tested on

ITEM	CUM. THROUGH FY 1964
NASA	
ELECTRICAL PROPULSION	104
NUCLEAR ELECTRIC POWER	89
	193
AEC	
SNAP - 10A	41
SNAP - 2	59
SNAP - 8	30
SNAP - 50	8
ADVANCED SPACE POWER	87
	225
AIR FORCE	
SNAP ORBITAL TESTS	36
ELECTRICAL PROPULSION SPACE TESTS	9
SPUR	8
SPUD	6
	59
TOTAL GOVERNMENT CONTRACTS	477
IN-HOUSE EFFORT	175
GRAND TOTAL	652

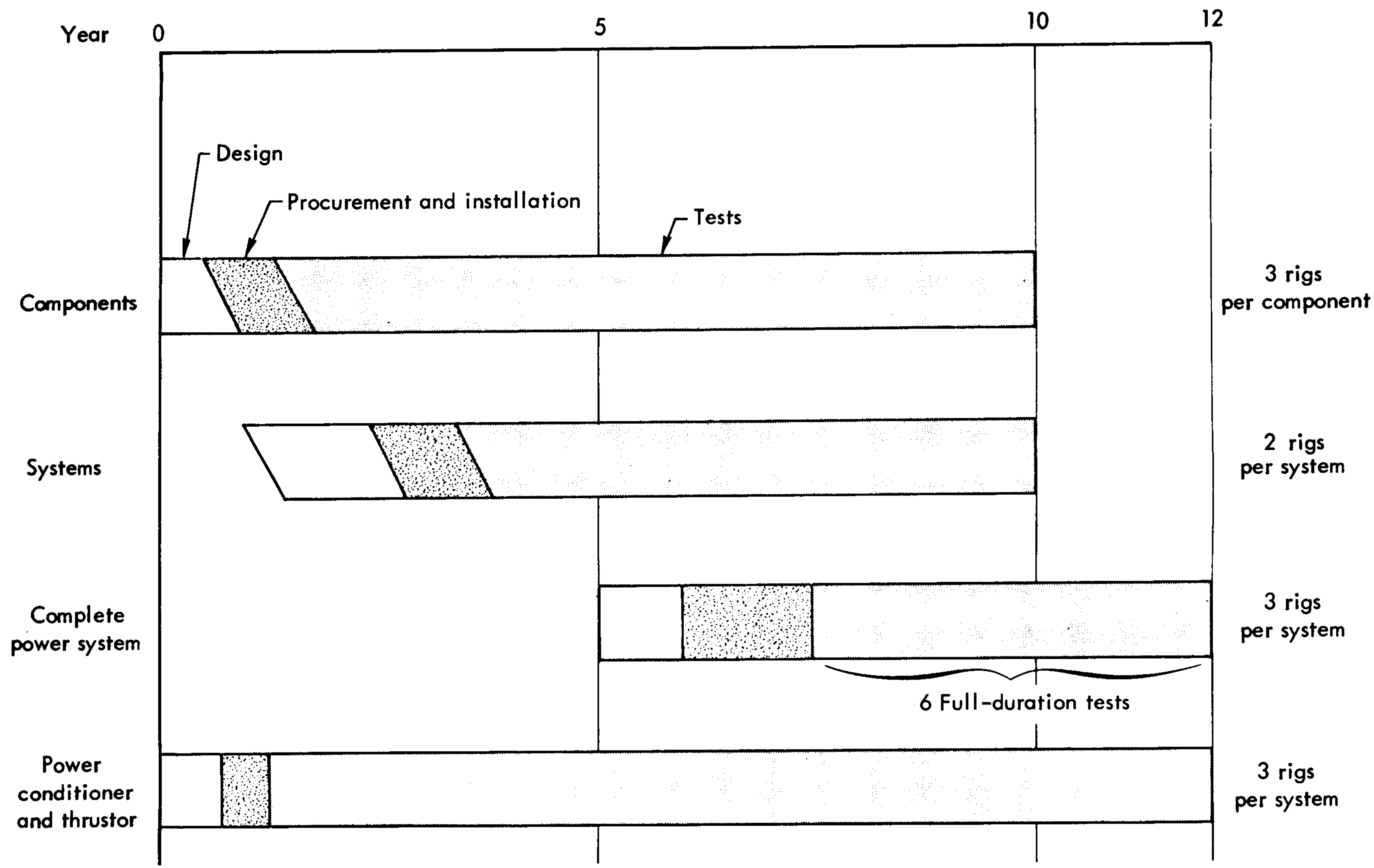
ESTIMATED FUNDING

Fiq. 12



ILLUSTRATIVE POWER AND THRUST SYSTEM

Fig. 13



CONCEPTUAL DEVELOPMENT PROGRAM

Fig 14

3 rigs and each subsystem on 2 rigs. Complete system tests were begun in the 7th year and a total of 6 full-duration (10,000-hr) ground tests were run. The entire ground test program takes 12 years. The power conditioners and thrusters are designed and tested separately because they are loosely tied to the rest of the system (i.e., through electrical circuitry). They are introduced into the complete system test in the 6 full-duration runs. These six runs represent a test of six reactors and 24 each of the converters, conditioners and thrusters.

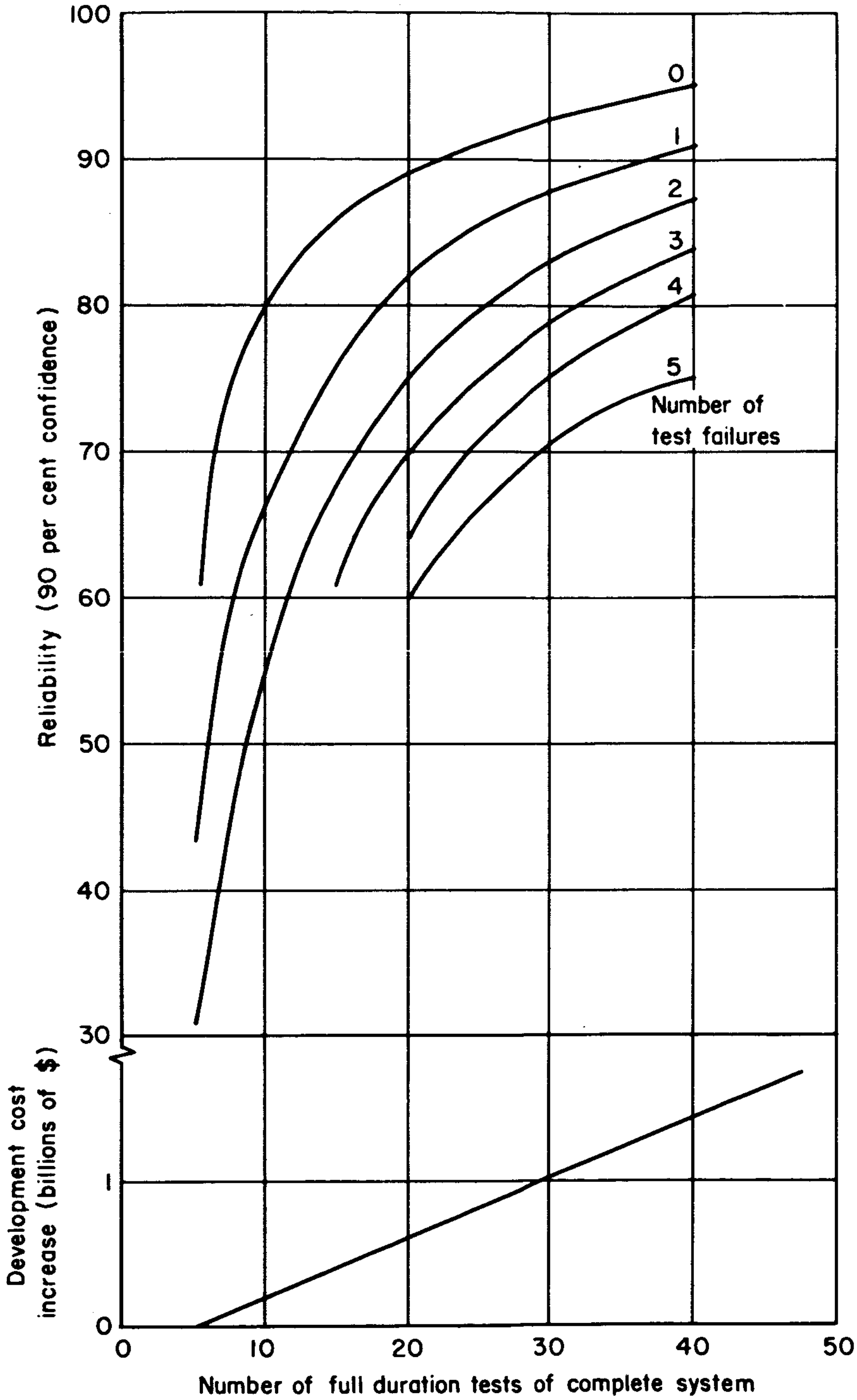
The design and development costs are given in Fig. 15 and total \$2.3 billion. The development engineering cost of \$300 million corresponds to an average of 1500 engineers working for 10 years. The largest cost component is development hardware at \$1140 million. The cost of fabricating one complete system during the development program was estimated at \$30 million; thus, the development hardware cost corresponds to approximately 35 equivalent complete systems used during the test program. The hardware cost was actually estimated in greater detail, but the number of equivalent systems is a convenient measure of the extensiveness of ground testing. Ground test operations are another major cost and include the costs of engineers and technicians directly associated with the tests.

After 12 years time and \$2.3 billion, what reliability could be expected for the system? The failure modes and distribution of failures are unknown for an electrical propulsion system, and therefore the binominal model which depends only on the number of successes and failures during the test program was used. It is well to caution that the reliability may be underestimated using the binominal model and that small samples are involved. Figure 16 shows the reliability at 90 per cent confidence as a function of the number of full-duration tests. For the development program that was costed, there were 6 full-duration tests which indicate a 68 per cent reliability for the complete system with 90 per cent confidence if all 6 tests are successful; if one test fails, then the reliability estimate is 50 per cent. If the number of full-duration tests is increased to 20, then the development cost will increase by \$600 million and the reliability improves substantially-- 89 per cent for no failures and 82 per cent for one failure out of 20

● DESIGN AND DEVELOPMENT	COST ($ x 10^6)
DEVELOPMENT RESEARCH AND DESIGN STUDIES	98
DEVELOPMENT ENGINEERING	300
DEVELOPMENT HARDWARE	1137
INSTRUMENTATION, SPECIAL TOOLING AND TEST EQUIPMENT	114
PROPELLANT (CESIUM)	46
INDUSTRIAL FACILITIES	215
GROUND TEST OPERATION	434
TOTAL DESIGN AND DEVELOPMENT	2344

DESIGN AND DEVELOPMENT

Fig. 15



DEVELOPMENT COST INCREASE AND SYSTEM RELIABILITY AS FUNCTION OF NUMBER OF FULL DURATION TESTS OF THE COMPLETE SYSTEM

Fig. 16

tests. The reliability estimate may also be improved if partial system tests are included. It should be noted that the reliability estimate for the power conversion system may be substantially higher than for the complete system since there are four power conversion systems in each complete system. Thus, if there are no failures in 6 full-duration tests of the complete system, the reliability of the power conversion system is 91 per cent versus 68 per cent for the complete system. (All quoted reliability estimates are at 90 per cent confidence.)

The system flight test costs are given in Fig. 17 and total $740 million. It was assumed that 20 single unit, 2 Mw_e systems were flight tested. The largest cost component is for the electrical propulsion system hardware--$280 million. The cost of launching the test systems into orbit is $240 million.

The research and development costs are summarized in Fig. 18. Design and development is $2.3 billion; the system flight test is $0.7 billion; and system management and technical direction is $0.1 billion (a function similar to that provided by STL for ballistic missile development). This gives a total cost of $3.2 billion. Considering the technological problems and all of the assumptions made, one naturally has to be suspicious of this figure. Therefore, to indicate the uncertainties in the cost estimate, several sensitivity analyses were performed.

In Fig. 19, the R&D cost is given as a function of two parameters, the number of engineering hours and the cost of fabricating an electrical propulsion system during the ground test program. The R&D cost is considerably more sensitive to the fabrication cost than to the development engineering cost. The two points marked in Fig. 19 represent the cost estimates in Phases I and II of the study. If the Phase II estimate is used as a base and the fabrication cost varied plus or minus $10 million and the engineering hours varied plus or minus 10 million, then the R&D cost will range between $2 billion and $4 billion.

The effect of varying the power level of the powerplant is shown in Fig. 20. During the study, a 2 Mw_e power level was assumed which would require that 5 engines be clustered for a typical Mars mission. If an engine with a 5 Mw_e power level were developed then the R&D cost

	COST ($\$ \times 10^6$)
● FLIGHT TEST HARDWARE	
ELECTRIC PROPULSION SYSTEM	280
ON-BOARD INSTRUMENTATION, TELEMETRY, GUIDANCE AND INTEGRATING STRUCTURE	11
PROPELLANT TANKS	5
● SPARES	88
● LAUNCH OPERATIONS	
ELECTRIC PROPULSION SYSTEM OPERATIONS	20
LAUNCH VEHICLE AND OPERATIONS	239
● LAUNCH FACILITIES AND GSE	20
● PROPELLANT (CESIUM)	29
● MISCELLANEOUS	48
	740

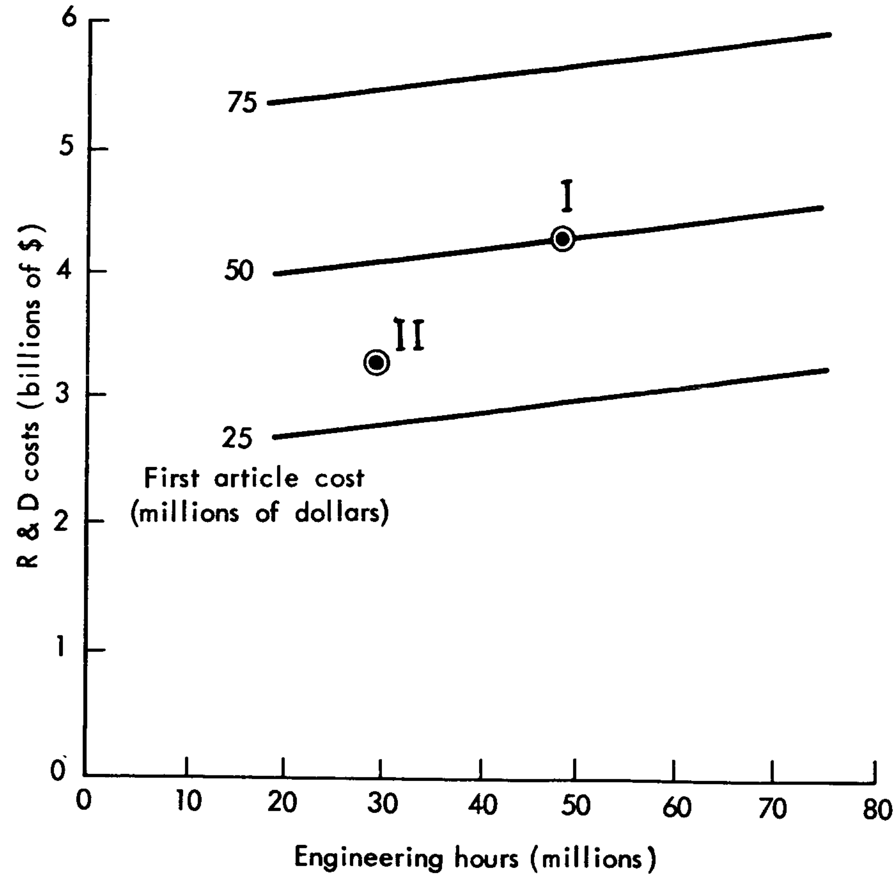
SYSTEM FLIGHT TEST COST ESTIMATE 20-2 Mw_e FLIGHT TEST UNITS

Fig. 17

		COST ($ x 10^6)
I	DESIGN AND DEVELOPMENT	2344
II	SYSTEM TEST	740
III	SYSTEM MANAGEMENT AND TECHNICAL DIRECTION	117
	TOTAL	3201

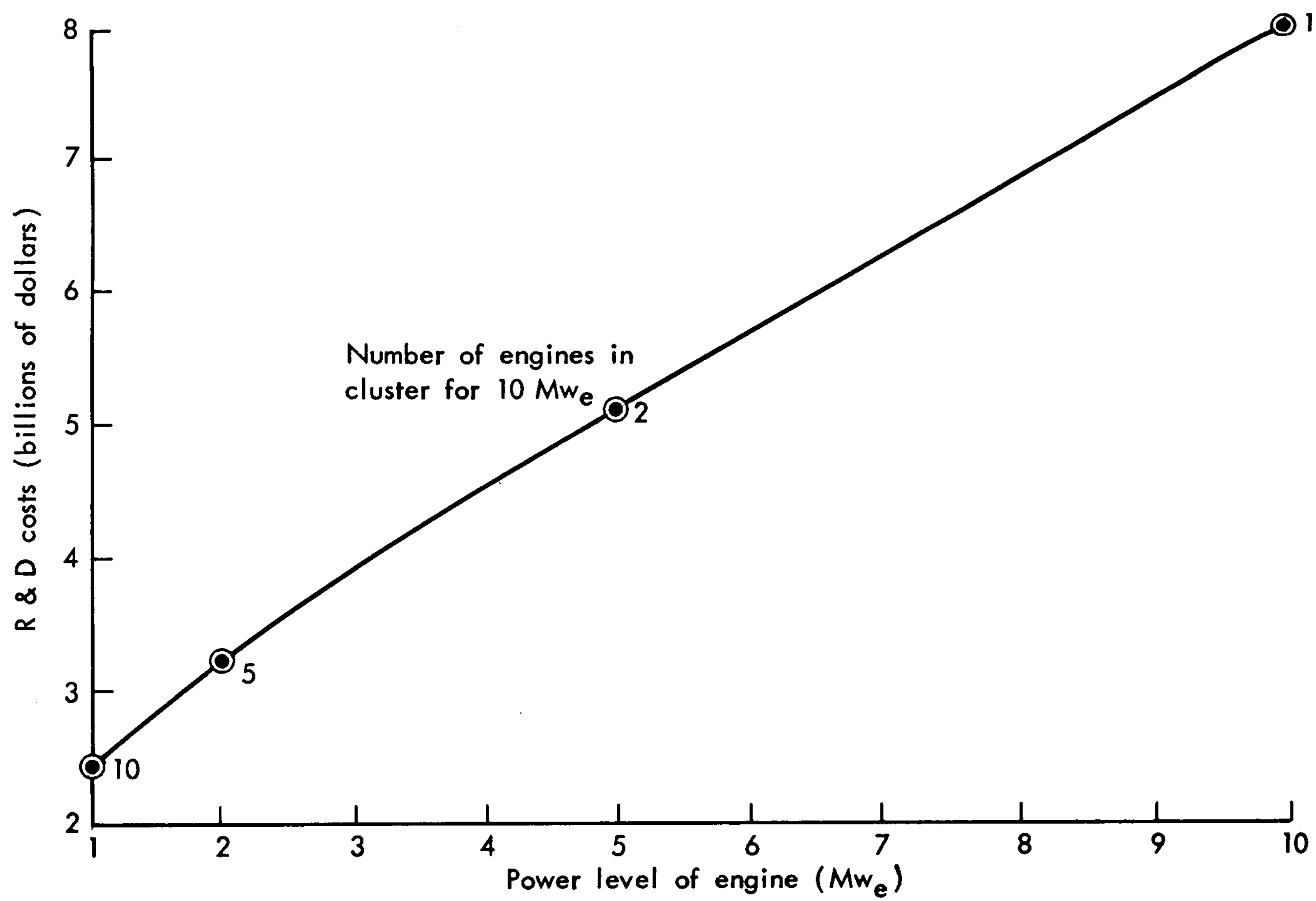
SUMMARY OF R & D COSTS

Fig. 18



COST SENSITIVITY TO HARDWARE COST
AND ENGINEERING HOURS

Fig. 19



SENSITIVITY OF R & D COST TO POWER LEVEL OF ENGINE
Fig. 20

would be over $5 billion. The reason for the rapid increase in cost with power level is the assumption that the number of test articles remains constant while the fabrication cost increases. The implication of Fig. 20 is, then, an engine of low power level should be developed and then clustered for the desired total power level. There are limitations, however, for as the power level is decreased beyond a certain range, the specific weight will increase substantially as is the experience with low power level SNAP devices.

RELIABILITY CONSIDERATIONS

It was pointed out that a development program, costing roughly $3 billion and running about 12 years, would provide an engine with only a moderate demonstrated reliability. For example, there are only sufficient full duration runs on each of the major systems like power converters, conditioners and thrusters to indicate a reliability of 90 per cent with a 90 per cent confidence even under the most favorable circumstances--namely, no failures. The reactor system having fewer full-duration runs will have a lower demonstrated reliability. The implications of these results with regard to mission reliability will be indicated and some of the avenues open to achieving high-mission reliability with these component reliabilities will be discussed. Rather than attempt to indicate optimum designs or operations, we will discuss concepts by making use of illustrative cases.

We will examine the reliability of one of the missions that was discussed previously--that is, the 500-day round trip exploration of Mars with a crew of 6 men. Figure 13 shows a diagram of a propulsion engine consisting of one reactor, four conditioners, four converters, and four thrusters. Assume that at the time the vehicle starts its heliocentric return flight from Mars, it has on board four such engines with a total of four reactors and 16 of each of the other components. The 16 thrusters are tied together by common bus bars as are also the 16 converters and the 16 conditioners, to minimize the degradation in jet power from the failure of any one of these components. For the

purposes of this illustrative case, the assumption will be made that each of these devices--each reactor, converter, thruster, and power conditioner--has a reliability of 90 per cent. These are optimistic values, particularly for the reactor, given the development program previously described. We will discuss first the effect of failures taking place at the start of the heliocentric flight back from Mars--a very unfavorable place for failure--and then will indicate the effect of failure at other points in the trajectory.

Figure 21, Curve I, shows the probability of completing the mission in the specified time or less, subject to the foregoing assumptions. To complete the mission in the design total trip time of 500 days requires that all of the components mentioned in the previous paragraph function; the probability that all of that equipment would work is practically zero, i.e., .004. If it is assumed that a 25 per cent jet power failure occurs at the start of the return trip from Mars, then for the same date of Mars departure the return trip can be accomplished with the reduced power by reprogramming the trajectory to take a longer time; for the case under discussion, a 20-day longer return trip will be required or a total trip time of 520 days. If, on the other hand, a 50-per cent loss of jet power occurs, the trip can be accomplished with a 55-day increase in trip time for a total trip time of 555 days. The probability that sufficient components will fail to give a 50 per cent or less degradation in jet power is 0.97, so we have a probability that we can return in 555 days or less of 0.97.

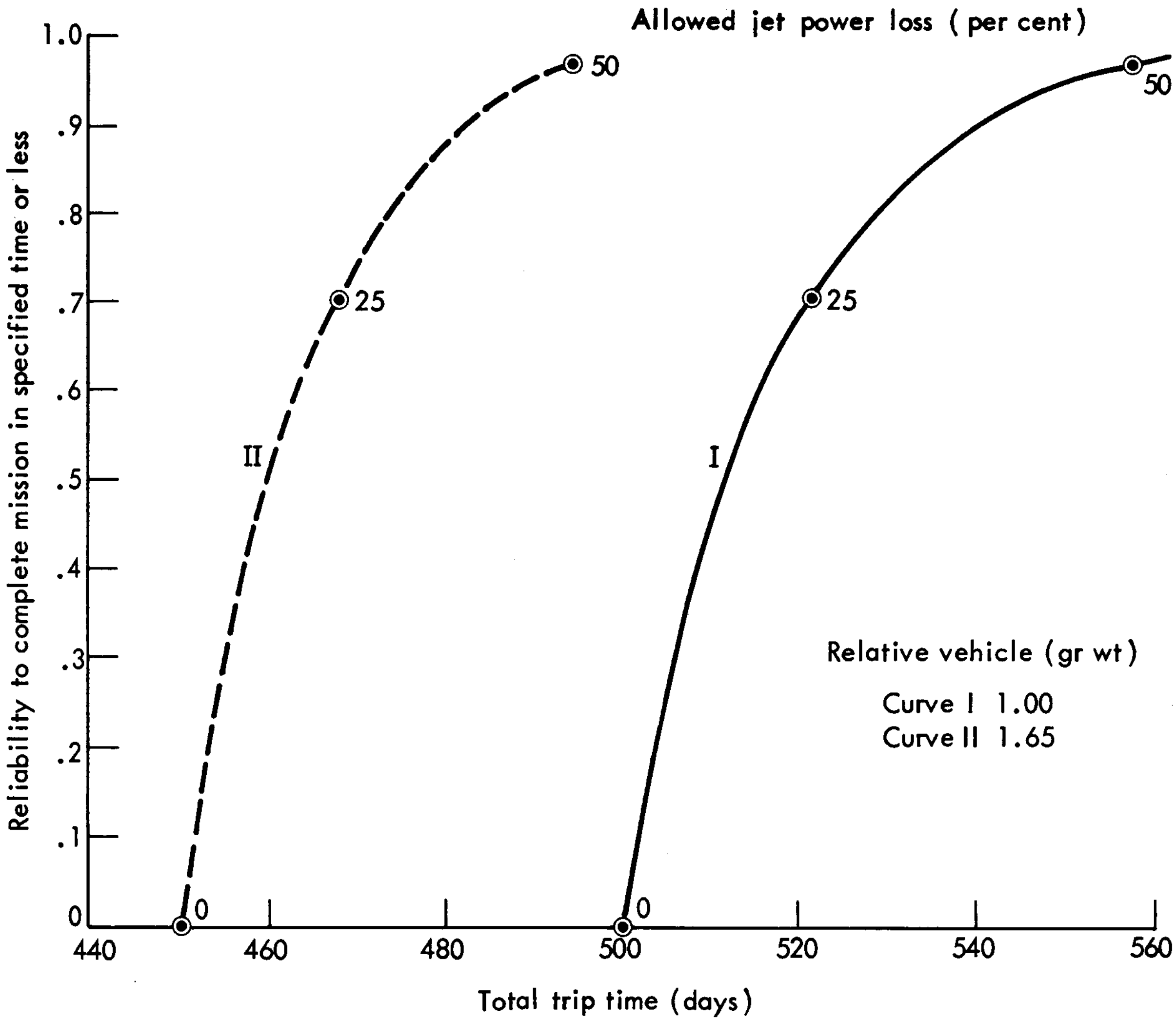
Suppose a trip of 500 days with high reliability is desired. Consider a vehicle that was designed to do the trip in 450 days, provided all of the equipment works. This case is represented by Curve II in Fig. 21. Again, the probability of completing the mission in the design time of 450 days, which requires that all of the propulsion system components work, is practically zero. With a 50-per cent failure in jet power at Mars departure, the trip can be reprogrammed for a 50-day longer trip time giving a total time of 500 days. Thus, we have a 500 day or less trip time with the Case II vehicle with a probability of 0.97. However, the Case II vehicle is roughly 65 per cent heavier

than the Case I vehicle, and its cost is higher by roughly the same proportion. Thus, for the case considered, the price of increasing mission reliability to 0.97 per cent with no increase in trip time is an increase in vehicle weight and cost of roughly 65 per cent.

Figure 22 shows the effect of increasing component reliability on the mission reliability for Case I. In this figure we have plotted against increased trip time rather than total trip time. To demonstrate a reliability of 98 per cent on a component would require several hundred full-duration tests.

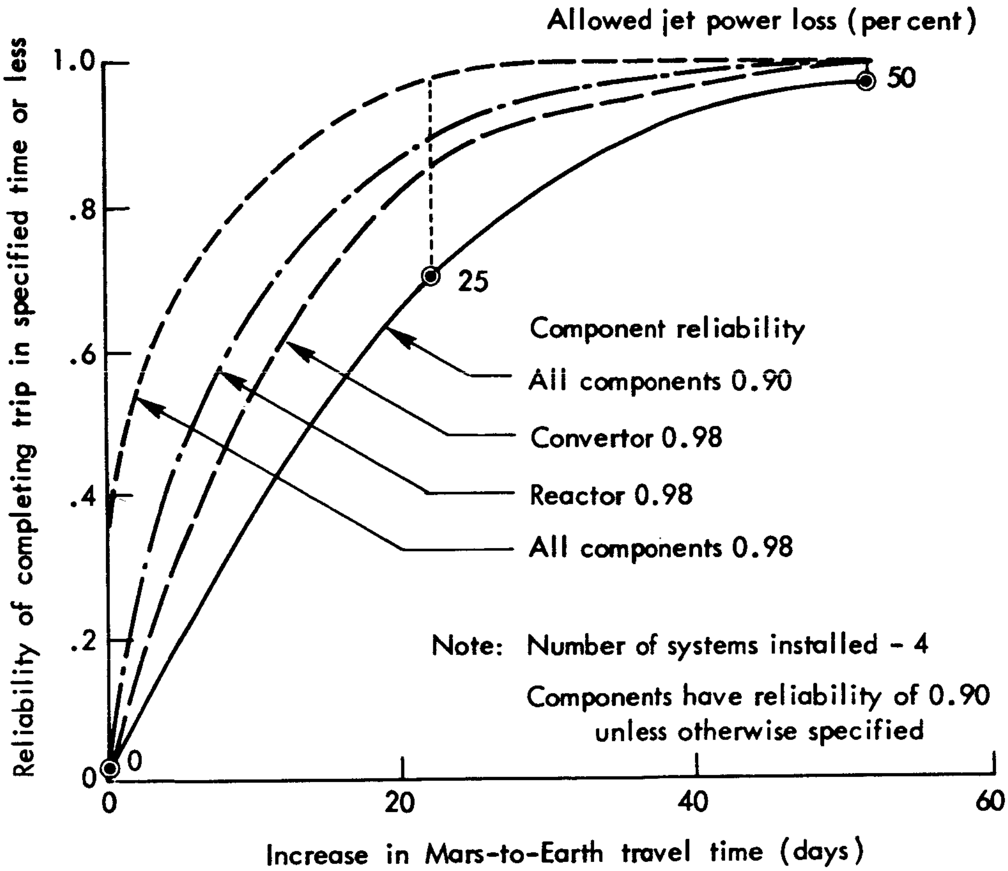
Let us consider briefly the effect of engine components failing at different times during the mission. If the components fail later than the start of the heliocentric flight back from Mars, then, of course, the penalty in increased flight time would be less than that displayed in Figs. 21 and 22. If a 23-per cent jet power failure occurs at the start of the heliocentric trip from Earth to Mars, then the trip trajectory can be reprogrammed for a 10-day longer flight time to Mars. This could then be compensated without change of leaving date from Mars by reducing the stay time at Mars by 10 days. Because of the jettison of 25 to 50 per cent of the propulsion system at Mars for reoptimization of the weight distribution, part of the failed system could possibly be eliminated. There is also the possibility of some repair at Mars. Failure of equipment later than the Earth departure date, of course, would reduce the flight time penalty. Also one has the option at the start of the trip of aborting the mission if appreciable power failure occurs.

We have used increased flight time as a means of increasing mission reliability on the assumption that the component reliability is not affected. Let us consider now the effect of operating time on component reliability. Figure 23 shows a heuristic diagram indicating the variation of expectation of survival of a group of components with operating time. Initially, there is a sharp drop in probability of survival associated with the trauma of start-up and the possibility of faulty components and installation. This is followed by a zone in which random failures may occur and the survival probability follows an exponential curve. Finally, the probability of survival drops



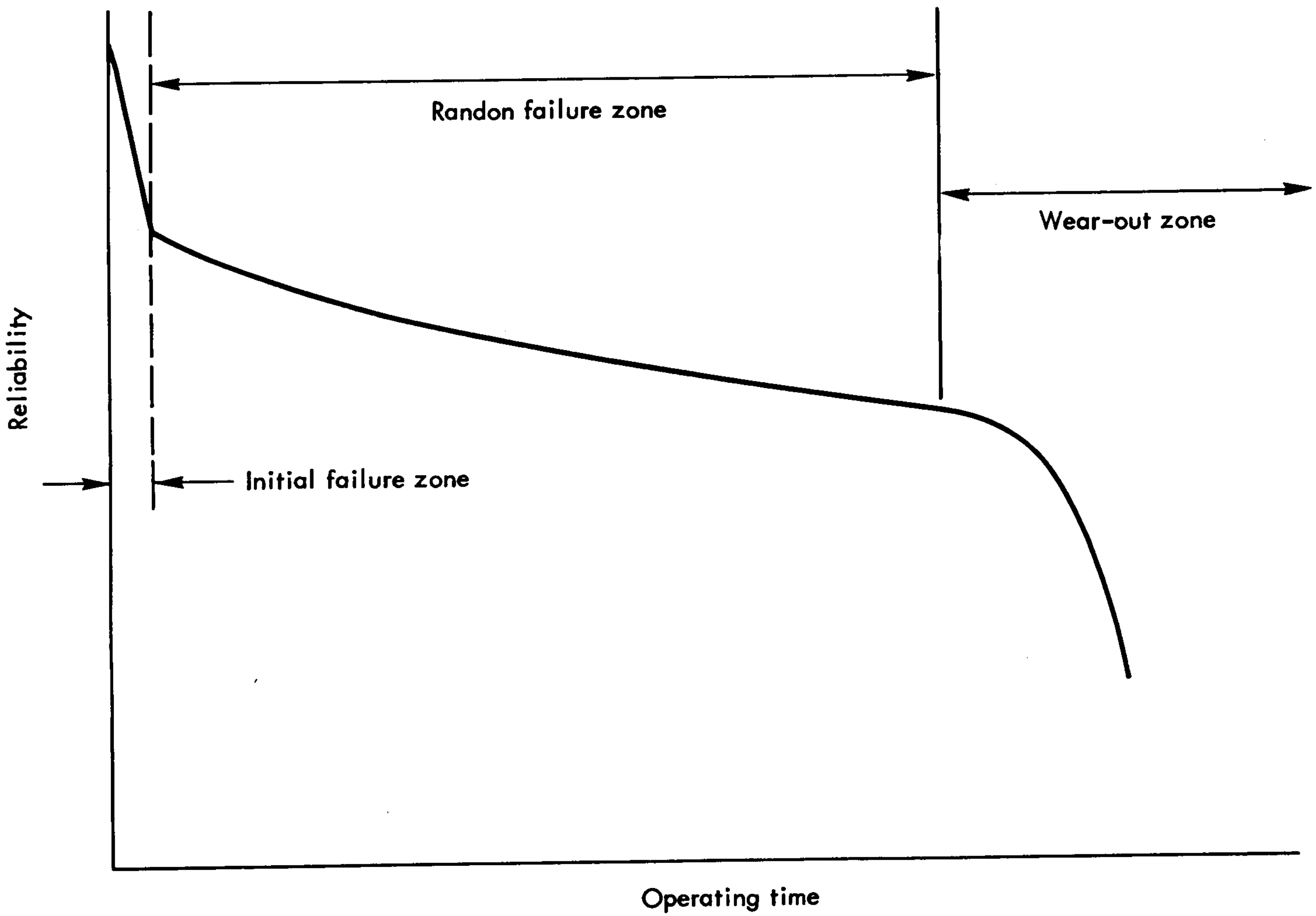
INCREASE IN MISSION RELIABILITY WITH INCREASE IN ALLOWED TRIP TIME AND VEHICLE GROSS WEIGHT, ASSUMING FAILURES OCCUR DURING RETURN TRIP

Fig. 21



ILLUSTRATING EFFECT OF INCREASING COMPONENT RELIABILITY ON MISSION RELIABILITY

Fig. 22



ILLUSTRATIVE CURVE OF VARIATION OF RELIABILITY
WITH OPERATING TIME

Fig. 23

sharply again as the operating time intrudes into the wear-out zone for the equipment. Let us assume for the moment that the flight operation time is completely in the random failure zone. Then for the examples discussed involving a 40-per cent jet power failure at Mars, the 11-per cent increase in operating time representing the 55-day extension in flight time would have a negligible effect on the component reliability and on mission reliability. However, if the addition of 55 days extends the operation time into the wear-out zone then, of course, the system is in difficulty. Thus, it is important that the development program be run to the extended operating time on the assumption that power degradation will occur during the mission.

As mentioned in the discussion of Fig. 10, by using a thrust-coast trajectory the operational endurance required of the propulsion system can be decreased to about 2/3 of the flight time with only a small increase in vehicle weight. The operational endurance achievable by the electric propulsion systems is one of the major uncertainties of the development program. If the 10,000-hour goal is missed by an appreciable amount, the cost advantage indicated for the electrical propulsion system over the nuclear rocket in the Mars mission disappears.

The influence of reliability considerations for the nuclear rocket case on the cost comparison between nuclear rocket and electric propulsion system has also been considered. The better utilization of development program data for reliability analysis and the implications derived from reliability studies relative to development program planning are also being studied. These topics will not be discussed here.

ALTERNATIVE POWER CYCLES

The several alternative power systems listed in Fig. 24 will now be discussed briefly. Most of the research and development emphasis is currently directed toward the Rankine cycle. A number of studies have been made by various organizations of the Brayton cycle; and, in general, there is agreement that in order for the Brayton system to provide the same specific weight as the Rankine cycle it must operate

1 RANKINE CYCLE

2 BRAYTON CYCLE

3 THERMIONIC CONVERSION

a. In-pile

b. Out-of-pile

POWER SYSTEM CONCEPTS
(1 Mw_e CLASS)

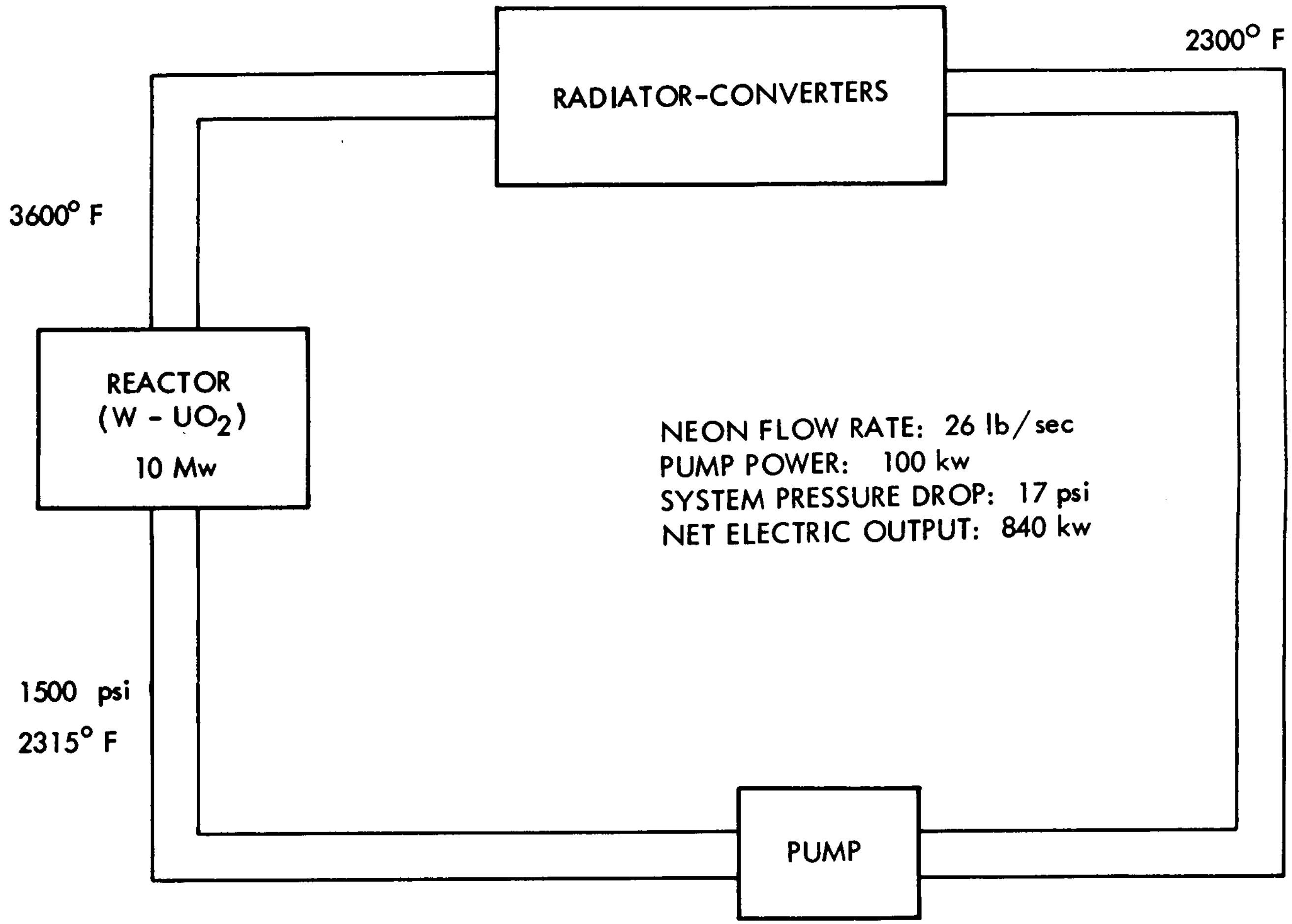
Fig. 24

at a peak cycle temperature about 500 to 1000°F higher than the Rankine cycle. The Brayton cycle can employ an inert gas (e.g. neon or helium, etc.), as the working fluid, and can thus make better use of the refractory metals. Thus some increase in cycle temperature over the Rankine cycle is admissible. Proponents of the Brayton cycle believe that in the long run it may win out over the Rankine cycle because it may prove to be easier to develop to the required reliability. However, it does not appear to hold any promise for a breakthrough in specific weight. This system will not be discussed any further here.

Considerable progress has been made in the development of thermionic converters which convert heat directly into electricity. Laboratory models have been run for durations approaching 10,000 hours, however, at low specific power and efficiency. Other devices have been run at the high specific powers and efficiencies desired for large space power systems; however, they have not yet achieved the required endurance. These devices are in their infancy and the progress is encouraging.

Design studies made by several organizations indicate that, when the thermionic converter is designed to function as both a reactor fuel element and as the device for converting the heat developed into electricity, a power generator specific weight of less than 13 lb/kw_e can be computed for the in-pile system. However, the development of this system may prove to be very difficult because the thermionic converter must meet the exacting requirements of the reactor fuel element and of its own function as a thermionic device. Test data are coming very slowly on this type of converter, principally because testing in a reactor is a very slow and costly process. A great deal more progress has been made in development on the thermionic devices that would be suitable for out-of-pile installations. We were, therefore, interested in exploring the potential of an out-of-pile thermionic device system. If such a system has an attractive specific weight, then it provides additional support to the idea that thermionic systems warrant further consideration.

Figure 25 shows a diagram of the system investigated. It is a very simple system which is to its advantage. The reactor, containing tungsten-uranium oxide fuel elements, is cooled by an inert gas, e.g., neon. The



NUCLEAR - THERMIONIC SCHEMATIC

Fig. 25

reactor heat carried by the neon at a temperature of 3600°F is transformed in part to electricity in the thermionic converter bank. The neon is returned to the reactor at a temperature of 2300°F by means of a molybdenum pump.

Figure 26 shows the estimated weight breakdown of this system. With 10 Mw of heat generated in the reactor, an electrical power output of 840 kw and a specific weight of 12.5 lb/kw_e are computed. Helium was also studied and indicated even better performance.

An investigation was made of the effect of compounding a Brayton cycle with a thermionic converter system as indicated in Fig. 27. The results are shown in Fig. 28. The addition of the Brayton cycle did improve the efficiency of the system substantially, e.g. from 8.4 to 28 per cent, but the over-all specific weight was higher than for the simple thermionic system because of the large radiator weight required by the Brayton system. The simpler system is the more interesting also from the standpoint of reliability.

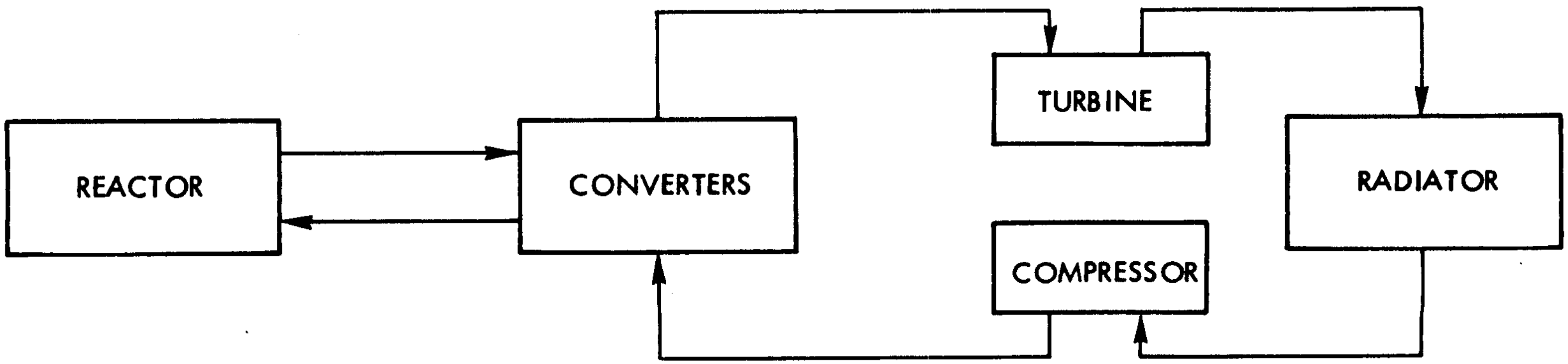
The thermionic diode performance used in the analysis just described was based on test data. Pressure drop, heat transfer, stress, and materials problems were examined, and although the analysis is not complete, there appear to be no problems yet that would resist experimentation. This analysis is comparable in sophistication to the early Rankine cycle analyses that led to 10 lb/kw_e for the power system. One can expect some increase in specific weight as more detailed design studies are made taking into account boost loads and reliability considerations. Even with the expected escallation in specific weight, the initial estimates are sufficiently low that both in-pile and out-of-pile thermionic systems may be considered competitive with the Rankine cycle. Only further experimentation will tell which of these systems is more amenable to development.

The out-of-pile version was not presented to indicate that this is the proper direction for development because it is too early yet to make this choice. It was presented rather to indicate that there is more than one attractive approach to thermionic systems to strengthen the case for these systems.

	(LB)
REACTOR (UNSHIELDED)	1,800
RADIATOR-CONVERTERS	7,200
PUMP AND MOTOR	500
STRUCTURE	1,000
TOTAL	10,500
POWER OUTPUT	~ 840 kw_e
SPECIFIC WEIGHT	~ 12.5 kw_e
POWER CONDITIONING EQUIPMENT	~ 4 - 5 kw_e
SHIELD (MANNED VEHICLE)	~ 2 - 4 kw_e

WEIGHT BREAKDOWN FOR NUCLEAR - THERMIONIC SYSTEM

Fig. 26



BRAYTON CYCLE - THERMIONIC SYSTEM

Fig. 27

	THERMIONIC SYSTEM		BRAYTON CYCLE (WITH RECUP.)
	WITHOUT BRAYTON CYCLE	WITH BRAYTON CYCLE	
REACTOR POWER, Mw	10	10	10
NET ELECTRICAL POWER, Mw	0.84	2.8	3.1
EFFICIENCY, %	8.4	28	31
RADIATOR AREA, FT^2	-	19,000	32,000
SPECIFIC WEIGHT, lb/kw_e	12.5	14.5	18

BRAYTON CYCLE - THERMIONIC SYSTEM PERFORMANCE

TURBINE INLET TEMPERATURE - 2300° F

Fig. 28

CONCLUSIONS

On the basis of the foregoing discussion, the following tentative conclusions are drawn.

- A $3 billion, 12-year development program on an electrical propulsion system would provide a system of only moderate proven reliability.

- High mission reliability can be obtained by designing for the contingency of a substantial degradation in jet-power. In low-acceleration trajectories, completion of a mission, in the event of loss of power, can be accomplished by increasing travel time and/or by redundancy in propulsion system equipment.

- In the Mars mission with a longer allowed round-trip time, (e.g., roughly 100 days) for the electrically propelled vehicle relative to the optimum trip time for the nuclear rocket propelled vehicle, the electrical system promises a substantially lower trip cost than the nuclear rocket. This savings would pay for the cost of developing the electrical propulsion system in roughly 20 manned Mars trips.

- Because of uncertainty in attainable system endurance and specific weight, the uncertainty in trip cost advantage in space missions of electric propulsion systems does not provide a strong incentive for early initiation of a full scale development effort on a large space power system.

- Current experimental programs relating to the several promising systems, e.g. Rankine, Brayton, thermionic, etc., should be allowed to mature to more completely

reveal their respective potential before a decision is made to initiate development of a space power system in the megawatt class.

PART 17

PANEL DISCUSSION

Moderated by

Dr. H. H. Koelle
George C. Marshall Space Flight Center

PANEL DISCUSSION

Moderated by Dr. H. H. Koelle, MSFC

SPEAKER: Dr. H. H. Koelle, MSFC

I would first like to introduce the panel members. First, to my right, is Mr. Thomas Dolan from the National Aeronautics and Space Council. The next four gentlemen are representing NASA Headquarters. First is Mr. Bill Fleming, representing Dr. Seamans' staff and himself. Then, we have the three program offices: Mr. Ed Gray, representing the OMSF; Mr. Dick Wisniewski, representing OART; and Mr. Don Hearth representing OSSA - that is, Dr. Homer Newell. Then we have the Center representatives: Mr. Max Faget from Houston and Dr. von Braun from Marshall. Unfortunately, Dr. Debus of the Kennedy Space Center was unable to attend.

In order to expedite the discussion, I have put myself into your shoes and listed here some eight questions which I would like the panel to answer, as much as they care to, and can. The following questions I feel are important: Why do we think manned planetary flight is an important role of the space program? What are some of the particular problem areas still in the mill? What has been eliminated thus far as not being very attractive? When can we expect a decision to be made on a manned planetary project? I think this is a very pertinent question. What mission philosophies are considered to be practical or impractical? Which mission objectives seem to be the best bets at the present time? How can we get a handle on the problem of value definition of individual missions? We have to have some criteria to make a choice. And, finally, what remains to be done before the alternative courses of action are defined clearly enough to make a decision? So, I would like the panel members to take about five minutes each and address themselves to answer any of these points, or any others they think are important at this time.

SPEAKER: Mr. T. Dolan, National Aeronautics and Space Council

Thank you, Dr. Koelle. It just occurred to me that I am the only outsider, so to speak, here, and this looks like a wonderful opportunity to get into serious trouble. That reminds me of that little jingle, something to the effect that - "He who in discussions would interpose, should be prepared to wipe a bloody nose." However, speaking as an outsider, before I address

myself to Dr. Koelle's questions, there is one thing that occurs to me. I think this symposium, bringing together 13 studies, I believe, into one meeting and presenting them to all of the audience, represents a very imaginative and constructive approach to the stimulation of the meaningful exchange of information in this area. I think I speak, for most of the people here, in taking this opportunity to congratulate Dr. von Braun, Dr. Koelle, Dr. Ruppe, and Mr. Smith, and the other members of the organization who set up and made this conference possible.

Now to address myself to one or two of the issues at hand. The first observation has to do with some particular problem areas. One that comes to my mind is the fact that, frankly, I get worried that some of the major space decisions come a little faster than people realize. I think that the decisions for the man in space and the man on the Moon programs occurred possibility a little faster than were conceived by some to have occurred. The space station decision is rapidly unfolding. I think that the manned interplanetary decision may come sooner than our unmanned organizations might project at this time. I believe that there is a definite relationship between the unmanned activities and the manned systems. I view the former as the precursor to the manned activities. It would seem to me to have merit if, in the study of the manned systems, and the projections of the development planning, we might give consideration to the development of the corresponding unmanned satellite program that would be an intimate and integral part of the total manned interplanetary flight program. We should, perhaps, deal with them as one subject, and treat them as one entity in the development planning part of the manned interplanetary flight studies.

Another question that we are often asked is when can we expect a decision to be made on the manned planetary project? I certainly do not know the answer to that question, and I don't think anyone does. But I have a suspicion that some of the ingredients, or the indicators of the forcing functions, or whatever you want to call it, of such a decision might take one or more of the following forms. So, if we keep our eyes peeled for these functions, I think we can get a fair clue, or an indication, as to when such a decision might be made. It might come possibly as a reaction, if the USSR lands a man on the Moon first. This is my opinion, anyway. We might then raise our sights to the manned interplanetary regime. It might come after the basic accomplishment of the Apollo mission and the organizational thrust to move on to the next regime of activity, and that can be a very vital function.

It could also come as a reaction if the USSR formally unveils a manned interplanetary flight program. I think it could come to pass if we had a true propulsion breakthrough. I am think now of something in the gaseous core concept where we would have orders-of-magnitude breakthroughs in propulsion concepts and then use that for interplanetary flight. That could be a forcing function. Another forcing function could be the threat situation, which might take the form of the space fleet concept. Those of you who are familiar with some of the work in that area know that the threat is always a forcing function in these decisions.

However, there is one point I would make, in summary, and that is that I think it is important that we recognize the political motivation of our major decisions. And, by doing the type of homework that is shown in the studies we have heard here for the last several days, we can be prepared to move forward on rather short notice when the decision is actually made. Thank you.

SPEAKER: Mr. W. A. Fleming, NASA Headquarters

I would like to second Tom Dolan's comments on the value of this three-day conference. I want to say that, in my opinion, one of the most important factors related to planning and studies is certainly part of planning - that is effective communication. There is no more effective communication than face-to-face, person-to-person communication. I think this is an excellent step. I think there is much more that we can do to get the most out of our study and planning effort so that everybody can take advantage of what the various people involved are learning.

With regard to the questions that Dr. Koelle has posed, there are two of them I would like to discuss, together, in rather general terms. One, which Tom Dolan also commented on: When might we expect a decision to move ahead on the national planetary program, and secondly, the question related very closely to it - what remains to be done before the alternate courses of action can be reviewed, and the most promising course selected to move forward.

I think we have a great deal yet to learn before we will be ready to move forward. First, we know nearly nothing about the planets that we are about to explore. I think there is much, much more that we must learn about the planets, and the environment on the planets to which the men and systems will be subjected, before we can proceed with a sound development and design program.

Secondly, there is much more we must learn about the operation of man in the environment of space for long periods of time: this includes the vacuum, the weightlessness, and the radiation environment of space. It will be through our manned programs that we will learn this. We need to learn more about the problems related to exploration of the surfaces of extraterrestrial bodies. What are some of the logistics and operating problems related to these? Certainly, propulsion is one of the key areas in manned planetary exploration. We must demonstrate the feasibility of some of the advanced propulsion systems that we are examining on paper today and on which we are conducting laboratory experiments.

What this means to me is that we must mount an intensive, unmanned planetary exploration program to obtain for us the data that we need about the planets, and the environment of the space between here and the planets. We need a program of manned operation in space for relatively long periods of time. This means operation of men in space stations to determine whether or not we must provide artificial gravity and, if so, how much. We must have the benefit of some of the operational experience of men landing on, and exploring, the surface of the Moon - a body which is two orders of magnitude closer than the planet Mars. And certainly we will have, not necessarily correspondingly easier, logistic problems, but we will, indeed, have much easier logistics problems than in the more distant explorations. And we must demonstrate the capability of our new propulsion techniques, such as nuclear rockets, electric propulsion, and nuclear electric power generation, and conversion.

And, in addition to this, there are many areas of technology that need to be further developed. I feel that then, and only then, is this nation going to be in a position to make a sound decision to proceed. The question, then, is: When will we be ready to proceed? Hopefully, in the early 1970's.

SPEAKER: Mr. E. Z. Gray, OMSF, NASA Headquarters

Dr. Koelle, I think you are to be congratulated that you were able to get these people down here for this kind of a get-together. One of the problems that we constantly run into at Headquarters is being able to assure everyone that the studies that are done are examined by the various people who have to make decisions, and that the information is being generally circulated so that some useful product results from the studies. And I am really happy to see the way this thing has worked out; this might be a forerunner of a mode of operation to proceed like this in certain areas again in the future. So I am most happy to have the activity carried forward; not only carried forward, but done so in such an efficient manner.

I think the message I might like to leave with the group is to talk about the study program in general, because it was not too long ago that I was on the other side of the fence in industry, involved in study activity. I think too often the people involved in studies lose sight of what the Agency has in mind when they are conducting these studies. Quite often, people get quite enthusiastic about a particular study they are involved in, thinking that it immediately will turn to hardware. And what is more, they get their management so enthusiastic about it that they are quite put out when they find that, in fact, that hardware is maybe a little further away than just around the corner.

Now, in particular, I think you first have to recognize that the purpose of a study program, as far as the Agency is concerned, is to gather together information for decision making purposes. Now this decision making, that we might become involved in, could many times be "no" as well as "yes" and, consequently, we might study a particular area just so we are in a good position to understand, if the answer is "no", why we said "no". And, on the other hand, in most cases, when we are studying something we are trying to find a "yes" answer that makes the most sense. Now, this kind of ties in with the question of "when might you embark on a mission." Now, the first thing you must remember is that, in the phase we are involved in, we are studying quite a few different missions, because we are trying to get information together around which we could make a decision as to which mission we may want to go forward on. So, at all times, we expect to be carrying on mission studies all of which would not possibly be funded with a realistic budget appraisal. But, we do want to have enough studies in depth in these various areas so that a decision can be made to go forward in one or more areas, and at the same time we will probably continue studying the areas that did not get started to see when they should be fitted into the pattern.

I think all of you recognize that, within a certain budget limitation, you cannot do everything. And it is our best guess right now that there is not a lot of money for new starts before FY 67, and even then it is problematic as to how much money there is going to be at that time. I do not think you should consider this from a negative standpoint and be disheartened, because I look at it quite in the opposite vein. This now gives us more time to do some realistic hard-headed planning; getting enough data together so that we can make an intelligent decision; and know whereof we speak when we say what the next program timing, costing, and performance will be.

I would also like you to understand a certain viewpoint that many of us have regarding what kind of studies we should be asking industry to do, and what kind we should be doing in-house. Now, there are certain kinds of thinking that you cannot ask someone else to do for you; or, if you do, we should not be getting these fantastic government salaries that we do. In particular, I am talking about setting up mission requirements, trying to evaluate the different approaches to arrive at a preferred approach, and determining a preliminary program plan and cost that we ourselves are willing to stand behind. Now we can ask for information to fill out the blank spaces, but in the last analysis we have to do this kind of work ourselves, because that is the job that we have been asked to do in the assignments that we have.

So, in many cases, if it sounds to you like we are asking you to do Agency thinking, I think you would be doing both yourselves and us a favor if you asked us the question, "Do you really want us to do this kind of thinking for you, or does this really belong in the kind of statement of work that you are asking industry to do for you?" Now I have had many discussions with Bill Fleming and others, in Dr. Seamans' and in Dr. Mueller's areas, and we all, I think, see eye to eye. I think that, basically, there are certain types of assignments that we must do ourselves, and unless we do a real workman-like job in these areas, the whole study program is going to be less meaningful than if we do our jobs ourselves. So, if you ever hear any criticism from us, I think you should also recognize that we are being critical of ourselves as well.

Now, as far as a planetary mission is concerned, it is my own opinion that we probably will not engage in full-scale development leading toward a time period any earlier than the early 1980's. And the reason I say this is, look at the time scale for development the items which might make such a thing feasible, then look at the budget that might be available. It seems to me we are going to be doing real well to have an effective planetary exploration capability before the early 1980's.

Now, that does not mean that we will not continue looking at what we might do in an earlier time period. We are always going to be interested in trying to understand what we could do in an earlier time period, because we need to make sure that we are not overlooking a really attractive approach because we became so preoccupied with a "magic date." There is nothing magic about any of these dates but as far as trying to have some sort of a plan, that is about the kind of a plan that the Office of Manned Space Flight is talking about.

There is one other aspect I would like to mention, and that is, even with a 1980 time period, this does not mean we have lots of time around which to do our planning. This is because, when you recognize the long development items and recognize that if you want a great deal of confidence of being able to accomplish a mission successfully, we should have had long periods of experience in space; probably Earth orbit, where we are trying out not only equipment, but the procedures. When you put all these items end-to-end it comes out that some of the next decisions we make - whether it be for a space station, extending Apollo, or whatever it is - that you have to keep in mind planetary exploration as part of the requirement around which you would be developing these long exposure equipment and procedures, to make sure that you are laying the ground work so you can do the job in 1980.

Therefore, we have to keep constantly reminding ourselves of that, because it tends to get lost in the shuffle unless you write on the wall that this is one of our goals, and that any decision we make has to recognize that planetary exploration is one of our goals, and so, I would like you to know that we are doing so. That is all I have. Thank you for coming.

SPEAKER: Mr. R. Wisniewski, OART, NASA Headquarters

From an OART point of view, we, of course, have always endorsed conferences of this type, and we find them most useful in solving the communication problem we all seem to have in the advanced study area. Of course, we use our studies to define problem areas needing advanced technical development, and possible to identify new concepts or lead to the invention of new concepts.

With regard to some of the questions that Dr. Koelle has posed, for example: "What has been eliminated so far as not being very attractive?" "What mission philosophies have been considered to be practical?" "What mission objectives or mission profiles seem best bets at this time?". I think that the conference pretty well illustrates that we are not smart enough to answer questions like this at this time.

Within OART, we placed our initial efforts, as far as studies went, in support of manned planetary missions; but, recently, a significant portion of our study efforts has been involved in examining the needs of the unmanned program. At this time, we feel that more emphasis should be placed on the

unmanned missions to obtain the information we feel is critically needed to support all the development, both manned and unmanned, in the planetary area. We consider the manned mission studies as still in an exploratory stage, and we feel that there are a number of new concepts that can still be investigated.

Well, in conclusion, we feel that our efforts in advancing the technology must be increased before any decision or courses of action are defined.

SPEAKER: Mr. D. Hearth, OSSA, NASA Headquarters

I am speaking for the Office of Space Sciences which has responsibility for the unmanned program and, with all these nice words said to my right, I think I will pass the collection plate now. But let me say that those of us who are concerned with the scientific exploration of space have a vital interest in the possibilities of manned planetary missions. As a matter of fact I think that we all firmly believe that these will occur.

We are interested for two reasons: one is, of course we want to be sure that our unmanned program is oriented to provide the most information possible which is required to support the manned effort. This is not an idle statement, believe me. We are extremely anxious for our Mariner program and for our Voyager program to obtain data to help define the manned missions requirements. I will have a few more words to say on this in a moment. We are also interested in manned planetary missions, and there is one thing that I think that Mr. Gray's studies could include more of; namely, what the man is going to do when he gets there. We are interested in the man because we think man can be very useful as a scientific observer and to perform experiments that cannot be performed in any other way.

The conference has been an extremely interesting one for me and I do not intend to compliment you any more, because I agree with the other speakers. The studies are extermely interesting, and I encourage you to continue with this sort of activity.

I would like to review, very briefly, five factors which I am sure we are all aware of, and which have been touched on by the other speakers. But these are the five that came to my mind when I found out about two days ago that I was going to be on this panel. One is the problem associated with the dollars, public support, and congressional support of a manned endeavor such

as this; and the important factor that has to be recognized - the phasing in of this effort with current porgrams. The uncertainty of the space environment is certainly an important element which I am sure you all recognize. Also, the uncertainty we have relative to the levels of radiation, the micrometeorite intensity, and so on. Of course, I must mention the uncertainty of man's reaction to the space environment, which we do not know today. And, then, of course, there are the unknowns of the planet itself.

I enjoyed Mr. Syvertson's comment yesterday relative to the Mars atmosphere. He made it sound as if we are a bunch of culprits in space science by not having pinned down the atmosphere better before this. I think that what has happened to the surface density of Mars over the past year is a good example of the kind of surprises we are in for. I think any of us who today think that we know what is going to be up there is frankly being quite naive. We are in for a lot of surprises and the fact that the surface pressure on Mars has decreased by a factor of 10 in the past year is an example of this. And, of course, there are a number of other things in addition to atmosphere. The fifth point that we should recognize is that today our unmanned program is very meager and, consequently, the collection plate. It actually represents only about one percent of NASA's total budget in FY 64 and FY 65. And this is an area which, if we are going to have a manned planetary mission, has to be definitely increased.

Well, with these five factors in mind, I think there are three things we should be doing. One is to, as soon as possible, develop a rapid buildup in our unmanned program, and we should be directing this unmanned program to obtain information not only for science, but information required for the manned trip. And believe me, these are not mutually exclusive. They go together to a large extent. I also think that, before a manned mission can take place, we must have a manned space station. The reasons for this are obvious.

Finally, I think that the studies we have heard here today, and these types of studies, should definitely continue. I would suggest that they be oriented a little more to take into account the kinds of surprises we might encounter, so that when we measure these surprises, you will be able to update your system. I suggest that they be used to help guide the unmanned program in terms of the needs and requirements. And, of course, they should be used to pace your advanced development efforts.

In conclusion, let me again say that those of you who would like to contribute to the Mariner and Voyager programs are welcome to do so.

SPEAKER: Mr. M. A. Faget, MSC

I would like to say that I am very enthusiastic about interplanetary flight. It is my hope that we do not fool around, and do it after I am dead. So, I am going to dedicate myself to trying to avoid that. It is necessary, if we want to study manned interplanetary flight, that we look into the future. Let us examine, say, the next five years. Let us assume that a decision is going to be made in that period. In the reality of what we know today, it is pretty obvious how well off we can be five years from now. In the next five years we will at best have developed the Saturn V vehicle and the Apollo missions. We will also possibly have developed either a Gemini or an Apollo version of some extended orbital laboratory type of vehicle.

Any thought that we would be beyond this development during these five years is unrealistic because it is a bigger program than any of these things and because the gestation period is obviously longer than that.

So, when I view it this way, it is not too difficult for me to see what we should consider as the next step for manned programs. First, the launch vehicle should be considered probably as a Saturn V class launch vehicle, Or, for more capability, several Saturn V's rendezvoused in orbit. This apparently means that if you want to talk about a planetary mission, it must be a flyby mission.

There is another reason that I say this and that is because we will have developed an Apollo spacecraft. This is a three-man spacecraft. It is a 10,000-pound vehicle, so it would seem also that our capability will be in the 10-to 20-thousand-pound spacecraft vehicle. So we should, I think, consider a flyby version - not of Apollo, but of that class of vehicle. Not a large vehicle, but something not much bigger than Apollo as far as the reentry vehicle is concerned. We also, of course, agree with Mr. Hearth, that we should go ahead with the space station at the same time.

Now, if we pursue these courses, we can ask ourselves, what do we get? The first thing we will get out of this is that from the point of view of learning more about the space environment, we will have moved ahead as rapidly as possible toward an interplanetary mission. This is assuming, of course, that the unmanned program goes as we hope.

The other thing we will get is that we will have moved the technology forward as rapidly as possible, unless somebody wants to triple or quadruple the NASA budget, which nobody seems to think will happen. I think the second

reason is probably the most important if we undertake a flyby mission as soon as possible: this simply means that we really have to face the problems of man flying out to interplanetary distances. We have to face the problem of creating systems that will be reliable enough to work for the duration of this class of mission. This would include, of course, also knowing enough about the space environment, say, of the meteoroid environment. And I think these are the keys to interplanetary flight.

If we wait till 1980 or some time in the future and decide to go all the way and make a landing, then we are going to over-design the system, or we will probably fail. I think we have to undertake a program that will force the technology, otherwise we will not get there in my lifetime, and I do not want to do that. Thank you.

SPEAKER: Dr. W. Von Braun, MSFC

I would like to use my five minutes to address myself to the problems of how one can get a program for manned planetary exploration started. I presume that all of you here in this room are for such a program, otherwise you probably would not be here.

Now, we got a little inkling from our association with the Apollo program as to what ingredients apparently are necessary to get a program of such magnitude rolling. And it seems to be pretty clear to me that, for the time being, our national space flight resources are committed so completely to our existing manned space flight projects, as well as space science projects, that we will probably have to wait at least a few more years until we are over the hump of the expenditure curves with these existing commitments. But, I think this is only one of the constraints that we have to consider: I think the history of technology is full of examples that prove rather convincingly that bit technological programs can only be started successfully if and when all the necessary ingredients are really together.

There have been rather pathetic examples in history where things were started too early with the result that they got stuck halfway. For example, there were proposals to establish trans-Atlantic aviation with the help of floating islands in the ocean using non-pressurized propeller aircraft to fly them from continent to continent with one, or several, stops on such islands. Well, for some reason or other, although the technical feasibility of such a system could not clearly be proved as fantastic, nor, could it be convincingly

proved that it was not feasible, it somehow did not catch on. It took other ingredients, such as pressurized cabins, jet propulsion, and flying capability above the weather, to really make trans-Atlantic aviation attractive enough for the public to buy.

I think we are very much in the same situation with interplanetary space projects. One could, conceivably, build a large manned expedition to our neighboring planets, even on the technology that we have today. But whether this would be attractive, and would be politically feasible, is another question. I personally have my doubts, from what I have been through in Apollo.

On the other hand, we must not overlook the fact that, once a program does get established, such as the Apollo and Saturn projects, you build up a certain amount of momentum which seems to be almost as difficult to stop as it was to get it established. And this has something to do with the fact that you are developing industries and facilities around these programs that show strong signs of an interest to survive. They build up political influences through Congressional representation. They develop their own lobbies, that try to sell future projects just in order to keep the factory doors open, and these elements undoubtedly help us to continue.

Just imagine what will happen if, after the first man has placed a sign on the Moon, "Kilroy was here" and we have all these huge facilities that were developed in connection with Apollo and no work, no assignments. I think this is probably one of the strongest forces to create follow-on programs, and that they will undoubtedly include planetary ventures.

I would also say again, rather in the philosophical vein, that there seems to be one pattern established in human affairs. It seems to have something to do with the make-up of the human mind. That is, that once the time has come for a new idea, and all the new ideas and ingredients are available to make it feasible, it seems to be virtually impossible to stop this idea. And, for this reason, I am very positive that there will be interplanetary exploration.

Now, how about the details of implementation. There is, of course, no doubt in my mind that we have to use all the unmanned techniques that we are presently developing to learn as much about the planets as possible. And I fully agree with Max Faget, it would be absolutely silly to go into the design of a manned planetary expedition without really knowing rather exactly what environmental conditions to expect. There are enough other things that we cannot find out this way that still warrant a manned expedition to follow. So, I think whatever we can do to use our existing launch vehicles and our spacecraft techniques to develop Mariners and Voyagers, and all follow-on type of vehicles, should be done.

I also agree with Mr. Faget, that our next step should be manned flyby missions. I think flyby missions, particularly flybys involving landing probes or, if you prefer, guided missiles with reentry noses that you can shoot into the planetary atmosphere, would be invaluable. These probes would descend by parachute, or other means, to the planet's surface, collecting information during the descent and additional information after arrival, which they would radio back to the flyby manned vehicle to relay to Earth. One such flight, giving us more information on what to expect, say on the surface of Mars, will be extremely valuable in helping us in laying out the equipment for the landing operation itself that would follow the first flyby flight. I am also inclined to believe that our first manned planetary flyby missions should be based on the Saturn V as the basic Earth-to-orbit carrier. The reason is that, once the production of this vehicle has been established and a certain reliability record has been built up, this will be a vehicle that will be rather easy to get. In other words, all you really need is money, and everybody else will be for it. And, of course, by that time, orbital rendezvous should be old hat.

I would like to spend just one more minute to talk a bit about the role of nuclear propulsion in interplanetary flight. I am convinced that after the flyby phase of manned interplanetary trips has been completed and we are seriously considering interplanetary voyages involving landings on planets, that we have to think substantially bigger than in Apollo terms. We are not talking here about an expedition of three men going on a round trip to the Moon, lasting 10 days, but we should be thinking in terms of at least a dozen people going out on a trip for over a year. Because, with these long durations, there are certain things that must be included. I would not want to go on a one-year trip without a dentist being around, or somebody at least to pull a tooth. So the expedition will, undoubtedly, grow in size. All, this, in my opinion, when added to the bigger Saturn V requirements, will call for nuclear power, at least for the deep space portion of the trip.

In this connection, I would like to say that I was quite impressed by some presentations we had recently on the Project Orion. Let me confess that for years I have been very skeptical of this project. In fact, my first reaction to the presentation was that this is downright fantastic. But these fellows at General Dynamics have really made great strides in recent years, making a rather attractive suggestion out of this rather fabulous proposal. In case you do not know, Orion involves the exploding of fission bombs, behind the vehicle, and a piston acts as a kind of a shock absorber to transfer the

impulse to the vehicle itself. The interesting aspect is that, here is a propulsion system that can produce both high thrusts, like a chemical rocket, and high specific impulses, far higher, than, for example, the Rover systems. And, it also does not create a serious radiation hazard problem, surprisingly enough. The only problem is that it seems to fall in the category of nuclear testing in outer space, and seems to be in conflict with some international treaties. But, maybe that can be cleared away in due time. At any rate, this is a system that, in my opinion, deserves some attention. It has one feature that seems to be particularly attractive for planetary voyages and that is, it permits us to use some indigenous material, even for propulsion. This can be any kind of dust. You name it. It can be used.

Well, as a parting remark, let me just thank you, as Director of the Marshall Center, for your being here. I am very sorry that I could not attend this whole meeting, but we had a little affair (launch of SA-5) down at the Cape yesterday that prevented me from being with you. Thank you.

SPEAKER: Dr. H. H. Koelle, MSFC

I think we can read three messages out of the panel discussion; that is what I got out of it. The first one is - let us continue to communicate, and, as President Johnson would say, - "Let's reason together." The second message I read out of this discussion is, Let us take up a collection for unmanned space flight, and I think almost everybody commented on this. We need simply more information, and preferably hot information. So--, let us say this was a conclusion of the panel. And the third message, I think also came through pretty clearly: the flyby mission, whether we like it or not, seems to be a good beginning and, therefore, we should study it further. I hope that Bill Fleming has taken note, and will pass this on to Dr. Seamans.

I do not think we have much time for questions from the floor, but according to my clock, we have five more minutes, so to give the audience a chance to participate, let us ask two or three pointed questions. So, whoever has a pointed question to any of the panel members, please raise his hand.

COMMENT FROM THE FLOOR

I hate to bring up details after the broad sketches of principles and broad thinking of the panel members, but I have been very concerned in listening to the proposals and the studies of the last few days, to have been unable to detect any concern whatsoever over the possibility of back-contamination from the planets. It is established NASA policy that we will not contaminate the planets with terrestrial organisms.

I am sure that it will be an established policy that we will not permit back-contamination and, should it not be, just the considerations of personal safety and well-being of the creature and humanity on Earth will enforce this. I sincerely hope that these considerations will be given due study in any future work.

REPLY: Dr. H. H. Koelle, MSFC

I am sure it will, but I think first things first. The problem is, first you have to find a way to get to the planet and back before you worry about what you bring back. I think we will come to that point and people will think about it, but you cannot start on that end. Does anybody have a question?

COMMENT FROM THE FLOOR

I heard several comments relative to flyby missions, and I have heard comments relative to the space station. I would like to hear the panel's comments on where in the total cycle they would like to see the flyby mission.

REPLY: Mr. W. A. Fleming, NASA Headquarters

Well, I think that we can rather definitely state that a manned space station is a step toward manned planetary flight. There are two elements in the operation of a manned space station that I think are crucial to manned planetary flight. One that I mentioned before - experience with men in space for long periods of time - just the physical and medical effects; and, second, experience in operating components and systems in a weightless and vacuum environment for long periods of time. Both are essential development steps in planetary exploration.

QUESTION FROM THE FLOOR

The first question on the list is one everybody seems to avoid. Why do we really want to go to Mars, other than if we have the vehicles standing around with nothing to do?

REPLY: Dr. H. H. Koelle, MSFC

I think Dr. von Braun tried to answer. He said there would be so much momentum in the technology we have that we perhaps have no choice but to go.

SPEAKER: Mr. M. A. Faget, MSC

The difference between a man who has a lot of money and a man who does not have a lot of money is sort of a standard of living. Now the man can be a playboy and he can spend all his extra money on going out to night clubs every evening and have a good time with his money, and there is nothing wrong with it. The country could do this, and to a certain extent it does. But also, there are some people who either have more money or are earning more money than their standard of living really requires. They have an excess of wealth to some degree or another. I hope our country is in that category.

Now, a lot of people, and, in fact, they are considered the ones that do the right things with their excess wealth, spend a good bit of their time improving themselves. They go to the right kinds of theaters, read books, and spend time on higher education and things like that. I think that this program short of improves our country in somewhat the same manner. I think it is a good thing, since our country has more capability to produce than is required to feed everybody in this country.

When you really get down to it, what are we really going to spend our extra money for, driving down the highways faster, drinking more? Or are we going to spend it on some sort of organized activity that philosophically seems to improve our country? I think it is unquestionable that the space program, and this applies to every bit of the NASA program, does tend to improve the country.

SPEAKER: Dr. H. H. Koelle, MSFC

Very good, Max. You had the last word, and I like it. I would like to thank the panel for their attendance, for their very good work, and their wisdom, The meeting is adjourned. Thank you.

NASA TM X-53049 APPROVAL June 12, 1964

PROCEEDINGS OF THE SYMPOSIUM ON MANNED PLANETARY MISSIONS 1963/1964 STATUS

By Future Projects Office Staff

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified. This report also has been reviewed and approved for technical accuracy.

H. H. Koelle

H. H. KOELLE
Director, Future Projects Office

DISTRIBUTION

DIR
- Dr. von Braun

R-FP
- Dr. H. H. Koelle
- Dr. H. O. Ruppe
- L. T. Spears
- J. N. Smith
- Dr. Willy Ley
- M. A. Page
- W. G. Huber
- L. H. Ball
- J. W. Carter
- V. Gradecak
- G. J. Detko
- J. L. Sanders
- G. R. Woodcock
- D. Paul
- W. Payne (20)

R-COMP
- Dr. R. F. Arenstorf

R-P&VE-A
- C. L. Barker, Jr.
- C. D. Swanson

R-P&VE-AA
- L. R. Cohen

R-P&VE-AB
- J. C. Raines
- F. San Jaun
- J. Rains

R-P&VE-ABO
- W. E. Thompson

R-P&VE-AL
- R. Gingras

R-P&VE-ALP
- C. E. DeSactis

R-P&VE-AN
- W. Y. Jordan
- J. W. Heyer
- J. C. Whiton

R-P&VE-ANA
- R. J. Harris

R-P&VE-ANS
- D. R. Saxton

R-P&VE-AV
- J. P. Gardner

R-P&VE-AVC
- M. Cash
- V. Hanes

R-P&VE-AVP
- R. R. Belew

R-P&VE-T
- A. J. Finzel
- A. R. Bond

R-P&VE-SAA
- R. G. Crawford

R-P&VE-VAE
- D. H. Ogden

R-P&VE-VAS
- B. J. Clingman

R-P&VE-N
- A. V. Clark

R-AERO-DP
- H. F. Thomae
- J. von Puttkamer
- R. D. Scott
- E. T. Deaton
- A. C. Young

R-AERO-P
- R. Schmidt

R-AERO-FOM
- M. K. Williams

R-AERO-S
- T. D. Dickerson

DISTRIBUTION

EX-P
R. M. Condrey
J. Waite
EX-T
J. A. Budaz
W. A. Hagen

R-AERO-SP
H. Schaefer
R-AERO-YS
R. E. Smith
W. L. Roberts
R. A. Jamieson

R-ASTR-A
F. P. Daniel
J. W. Cole
R-ASTR-AC
C. R. MacCrone
G. W. Neal

R-ME-X
L. H. Berge

R-RP-DIR
Dr. E. Stuhlinger
R-RP-P
O. K. Hudson
E. A. Mechtly
R-RP-RN
Dr. R. D. Shelton
R-RP
J. C. King

R-QUAL-J
J. P. Bates
F. Dhom
G. J. Hofman
R-QUAL-PF
J. P. Crisler

I-RM
R. J. Stein
I-RM-E
D. J. Winslow

I-V-SIC
L. G. Thompson

F&D-M
H. Luehrsen
F&D-RT
I. Hirshburg

R-TEST-C
W. J. Halbrooks
R-TEST-SP
J. W. Williams

DEP-T

CC-P

HME-P

MS-H

MS-IP

MS-IPL (8)

Scientific and Technical Information Facility (25)
Attn: NASA Representative (S-AK/RKT)
P. O. Box 5700
Bethesda, Maryland

EXTERNAL DISTRIBUTION

NASA Headquarters
National Aeronautics and Space Administration
Washington, D.C.
Attn: J. W. Stone, Code MTC
J. W. Haughey, Code SB
D. R. Lord, Code MT-2
Dr. S. Ross, Code MT-2
R. Livingston, Code MT-2
R. W. Gillespie, Code MTG
W. L. Haberman, Code MTG
Dr. F. G. Casal, Code MTG
M. Gruber, Code MTG
W. J. Hamon, Code MTG
N. B. Cohen, Code MTG
R. J. Wisniewski, Code RV-1
P. A. Cerreta, Code RV-1
C. E. Pontious, Code REG
R. F. Trapp, Code RBM
D. P. Hearth, Code SL
P. G. Johnson, Code NPO
W. A. Fleming, Code PT
A. M. Nelson, Code PT
J. O. Spriggs, Code SA

National Aeronautics and Space Council
Attn: T. E. Dolan
The Executive Office of the President
Washington, D. C.

National Aeronautics and Space Administration
Ames Research Center
Moffett Field, California
Mountain View, California
Attn: H. Hornby
C. A. Syvertson

National Aeronautics and Space Administration
Goddard Space Flight Center
Greenbelt, Maryland
Attn: L. E. Richtmyer
Dr. F. O. VonBun, Code 513

National Aeronautics and Space Administration
Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena 3, California
Attn: J. S. Martin
P. N. Haurlan

National Aeronautics and Space Administration
Langley Research Center
Langley Field, Virginia
Hampton, Virginia
Attn: C. H. McLellan, Code 31.002
E. B. Pritchard, Code 31.002
W. H. Phillips, Code 14.000

National Aeronautics and Space Administration
Manned Spacecraft Center
Houston, Texas
Attn: H. E. Gartrell
C. R. Darwin, Code ESTO
Z. H. Byrns, Code ESTO
E. J. Gavlann
G. Xenakis, Code EST
C. E. Humphries, Code ESTP
T. W. Briggs, Code PAE
W. L. Davidson, Code OFO
R. B. Merrifield, Code AMA
J. W. Kiker, Code ESDM
G. Strouhal, Code ESDS
M. Faget, Code E
W. E. Stoney, Code EST

AC Spark Plug Electronic Division, General Motors
Holiday Office Center, H
Huntsville, Alabama
Attn: H. Dusek
G. Kleimhesselink

Aerojet General Corporation
Holiday Office Center,
Huntsville, Alabama
Attn: R. E. Toliver
R. D. Waldo
A. F. Felelbush
H. L. Johnson
W. C. Beckwith
K. P. Conrad
A. Muraszew

American Machine and Foundry Co.
Holiday Office Center
Huntsville, Alabama
Attn: Arthur L. Givson
R. N. Baker
A. L. Gibson

AVCO Corporation
Holiday Office Center
Huntsville, Alabama
Attn: H. J. Sorenson
P. R. Swan
R. D. Enzmann
W. T. Knowe
W. D. Kuhns
J. A. Dodge
R. Bernert

Boeing Company
Sahara Office Center
Huntsville, Alabama
Attn: T. P. Snow
G. V. J. Yuscavage, HIC Bldg.
W. Robinson
J. G. Brunk

Boeing Company
Sahara Office Center
Huntsville, Alabama
Attn: D. B. Jacobs, Huntsville
R. Hardy
W. M. Howard
T. P. Snow
J. A. Stern

Chrysler Corporation
Post Office Box 857
Huntsville, Alabama
Attn: Howard Blood
H. Genthner
G. Hopkins
P. E. Beck
P. F. Fahey, Jr
L. J. Gellman
J. S. Hamilton
D. A. Veith
M. L. Scott
F. A. Ayer

Douglas Aircraft Company
Holiday Office Center
Huntsville, Alabama
Attn: Peter Browning
J. P. Fredricks
J. D. MacGregor
W. E. Matheson
G. V. Butler
R. W. Hallet
G. C. Goldbaum
R. L. Johnson
S. Gronich
R. L. Gervais
W. H. Kuhlman
L. M. McKay
E. S. Mills
R. T. Kendall
D. N. Lascody

General Dynamics/Astronautics
Holiday Office Center
Huntsville, Alabama
Attn: Dr. R. N. Austin
T. B. Taylor
W. S. Evans
W. C. Strobl
B. L. Brown
S. P. Horio
B. H. Oman
D. H. Robey
R. E. Mannion
Dr. T. S. Webb
B. A. Kress
Dr. V. A. Lee
T. E. Peace
C. B. Moore
R. N. Oliver
J. S. McKnight
T. P. Robinson
R. A. Gorrell
Dr. K. A. Ehricke
G. B. Paddock
L. M. Nelson
J. W. Evans
W. J. Couchois
F. A. Manley
P. R. Shipps

General Electric
Holiday Office Center
Huntsville, Alabama
Attn: S. Calder
W. Raishel/Pennsylvania
J. W. Larson/ Pennsylvania
D. M. Cole/Pennsylvania
C. A. Taylor/Pennsylvania
C. C. Adams/Pennsylvania
T. Widmer/Pennsylvania
J. W. McKee/California

General Electric
Holiday Office Center
Huntsville, Alabama
Attn: M. B. Lees/Washington, D. C.
C. L. Smith/Washington, D.C.
C. B. Appleman/ Texas
S. Calder/Huntsville (Local Representative)
H. Brown/Ohio
D. Cochran/Ohio

Hughes Aircraft
2806 Memorial Parkway, S.W.
Huntsville, Alabama
Attn: J. M. Miner
R. N. Olson

ITT Federal Laboratories
Holiday Office Center
Huntsville, Alabama
Attn: Mr. Charles A. Thomason
P. Zilczer,
S. Veniar
C. Besselsen

Ling-Temco Vought Corp.
Holiday Office Center
Huntsville, Alabama
Attn: Jin Lott
A. M. Mayo
J. S. Buchan

Lockheed Aircraft Corporation
Holiday Office Center
Huntsville, Alabama
Attn: Mr. Bill Hussman
J. F. McLaughlin
C. J. Liddell, Jr.
B. P. Martin
R. V. Ragsac
Dr. D. J. Shapland
R. L. Nelson
F. W. Husman

Malaker Labs
702 Oak Park Drive, N. E..
Huntsville, Alabama
Attn: T. H. Durr
Dr. S. Malaker

Martin Company
Sahara Office Center
Huntsville, Alabama
Attn: Stanley W. Connelly
J. Jensen/Baltimore
N. J. Paradiso/Baltimore
D. J. Markarian/ Baltimore
C. DeMoraes/Baltimore
R. L. Cox, Jr. /Baltimore
W. F. Furth/Baltimore
Dr. R. Novasad/Denver
R. J. Greenspun/Denver
P. H. Ney/Denver
E. C. Payson/Denver
W. H. Clohessy/Denver
D. G. Stechert/Denver
R. B. Bowersox/Denver
E. H. Dingman/Denver
J. W. Tyndall/Denver
P. C. Thayer/Denver
M. E. Capehart/Denver
Dr. G. Morgenthaler/Denver

Northrop Corporation
3322 Memorial Parkway, S.
Huntsville, Alabama
Attn: O. D. Meyers
M. Hesse
H. W. Bixby
R. F. Kinzy
R. N. Worth
D. Waltz
O. D. Mylers
E. J. Holmbeck
Bob Bruce

Philco Corporation
Suite A, Lane Plaza
250 Governors Drive, S. E.
Huntsville, Alabama
Attn: H. J. Burton
Dr. F. P. Dixon
J. S. Hanrahan
R. L. Pons

Pratt & Whitney
Holiday Office Center
Huntsville, Alabama
Attn: John Campbell
E. L. Quinn
L. M. Moran
W. H. Dunham
F. R. Tomlinson

Rand Corporation
1700 Main Street
Santa Monica, California
Attn: H. L. Shulman
A. Leonard
D. Trapp
B. Pinkel
W. Krase

Raytheon
3322 Memorial Parkway, S.
Huntsville, Alabama
Attn: O. M. Lowery
A. C. Doskocil
C. L. Ashley

Radio Corporation of America
Holiday Office Center
Huntsville, Alabama
Attn: A. C. McDermott
E. P. Rush

EXTERNAL DISTRIBUTION (CONT'D)

Melpar, Inc
3000 Arlington Blvd.
Falls Church, Virginia
Attn: J. A. Ryan

North American Aviation
Holiday Office Center
Huntsville, Alabama
Attn: J. W. Scheuch
A. L. Jones
W. V. McRae
W. Hissim
R. L. Powell
C. A. Butler
M. J. Schuster
L. G. Vargo

Pratt & Whitney
Holiday Office Center
Huntsville, Alabama
Attn: John G. Campbell
J. P. Carstens
A. A. LeShane

United Technology Corp.
Sands Office Center
Huntsville, Alabama
Attn: Robert Brazil
D. G. Wright

North American Aviation
Holiday Office Center
Huntsville, Alabama
Attn: N. W. Langford, Rocketdyne
A. L. Negro, Rocketdyne (Canoga Park)
V. R. Larson, Rocketdyne

Space General Corporation
Holiday Office Center
Huntsville, Alabama
Attn: Paul Sorracco
A. B. Hazard
C. Lysdale

Space Technology Lab., Inc.
Sahara Office Center
Huntsville, Alabama
Attn: Steve Harris
E. Dean
R. L. Sohn

Municipal Utilities Building
Huntsville, Alabama
Attn: J. J. Weaver, Sylvania Electronic Systems

Westinghouse Electric Corp.
Holiday Office Center
Huntsville, Alabama
Attn: W. E. Douglass
G. L. May
G. Sege
W. M. Jacobi
G. S. Axelby
H. C. Moses
C. R. Flee